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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 \_\_\_\_\_ N at nominal cruise conditions. Previously wing loading at takeoff was calculated to be \_\_\_\_\_\_ N. 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 found to be \_\_\_\_\_\_\_ N as calculated in the previous Assignment. Using the previously calculated values an the definition of lift equation we can calculate the coefficient of life 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 \_\_\_\_\_. 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 = \_\_\_\_\_\_\_ and CL end \_\_\_\_\_\_\_. The table contains all the variables critical to the calculation of CL.

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

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

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 \_\_\_\_\_\_.

|  |  |
| --- | --- |
| Cl |  |
| Sweep Angle |  |
| T/S |  |

* 1. Airfoil Selection

We found our airfoil design through the DesignFOIL and JAVAFOIL software. Since our aircraft is to fly at a very low speed and glide for large portions of its flight profile a airfoil that maximizes aspect ratio was critical to the design of our aircraft. With this in mind using the calculated of Cl of \_\_\_\_\_\_ and the T/C ratio of \_\_\_\_\_\_\_ we found that the \_\_\_\_\_\_\_\_\_\_ airfoil best suited the needs and specifications of our aircraft.

The acting speed on the airfoil is then calculated to be \_\_\_\_\_. As a supplement to the aifoil selection process a Reynolds number of \_\_\_\_ 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 \_\_\_\_\_\_\_\_.

Keeping the calculated Reynolds number in mind we selected the \_\_\_\_\_\_\_\_\_ airfoil for our design.

|  |  |
| --- | --- |
| Chord Location of Minimum Pressure |  |
| Design lift coefficient |  |
| 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 |  | Deg  rad |
| Clmax | Maximum lift coefficient |  | - |
| alhpa0 | zero angle of attack lift coefficient |  | - |
| T/c | Thickness to chord ratio |  | - |
| Mcr | Critical Mach number at 0o 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 \_\_\_\_\_. 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

Description automatically generated

* + 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

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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 CLmax

From these graphs, the CLmax/Clmax = 0.78 and the δαCLmax= 3 0 = 0.052 rad. From equation 2.19,

the CLmax can now be determined, using a δCLmax equal to 0, since this is at low speed.