1. Introduction

This report contains two integral parts, the first of which being the preliminary sizing of a UAV. This is completed using data from reference aircraft as well as the given requirements of the aircraft. The second part will center around sizing and selection of propulsion for the UAV. Chapter 2 will explore the calculations behind estimating the takeoff weight. The following chapter 3 will explore the creation of the aircraft T/W – W/S graph. Chapter 4 will detail the generation of the drag polar. Chapter 5 will discuss the optimal takeoff weight as a function of the payload range diagram.

Part two of this report details the proposal for a propulsion system for the UAV to make an order to motor manufacturers. Initially in chapter 5 the required thrust for the aircraft at several weight conditions will be calculated. Next chapter 6 details a selection of engine type, weight, and dimensions. Following this in chapter 7 the engine type that most closely meets the needs of our requirements will be selected and scaled to better fit the needs of the aircraft. Chapter 8 will detail optimization of the thermodynamic cycle of the motor and it energy expenditure. In this chapter validation of previously selected design values will also be performed.

1. Determining the takeoff weight

To determine the takeoff weight, the different parts of the overall aircraft weight have to be identified. The takeoff weight often consists of operational empty weight, fuel weight, and payload weight. Because our aircraft is battery powered UAV, fuel weight is a fixed value, consisting solely of the weight of onboard batteries. Thus, in this chapter fuel weight will be calculated using 2 different static battery conditions that were briefly described in assignment 1. The battery weight is inversely proportional to the payload weight thus the 2 separate battery conditions will also entail 2 separate payload conditions as a result. At the end of this chapter the total takeoff weight will be calculated using the equation below.

W TO = W PL + WF + W OE Eq 2.1

* 1. Payload

As already calculated in the last assignment, the payload will be calculated as the total weight of all components of the sensor package deployed with the UAV. The likely payload weight of the aircraft based on similar reference aircraft will be 4.9985879 kg.. However, due to the previously mentioned ability for the aircraft to have battery space exchanged for additional payload space a secondary large payload arrangement can also be made for the aircraft. This secondary payload configuration would see one of the two 8.82 lbs. batteries removed sacrificing aircraft range for increased payload leaving us with a second payload weight of 9.9971758 kg.

|  |  |
| --- | --- |
| Payload Weight 1: | 4.9985879 kg. |
| Payload Weight 2: | 9.9971758 kg. |

* 1. Batteries

Unlike hydrocarbon powered aircraft where total fuel weight is the combined weight of both the actual used fuel and the reserve fuel electric aircraft fuel weight consists solely of the weight of the aircraft battery pack. These battery packs see negligible changes in weight during the totality of flight time thus their weight can be considered a static value. However, as mentioned in Section 2.1 the battery weight is tied to the payload configuration within the LAHE UAV. Since the LAHE UAV has two payload weight configurations it will also have two battery configurations, using 1 and 2 40Ah batteries, each weighing 4.000685 kg. As mentioned earlier, in a standard hydrocarbon fueled aircraft used fuel is a fraction of the takeoff weight and typically consists of three parts. Though this aircraft will not see changes in fuel weight over time as it flies it still will use variable amounts of battery charge in different parts of its flight. Thus, calculations for normal, cruise, and loiter flight fractions will still be calculated in order to understand aircraft energy consumption even though the battery weight of the aircraft will remain static throughout the operation of the aircraft.

|  |  |
| --- | --- |
| Battery Weight 1: | 4.000685 kg. |
| Battery Weight 2: | 8.00137 kg. |

* + 1. Normal Flight Fraction

The normal flight fraction typically consists of the non-energy use intensive stages of aircraft flight. The individual components of the flight fraction are listed in the table below. Using the data provided in the lecture notes on page 32 the values were found. (Since there is no direct equivalent for UAV values for homebuilt aircraft were used for calculations.) It is important to note that values for the descent and climb phases were multiplied by two since in Assignment 1 the aircraft mission profile details a small descent and climb when entering the loitering stage as well as a small descent and climb for the diversion cruise both of which must be accounted for in the normal flight fraction calculations.

|  |  |
| --- | --- |
| Engine start and warm-up | 0.998 |
| Taxi | 0.998 |
| Take off | 0.998 |
| Climb | 0.995 |
| Descend | 0.995 |
| Landing taxi shutdown | 0.995 |
| Normal flight, with adjustment for additional climbs an descents | 0.9694 |

* + 1. Cruise Fraction

To calculate the cruise fraction, the following equation must be used.

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Description automatically generated

This equation can be further simplified in the following manner.

