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1. Introduction

In the following report the wing sizing for the aircraft corresponding with the requirements listed in Table 1.1 will be completed. For this task to be completed several design tasks must be completed, each of which will be detailed in the following chapters. For the first chapter (Chapter 2) the calculations needed for wing sizing and other wing design properties. In Chapter 3 the take-off and landing conditions as well as the selections and sizing of the required flaps will be completed for the designed wing in Chapter 2. Lastly Chapter 4 includes a consideration for wing geometry that could support additional battery weight loading.

1. Clean Wing Design

Before progress can be made on the design of the wing a clean wing design must first be created. This clean wing is designed without any concern for landing gear, fuel weight, or control surfaces. The wing will initially be considered to be infinite so that we can properly define the airfoil, once defined this airfoil will be integrated into the greater wing design.

* 1. The infinite Wing

The first step is calculating the total required lift by multiplying the aircraft weight by 1.1 which is detailed below.

Thus, the wing must generate 340.996 N at nominal cruise conditions. Previously wing loading at takeoff was calculated to be 150 N/m^2. This value can be used as our start cruise wing loading due to its proximity in the fight profile. The end cruise wing loading can also find to be 150 N/m^2 N as well since the lack of fuel weight dynamism results in no relative change in wing loading during different stages of the flight. Using the previously calculated values and the definition of lift equation we can calculate the coefficient of lift using the following equation.

A close-up of words

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The cruise Velocity and cruise Density are found from the atmospheric conditions around the aircraft at cruise. The W/S is the average of the weight at cruise (which does not change for our aircraft due to its electric powered design), and the total design CL can be multiplied by 1.1 in the same manner as the aircraft weight. These changes provide the adjusted equation below.

A close-up of a math problem

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Now using the values, we know the coefficient of lift for the wing can be calculated to be 1.010625. Since our aircraft is flying at such a low velocity the aircraft sweep will remain at 0 degrees, similar to our reference aircraft which fly at similar velocities. Exchanging starting conditions also allows us to calculate the CL start =1.010625 and CL end 1.010625 which once again sees minimal change as the lack of weight fluctuation in the aircraft prevents the start and end Cl from varying significantly. The table contains all the variables critical to the calculation of CL.

|  |  |  |
| --- | --- | --- |
| Parameters | Values | Units |
| W | 31.6 | kg |
| Pressure at sealevel | 1.225 | Kg/m^3 |
| Pressure at ceiling | 1.204 | Kg/m^3 |
| S | 3.5 | m^2 |
| Vcruise | 15 | m/s |

We must also calculate our thickness to chord ratio, which using our reference aircraft and historical considerations can be found to be 0.2.

Since the aircraft wing is not swept, our airfoil coefficient of lift remains the same as the coefficient of lift for the whole of the wing.

With this consideration in mind, the design of the airfoil should attempt to create a coefficient of lift around the value of 1.010625.

|  |  |
| --- | --- |
| Cl | 1.010625 |
| Sweep Angle | 0 |
| T/C | 0.2 |

* 1. Airfoil Selection

We found our airfoil design through the XFoil software. Since our aircraft is flying at a very low speed and is gliding for large portions of its flight profile, an airfoil that maximizes coefficient of lift was critical to the design of our aircraft. With this in mind using the calculated Cl of 1.010625 and the T/C ratio of 0.2 we found that the Selig 1223 airfoil best suited the needs and specifications of our aircraft.

The acting speed on the airfoil is then calculated to be the same as the cruise speed due to the low cruise altitude of the aircraft. As a supplement to the airfoil selection process a Reynolds number of 0.4568370165745857 x 10^6 can be calculated with all conditions at cruise. The mean aerodynamic chord is used as our chord value for all calculations and will be calculated in the following section.

A mathematical equation with black letters

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For Cruise conditions the Reynolds number can be calculated to be 0.4568370165745857 x 10^6.

Keeping the calculated Reynolds number in mind we chose to continue with our previously selected airfoil the Selig 1223 for our design.

|  |  |
| --- | --- |
| Chord Location of Minimum Pressure |  |
| Design lift coefficient | 1.010625 |
| Maximum Thickness |  |
| Mean line parameter |  |

The airfoil flow field at approach speed is shown in figure \_\_\_\_. Drag Polar and Lift Curve of the airfoil are shown in figure \_\_\_\_. And in figure \_\_\_\_ the lift curve is observed at a smaller level to find typical values that the aircraft will operate at. These values are taken from the graphs and compiled in the table below.

|  |  |  |  |
| --- | --- | --- | --- |
| Symbol | Info | Value | Unit |
| Alpha0L | Zero lift angle of attack |  | Deg  rad |
| dCl/dAlpha | Lift curve slope |  | 1/deg  1/rad |
| cd0 | minimum drag coefficient |  | - |
| AlphaStall | Stall angle of attack | -2.5 or 13 | Deg |
| Clmax | Maximum lift coefficient |  | - |
| alhpa0 | zero angle of attack lift coefficient |  | - |
| T/c | Thickness to chord ratio |  | - |
| Mcr | Critical Mach number at 0 degree Angle of attack |  | - |

* 1. The finite wing and mean aerodynamic chord

With the requirements for the 3D wing completed we can continue further calculations. From the previous assignment our wing surface area was found to be 3.5 m^2. With an aspect ratio of \_\_\_\_\_. Using the formula below a wingspan of \_\_\_\_\_\_ m is found.

