Design Report 3 - Detailed Design

Engi 7926 - Mechanical Design Project I

Group M10 - FICE
"*Flying In Control Engineering*"

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Abstract

As part of the design process to determine the optimal aircraft design, detailed design must be performed on the aircraft. The third design report created by Flying in Control Engineering (FICE) will cover the basic sizing and performance specifications, XFLR5 design analysis, refined sizing using analysis results, and a preliminary computed aided design (CAD) model of the remote control (RC) airplane. This report is the third in a series of four reports which will be completed throughout the aircraft design, building and testing process.

Before system analysis and a CAD model can be created, basic sizing and performance specifications must be performed. By using the design specifications found in Report 1, the critical performance specifications of the aircraft were obtained such as wing loading, power loading, aircraft lift coefficient, and lift to drag ratio and stored in a spreadsheet using Excel. These values were then compared with values of the FT Explorer to check their viability and the compared values fell between an accepted range. Next, a basic sizing sheet was created of the aircraft which included values such as Oswald efficiency, zero lift drag coefficient, and propulsion efficiency.

After basic sizing was completed, dynamic analysis was performed using XFLR5 software using the airfoil and parameters chosen by FICE. Once XFLR5 analysis was completed, the values of max coefficient of lift, Oswald efficiency, and zero drag lift were obtained using the drag polar graph obtained.

Once XFLR analysis was finished, refined sizing analysis was completed on the aircraft. Values for the updated max lift coefficient, Oswald efficiency, and zero lift drag coefficient were updated on the sizing sheet and updated thrust and power values were obtained. Using the updated values, plots for required and max thrust/power versus airspeed were generated to obtain system thrust and power efficiencies. The minimum aircraft turn radius was then determined by analyzing the aircraft load factor plots.

Finally, this report includes a CAD model of the aircraft initial prototype and an assembly drawing. The CAD model was used to obtain the prototype’s airframe weight which were then updated on the Excel sizing sheet.

As a result of the analysis and design work prepared in this report, the manufacturing of FICE’s rc airplane is ready to commence.

Contents

Figures

Tables

1.0 Introduction

The Mechanical Engineering Class of 2019 has been challenged to design, build, and test an RC aircraft. Each team will enter their RC aircraft into a competition where the model will be judged based on its ability to carry a payload. The team that achieves the highest payload fraction, while successfully navigating a full loop of the Techniplex, will be deemed the winners.

The following specifications fit FICE's desired aircraft build:

Table I: Desired Aircraft Specifications

Stall Speed	< 8 m/s
Cruise Speed	10 - 20 m/s
Wing Loading	0.3-2.5 lb/ft^2
Power Loading	2 - 18 lb/hp
Payload Percentage	35 - 55%
Wing Span Length	30 - 36"
Take-off Velocity	8-10 m/s
Minimum Turn Radius	10.95m (@12m/s)
Total Aircraft Weight (w/ Payload)	1100 g

Figure 1 shows a sketch of the conceptual design generated in Report 2.

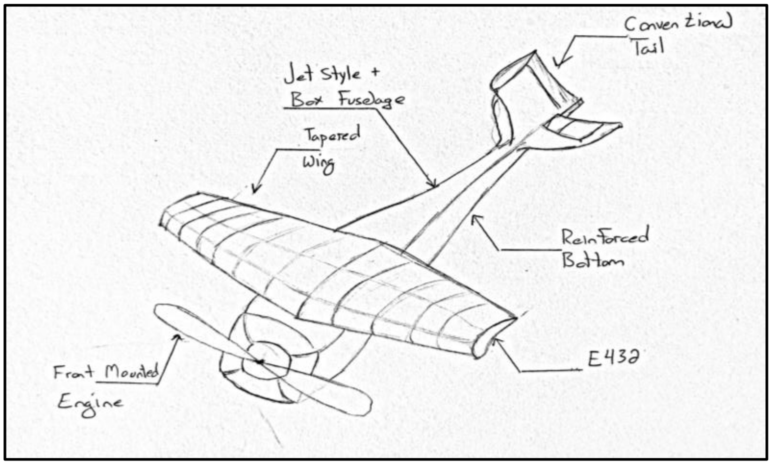

Figure 1: Conceptual Design Sketch

This report serves to document the detailed design phase of FICE's RC plane for the Memorial Cup competition. This phase commenced with estimating the initial aircraft sizing parameters. This was then followed up by detailed engineering analysis using XFLR5 simulation tools. Simulation results were then used to refine the system size parameters to obtain the final set of system size values. Following this process, FICE has developed an RC aircraft design which can carry a high payload fraction while staying under budget.