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Description automatically generated

W7/W4 represents the power consumed while in cruise, R is the cruise range, g is the gravitational constant (9.80665 m/s2), V is the cruise-speed and L/D is the lift-drag ratio while in cruise. From our requirements R = 6.21 mi or 9.994206 km, V = 49.21 ft/s or 14.999208 m/s , the specific fuel consumption during cruise Cp will be 0.4 lb/lb/hr or 0.000011 kg/Ns in SI. Lastly for L/D ratio our group chose to use a value of 20 as our aircraft is designed to function in similar fashion to a glider thus requiring a higher ratio. However, it is still a prop powered aircraft and limitations exist on our ability to machine an airfoil that could produce an incredibly high L/D ratio thus the value of 20 was chosen as an achievable target ratio.

After putting all these values into the equation, it can be calculated that the fraction used during the cruises is 0.996413.

* + 1. Loiter Fraction

In a similar manner the loiter fraction can be calculated with the equation below. E is for aircraft endurance, g is still the same gravitational constant, cp is the constant fuel consumption during loiter, and L/D the lift drag ratio in loiter.

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Description automatically generated

The aircraft endurance as written in the aircraft requirements in Assignment 1 is 3 hours. The gravitational constant is still 9.80665 m/s2 and the cj will remain 0.000011 kg/Ns since the aircraft AOA is incredibly similar to that in the cruise fraction resulting in minimal difference in specific fuel consumption between the two fractions. L/D ration will be slightly higher at 22 as in loiter the aircraft will be to take full advantage of its large glider-based wingspan.

Using this equation and the listed variables the loiter fraction of the aircraft can be calculated to be 0.9484.

* + 1. Total Battery Fraction

As stated earlier, the use of battery power on the aircraft will not change the weight of the batteries at any point during the duration of flight. Thus, the battery weight will remain static at one of the two possible battery configurations weights. These values instead serve as an exercise to better understand power consumption for our electric aircraft and how much of our battery power will be lost during power use intensive parts of the mission profile.

* 1. Operational Empty Weight

The Woe can be calculated using the following equation. This equation sums the aircraft empty weight, the weight of trapped fuel and oil, as well as the weight of the crew.

A black and white symbols

Description automatically generated with medium confidence

As an electric UAV our aircraft has both a Wtfo and Wcrew of 0, leading to Woe being equal to We. We is a fraction of the aircraft empty take off weight and can be found through plotting a graph of the Maximum takeoff weight versus empty weight of our reference aircraft. Using the trendline found in this graph an equation to find the proportion of Wto equal to We can be derived.

From this graph the formula of the empty weight as a function of the takeoff weight was derived and can be seen below.

We = 0.7276Wto+4.4274

* 1. Total Weight Estimation

Using the previous calculated values and equations Wto can be calculated as,

Wto=4.9985879+8.0920879+0.7276Wto+4.427

This gives us a Wto of 64.3086kg or 141.7761943lbs

This value, however, represents the takeoff weight required when operating simultaneously at both maximum range and endurance as well as maximum payload size. These are not realistic conditions as our initial specifications do not require all of these factors to lie at a maximum value. As a result, a much smaller Wto of 31.6 kg and We of 25 kg were selected for our aircraft.

1. Determining the T/W-W/S Diagram.
   1. Stall Sizing

W/S can be calculated using the equation below.

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Stall speed Vs and Max Coefficient of Lift Clmax are currently unknown but can be found using estimations derived from reference data.

|  |  |  |
| --- | --- | --- |
|  | Clmax | Vstall |
| Configuration | [-] | m/s |
| Clean | 1.2 | 12.7 |
| Take Off | 2.0 | 9.836 |
| Landing | 2.1 | 9.599 |

W/S Clean: 118.54815 N/m^2

W/S Landing: 118.516 N/m^2

* 1. Take off sizing

A variety of factors are at play during the takeoff of an aircraft, these various factors are represented numerically using the take off parameter which is derived from the equation below.



The variables W/S and W/P are both derived from aircraft reference data. Clmax is not available in design specifications for most aircraft. For most drone aircraft Clmax lies between 1.6 and 2.0. Using a graph of the TOP and the takeoff length of the reference aircraft TOP for our aircraft can be found using the equation created by the trendline in the graph below.

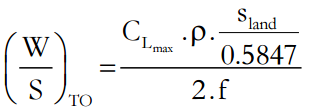
By putting our desired take-off length into the equation created by the trendline we can find two values \_\_\_\_\_ and \_\_\_\_\_\_ the first of which being the one we will use moving forward. With this data.

* 1. Landing Sizing

Using the equation below we can derive the landing length of our aircraft and references.



Using data derived from previous statistical analysis Wo/Wl or f can be found to be \_\_\_\_.



With the above equation the vertical lines of the W/S diagram can be calculated by varying the Clmax at landing.