A mathematical equation with letters and numbers

Description automatically generated

Our single wing length can be found to be \_\_\_\_\_\_ m. In the original aircraft drawings \_\_\_m of space was left for the installation of the wing. The root chord will thus be selected to be \_\_\_\_\_. Using the equation below the wing taper ration can be found. From this, the chord tip cT can also be found.

The wing tip chord can then be found to be \_\_\_\_\_ m. The mean span of a single wing and mean aerodynamic chord can be calculated using the equations listed below.

This means that the quarter means aerodynamic chord, where the location of the resultant lift force acts on the wing, can be calculated to be \_\_\_\_\_. Since this is where bending loads and wing bending relief are at optimal conditions the support spars for the aircraft butterfly tail will be located here. In the adjusted new drawings in the assignment appendix this change in spar location can be seen \_\_\_\_\_\_ m from the fuselage center.

* 1. Validity and Sanity Checks

The following sections will detail in depth comparisons with other aircraft to ensure that our calculated values remain within realistic bounds.

* + 1. Pitch up Tendency

The pitch up tendency can be found using data from figure \_\_\_\_. The quarter chord sweep angle can be found to be zero since there is once again no sweep of the wing. With x/c = \_\_\_\_, the root chord = \_\_\_\_m, the taper ratio = \_\_\_\_, and the span = \_\_\_\_ m. Using these values the pitch up tendency can be found to be \_\_\_\_\_\_.

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This pitch up tendency is not wholly indicative of the pitch up tendency of the entire aircraft as there is no contribution by the aircraft tail. With this additional consideration it is likely that the pitch up tendency would be significantly lower.

* + 1. Sweep Angle

By analyzing the wing design of similar reference aircraft we can confirm that having no sweep will not negatively affect our aircraft due to the low cruise speed.

* + 1. Aspect Ratio

Aspect ratio must be checked in order to ensure the structural viability of the wing design, the aspect ratio is found using the wind cantilever ratio, which is calculated as follows.

A mathematical equation with black text

Description automatically generated

Our span is already known and our thickness at the root can be calculated using the thickness to chord ratio and the root chord. This value is found to be \_\_\_\_ m. With sweep angle at half chord length being 0.

With these values we can calculate a cantilever ratio of \_\_\_\_\_. To remain within the sanity check we must have a ration between \_\_\_ and \_\_\_\_. The value remains within the sanity check bounds and thus we can proceed.

* + 1. Taper Ratio

Taper Ratio can be used as another sanity check. The taper ratio we previously calculated is \_\_\_\_ and we have a quarter chord sweep angle of 0 degrees, the above graphic shows that our value ought to be above \_\_\_\_\_.

With this in mind, we can confirm that this design passes the sanity check.

* + 1. Drag divergence

Due to the low cruise speed of the aircraft drag divergence is negligible at all times for the aircraft and thus has minimal use as a sanity check for the aircraft.

* + 1. Other wing parameters

Other parameters of the wing are the dihedral angle, the twist angle, and the incidence angle.

Dihedral angle is set at \_\_\_\_ taking into consideration the needed clearance for the landing gear as referenced in the aircraft drawings.

The twist angle will be set to 0 as the additional complexity of the wing geometry is unnecessary for the low velocity operation of the aircraft and the need for the aircraft aoa to remain close to zero for most of its operation.

Incidence angle can be calculated using the ideal aoa in combination with the parameters of the airfoil. The selected airfoil has a design lift coefficient of \_\_\_\_ and the value of the lift coefficient at zero angle of attack is \_\_\_\_. This value is within the acceptable range of our previously calculated value of \_\_\_\_, thus no changed need to be made and the incidence angle will be set at \_\_\_\_.

* 1. Clean Wing Lift Curve

With the majority of the necessary calculations completed the determination of the lift curve can be started. However, a few more angles are needed to complete all of the calculations. These angles include the trim angle, stall angle, and the slope angle which will all be calculated below.

* + 1. Curve slope angle

Curve slope angle will only be calculated at the cruise condition as its overall range of speed is already so low. Thus, the differences in the curve slope angle are negligible. Before any other calculations the Prandtl-Glauert compressibility factor is found using the first equation below. The Meff used is the normal mach number at cruise since there is no sweep of the wings the airfoil doesn’t impact the effective Mach number. The cruise speed is 0.04357 or 15 m/s.

A mathematical equations with numbers and lines

Description automatically generated with medium confidence

The derivative of the CL with respect to aoa is found using the equation below, where the aspect ratio is \_\_\_\_\_, the η or wing efficiency factor is \_\_\_\_\_\_\_\_\_\_\_ and the half chord sweep angle is assumed to be zero.