2.0 Basic Sizing and Critical Performance Specifications

By using the conceptual design and the design specifications shown in **Table I**, the basic sizing of the plane was obtained. Additionally, the critical performance sizing parameters of the aircraft were found as shown in **Table II**.

Table II: Basic Sizing Parameters

Lift Coefficient	1.6
Lift to Drag Ratio	52
Wing Loading	2.27 lb/ft^2
Power Loading	5.38 lb/hp
Oswald Efficiency	92%
Zero Lift Drag	0.142
Propulsion Efficiency	40%

The aircraft lift coefficient and lift to drag ratio were taken from a basic polar of the E423 airfoil as the starting points to use before performing XFLR5 analysis. The wing loading was found as a function of total weight and the wing area. The team chose to use the longest span possible for a single piece wing, thus making the span 36 inches. A high aspect ratio was also very important, for that reason the team chose a moderately small chord of 11 to make the aspect ratio 8.2. With this span and chord, the wing area was found to be 990cm^2. The total weight was found by adding the weight of the airframe, payload, and electronics, giving a value of 1.1kg. Having both total weight and wing area, the wing loading was calculated to be 2.27lb/ft^2. The power loading is a function of the total weight and the motor power, the total weight was the same as previously mentioned and the motor power was a set value of 336W. This allowed the team to calculate the power loading as 5.38lb/hp. Note, both the wing loading and power loading are in the range of the recommended values and correspond well with the values of the FT Explorer.

The Oswald efficiency was estimated using the following formula[1]:

$$e = \frac{Cl * 2}{AR * \pi * \mathrm{a}}$$

When alpha is 5 degrees, this equation yielded the team with a value of 92%. The zero lift drag was estimated by finding the point of zero lift on an E423 CL vs CD polar and then taking the corresponding drag value. This gave the team a zero lift drag of 0.142. The propulsion efficiency was found using the following equation:

$$\eta = \frac{T * Vcruise}{V * I} = \frac{1.13 * 9.81 * 12}{336} * 100 = 40\%$$

This propulsion efficiency was found to be 40% at the desired cruise speed of 12m/s.

3.0 Initial XFLR5 Analysis

The procedure used in the XFLR5 analysis is as follows:

1. Import the E423 airfoil for the main wing.
2. Select the NACA 0009 airfoil for the tail.
3. Increase airfoil panels for added accuracy.
4. Go to the direct analysis tab and create a new multi batch analysis.
5. Select both airfoils for the analysis and an appropriate Reynolds number range.
6. Run the analysis and then go to the wing and plane design tab.
7. Create a new plane and define all parameters.
8. Define an analysis, one for fixed speed and one for fixed lift.
9. Run the analysis and then use the generated graphs to obtain parameters needed for sizing.

Following this procedure, the following graphs were generated.