* 1. Drag Polar

To define and align with the climb requirements the Drag Polar of the aircraft must be calculated. This can be done using the equation below.

A math symbols with a plus and a positive symbol

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Cd0 is equal to F/S with F being the parasitic drag and S being the wing area. These two variables can be found using the two equations listed below.

A math equations on a white background

Description automatically generated

Swet must first be found using the second equation using the Wto or aircraft takeoff weight off 31.6 kg and the previously calculated values of c equal to 3 and d equal to 0.5 common for fixed wing drone aircraft. The values of a and b are found from previously calculated data dependent on the friction coefficient. With a friction coefficient of \_\_\_\_\_ a is then equal to \_\_\_\_ and b to \_\_\_\_\_. With the value of Swet the parasitic drag can then be calculated. With these two values the Cdo can be calculated to be 0.35. This value makes sense as our aircraft lies in the middle of the statistical range of our reference data. We also must find the value used for Oswald Efficiency or e. Ultimately, we decided that a value of 0.5 would best suit our aircraft. Since Cd and Cl change significantly based on the mission stage of an aircraft the results of the equation are displayed in the graphic below showing the drag polar at take-off, clean, approach, and landing conditions.

* 1. Climb performance requirements

In order to find climb performance requirements certification regulations must be consulted. For this aircraft the climb was determined to be \_\_\_\_\_\_. While typically it would be prudent to further modify our calculations for CL and Cd to maximize the climb of the aircraft the lack of control surfaces on the LAHE UAV means that the values of Cd and CL are static and cannot be altered by maximizing the use of control surfaces.

A diagram of a mathematical equation

Description automatically generated with medium confidence

With this information the climb rate data can be inserted into the W/P – W/S diagram.

* 1. Climb Gradient Requirement

The optimal climb gradient requirements are found using the max lift to drag ratio. However, as previously stated there is no maximum ratio due to the control scheme of the aircraft thus additional maximum values do not need to be calculated.

A math equation with numbers and symbols

Description automatically generated

Thus, a P/W ratio of can be found to be 0.125 , appearing as a single vertical line on the P/W – W/S diagram.

1. P/W – W/S Diagram

The P/W – W/S Diagram consists of all the calculations made in the past two chapters. Using this diagram many critical values necessary in preliminary sizing of the aircraft can be found. The diagram can be found in the appendix below.

`A value of P/W of 0.125 was chosen, along with a wing loading of 100 N/m^2

The required takeoff thrust is \_\_\_\_ N and the wing area is found to be \_\_\_\_\_ m^2.

|  |  |  |
| --- | --- | --- |
| Parameters | Value | Unit |
| Cj cruise | 0.000011 | Kg/Ns |
| Cj loiter | 0.000011 | Kg/Ns |
| Cj climb | 0.000011 | Kg/Ns |
| Wto | 31.6 | kg |
| We | 25 | kg |
| Wpl | 6.6 | kg |
| Ww |  | kg |
| S wing | 0.26334664 | m^2 |
| A | 2.7 |  |
| e |  |  |
| C/V | 0.15 | m/s |
| Rho cruise | 1.225 | Kg/m^3 |
| C/V engine failure |  |  |
| Cruise velocity |  | m/s |
| Number of engines | 1 |  |
| Cd0 | 0.35 |  |

5 Propulsion Design

In the following chapter the cruise, climb, and takeoff thrust required for flight will be calculated using the parameters found using the P/W – W/S diagram. Airworthiness requirements will also be explored in necessary depth. Only one engine is required for this aircraft.

|  |  |  |
| --- | --- | --- |
| Parameters | Value | Unit |
| Cj cruise |  |  |
| Cj loiter |  |  |
| Cj climb |  |  |
| Wto |  | kg |
| We |  | kg |
| Wpl |  | kg |
| Ww |  | kg |
| S wing |  | m^2 |
| A | 2.7 |  |
| e |  |  |
| C/V |  | m/s |
| Rho cruise |  | Kg/m^3 |
| C/V engine failure |  |  |
| Cruise velocity |  | m/s |
| Number of engines |  |  |
| Cd0 |  |  |

|  |  |  |  |
| --- | --- | --- | --- |
|  | climb | cruise | Climb engine failure |
| Cl |  |  |  |
| Cd |  |  |  |

5.1 Treq,takeoff

In the last chapter P/W was determined to be 0.125 and the takeoff weight was 31.6 kg.

Pto = (T/W)\*W =

5.2 T req, cruise

A math equation with numbers and symbols

Description automatically generated

When calculating the required thrust for climb flight the above equation from chapter 3 can be used once again to calculate it. The same lack of dynamism in the Cl and Cd of the aircraft still stands due to the lack of control surfaces. From the P/W – W/S diagram C/V can be calculated to be \_\_\_\_%. The values of Cdo, A, e, and the Wto are also found in the previously mentioned diagram. The required climb thrust can then be calculated to be \_\_\_\_\_.