Now, the dCL/dα can be found using equation 2.16, where A is the aspect ratio (A=9), η is the

wing efficiency factor (set at 0.95) and λ0.5c is the half chord sweep angle, calculated with equation

A math problem with numbers and symbols

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* + 1. Trim angle

In order to determine the trim angle, we use the calculation below.

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Alpha0L is the angle determined previously in section 2.2, Cldes is the designed lift coefficient of \_\_\_\_ similarly determined earlier in this section, and CL alpha is the curve slope which is equal to \_\_\_\_. The trim angle is calculated to be \_\_\_\_\_ deg. This angle was not critical for Mach divergence of critical Mach number when analyzed in XFoil. This makes sense as the aircraft will not see any sonic conditions whatsoever during its operation.

* + 1. Stall angle

For the determination of the stall angle a check must be completed on the taper ratio. From reference data, the aspect ratio should be larger than the value from the right hand side of the equation below.

A black and white math equation

Description automatically generated with medium confidence

C1 can be derived from figure \_\_\_\_ to be \_\_\_\_, using a taper ration of \_\_\_\_\_.

This value provides us with a minimum aspect ratio of \_\_\_\_, since this is smaller than the aspect ratio previously set for the aircraft at \_\_\_\_ then this value can be accepted for design use and the high aspect ratio method then can be implemented for further calculation. First the leading edge of the wing’s sharpness must be calculated. This is done by identifying the difference between the vertical location of the \_\_\_\_\_\_ point and the \_\_\_\_\_ point at the top surface of the wing. From XFoil, this value is found to be \_\_\_\_. No further calculations are required for low-speed flights due low cruise velocity and velocity range of the aircraft. As calculated in the previous assignment the cruise velocity is 15m/s or 0.04357 Mach the relationship between these variables is further explored in the graphs below.

From the above graphs the CL/Clmax can be found to be = \_\_\_\_ and the change in aoa with respect to CL max is \_\_\_\_\_\_. From the exquation below Clmax can now be determined using a change in aoa with respect to CL max of 0.

With these value in hand the below equation can then be used to calculate the stall angle of attack.

2.5.4 Graphical Representation

Alpha0L is the same for the airfoil and wing and the CL alpha was already previously calculated.

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1. High Lift Devices

For several reasons we decided that our aircraft design would not require high lift devices for its operation. This decision was made based on a variety of factors further explored below that justify the lack of their addition.

* 1. Wing Loading

One of the primary purposes for high lift devices is to adjust for variable wing loading and lift coefficients on aircraft. This is typically due to the burning of inboard and outboard fuel reducing overall aircraft weight. This causes a resulting variable wing loading throughout the aircraft flight profile necessitating high lift devices so that the lift produced by the wings can be adjusted for the changing wing loading as the craft proceeds through its flight profile.

In the aircraft design which we have chosen the electrical method of propulsion makes it so that wing loading is static. With a static wing loading no variable adjustments for increases or decreases in lift output are required due to the unchanging nature of the aircraft weight throughout the profile. Thus, with the absence of this typical justification for high lift devices, a decision was made to not include them in the design.

* 1. Structural Complexity

This aircraft is designed with a very high aspect ratio, as much of its flight profile will be spent in low aoa descents or ascents and periods of low powered gliding. Thus our selection of Selig 1223 airfoil was primarily driven by its high aspect ratio and lift output. This airfoil however has a very complex geometry making it difficult to manufacture in its own right. The addition of high lift devices would cause two main issues for such a complex airfoil. For one such added complexity would only further complicate the internal structure and geometry of the airfoil and would likely take away from its overall aerodynamic efficiency and negatively impact its defining high aspect ratio. Secondly the addition of further structural complexity would serve to increase the aircraft wing loading by adding additional outboard weight in the form of servos and actuators. Since the benefits these devices would provide are already being accounted for using the highly efficient Selig 1223 airfoil this added complexity would seriously complicate our designs and require extensive recalculation over the overall wing structure while providing little to not benefit to the design.

1. Outboard Energy Storage

Additional outboard fuel storage within the aircraft wings will not be needed in our design for several reasons. First and foremost the electrical nature of our aircraft dictates that no liquid fuel storage is necessary as all energy used by the aircraft is stored in its batteries. Additionally, in our initial fuselage design in Assignment 1 all necessary battery storage was identified and placed within the fuselage. Thus, additional outboard battery storage is not necessary.

On top of this additional outboard battery storage would only serve to further complicate and undermine the design of the aircraft. The batteries we are using are premanufactured thus the density and volume of our battery packs are fixed. This would mean that any additional battery load in the wings would not be evenly distributed and would be concentrated in a single point. Addition of so much weight in the outboard would undermine the aircraft cg, place large moment forces on the aircraft wings undermining their structural viability, and would require changes to the dimensioning of the airfoil. This would only serve to needlessly complicate the aircraft design by adding additional wing loading and lowering the Clmax for no practical gain.

All these factors when accounted show that this aircraft does not need additional outboard energy storage and would only suffer from its addition.