Cruise speed and cruise altitude were provided in the initial aircraft requirements table. At the cruise altitude of 121.92 meters air density is 1.225 kg/m^3. We can assume that in cruise flight thrust is equal to drag thus the values for S and Cd derived from the P/W – W/S diagram can be used to calculate this thrust as well. Using the equation below we find that the required cruise thrust is \_\_\_\_.

A number and a mathematical equation

Description automatically generated with medium confidence

5.3 T req, engine failure

Due to the guiding design choices for our aircraft T req for engine failure will not be calculated. Upon engine failure the single engine design of the LAHE UAV provides no auxiliary source of propulsion and due to the high aspect ratio of the aircraft would see the craft stall almost immediately. This is not of great concern as the craft is designed to be relatively inexpensive and not carrying highly valuable or human payloads.

1. Engine Selection and modification
2. Engine Scaling of the
3. Required checks and engine improvements

In the following section, a sanity check will be performed to see if the battery capacity calculated in chapter 2 is enough for us to meet the specified range with our selected engine. Along with this a sanity check will also be performed on the fuel weight, engine weight, and payload. If needed some parameters will be changed to reach the performance requirements. All previously calculated design parameters were compiled and placed on the table below.

|  |  |  |  |
| --- | --- | --- | --- |
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8.1 Battery Weight Check

As discussed in chapter 2, since the aircraft is an electric powered aircraft the weight of the batteries serves as the aircraft fuel weight component. Another result of this design aspect is that fuel usage fractions do not have an impact on battery weight or aircraft weight overall. Thus, the only reality check that needs to be performed on the battery weight is as to whether both battery schemes listed in chapter 2 are still viable.

Our first battery configuration used a single 40ah 4.000685 kg battery, allowing for more of the aircraft weight to be dedicated to the payload. Due to lower electric efficiency than anticipated in the selected engine endurance is negatively impacted. This configuration combined with the selected motors would result in total aircraft endurance of 1 hour and 15 minutes, significantly less than that which we had originally listed in our requirements. This would also significantly lower our range to \_\_\_\_ which is unacceptable in comparison to the original design requirements.

Our second battery configuration used two 40ah 4.000685 kg batteries. This configuration would leave us with a smaller yet still totally functional 4.9985879 kg payload capacity. Due to the previously mentioned lower than anticipated electrical efficiency of the engine our endurance would sit at around 2 hours and 30 minutes of airtime. This value is still slightly below our original design specification of 3 hours of endurance however the functionality of the aircraft system is still retained at this slightly lower endurance value and thus we believe that the design can proceed further into development and that the design specifications can be adjusted accordingly. After the sanity check our previous Wb of 8.00137 kg.

* 1. Engine Weight Check

A driving factor behind the engine selection was weight. Due to the heavy nature of the aircraft batteries, an excess weight must be conserved as much as possible to ensure that a reasonable available weight is retained along with the range and endurance specifications of the aircraft. The selected engine \_\_\_\_\_\_\_\_\_ weighs in at \_\_\_\_\_\_ kg resulting in an empty weight percentage of \_\_\_\_\_\_\_% for the engine. A better value than the comparative engines explored in chapter 6.

* 1. Payload Check

The previously asserted weight of the batteries is 8.00137 kg and the engine weight lies at \_\_\_\_\_\_. After calculating our required materials weight for wings and fuselage of \_\_\_\_\_\_\_ kg and \_\_\_\_\_\_ kg we find that with a maximum takeoff weight of 31.6 kg our potential payload capacity is 6.6 kg. As previously stated in chapter 2 Wto was adjusted down from its maximized value of 64.3086 kg as it did not need to simultaneously operate with payload, endurance, and range requirements maximized. Thus, a healthier and more realistic Wto of 31.6 kg was selected for the aircraft. After having calculated the engine, battery, and body/wing weight we found that an empty weight of 21.6 kg could be achieved. This empty-weight value would leave our aircraft with a 6.6 kg payload weight, 1.6014121 kg greater than previously calculated. This value is exceptional as not only is our previous payload requirement met, but the design has also allowed us to exceed it opening our aircraft to a greater range of operational capabilities.

* 1. Improvements to Specific Power Consumption

The electric nature of the aircraft requires that any improvements to the Specific Power Consumption come from either the batteries or the electric engine. Since neither of these use a thermodynamic cycle to derive aircraft power improvements must instead be made to the material aspects of these systems themselves.

8.4.1 Proposed Improvements