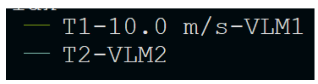

Figure 2: Legend

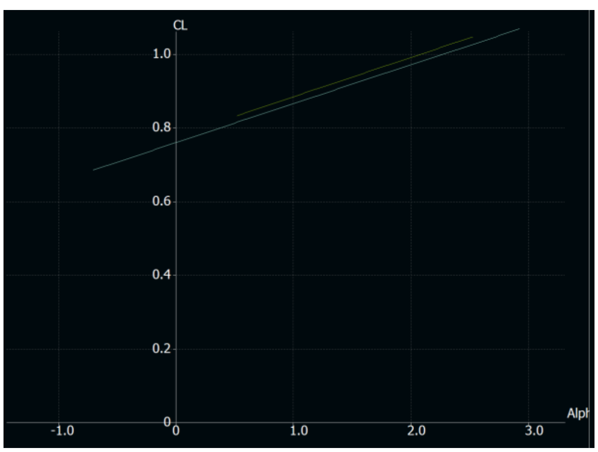

Figure 3: CL vs Alpha

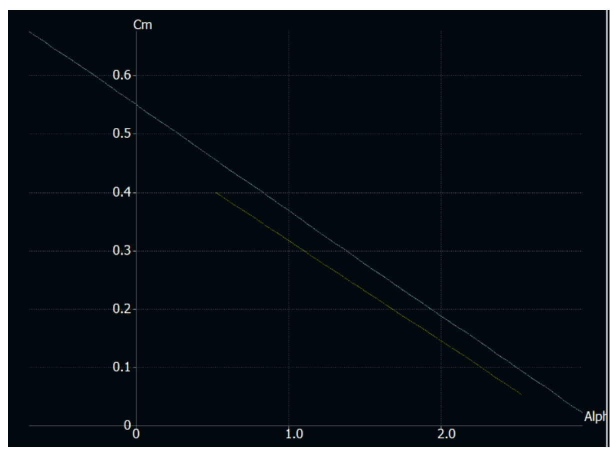

Figure 4: Cm vs Alpha

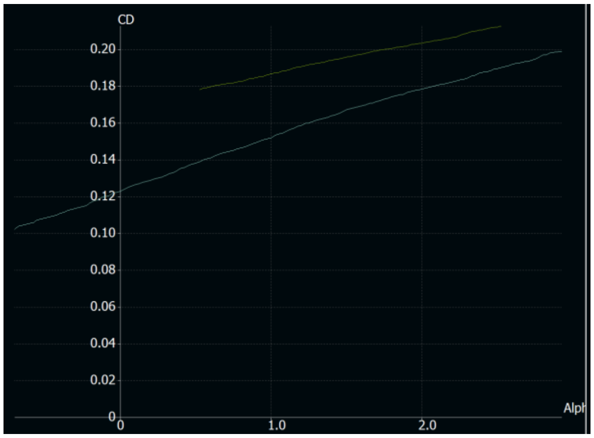

Figure 5: Cd vs Alpha

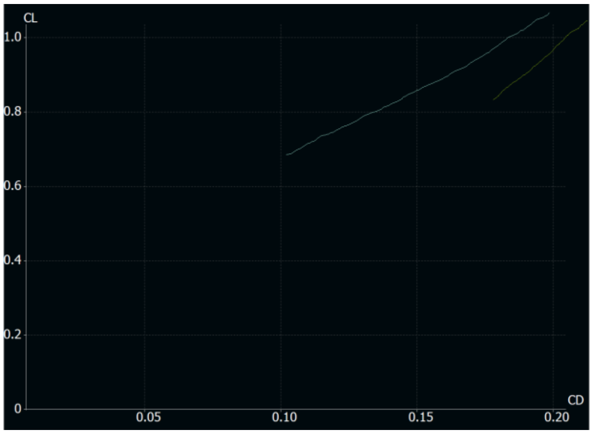

Figure 6: CL vs Cd

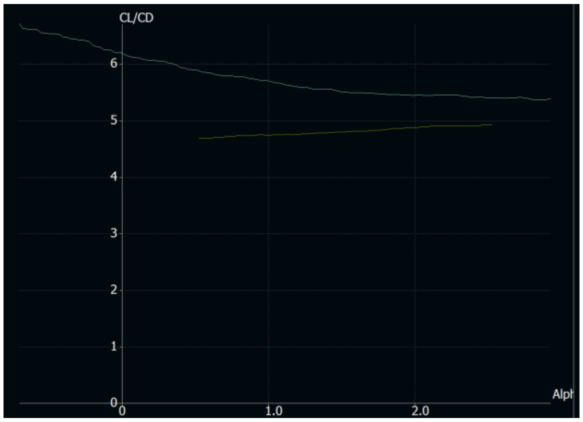

Figure 7: CL/CD vs Alpha

Using these graphs, the max coefficient of lift, zero lift drag, and Oswald efficiency can be calculated more accurately. The max coefficient of lift can be read directly off the graph as 1.1, the team believes the lift can actually go higher at a high angle of attack, but the XFLR5 analysis produced inaccurate results at higher alphas. To find the zero lift drag the team took the drag when CL equaled zero. The zero lift drag value was found as 0.07. To find the Oswald efficiency, first k must be found using the following equation where Cl is calculated from the cruise speed and CD is the corresponding drag from the CL vs CD graph:

$$k = \frac{CD - CDo}{Cl^2} = \frac{0.14 - 0.07}{0.88^2} = 0.09$$

With k known, the Oswald efficiency can be calculated:

$$e = \frac{1}{AR * \pi * k} = \frac{1}{8.18 * \pi * 0.09} * 100 = 43\%$$

Note: Cm has a negative slope indicating stability. At approximately 5 degrees angle of attack the team believes parameters would be optimized based off online research. Unfortunately, XFLR5 was unable to do analyze this value.

4.0 Refined Sizing Using XFLR5

The turn radius was calculated using the following equations and plotted in **Figure 9** as a Turn Radius vs Velocity graph. The resultant equation was used to obtain the plot:

$$R = \frac{V^2}{g\sqrt{n^2-1}} \text{ where } n = \frac{\rho V^2 SCl}{2W}$$

To obtain the minimum turn radius (turn radius at minimum flight speed), a load factor vs. velocity plot was created as is shown in **Figure 8** below. The intersection point is the point at which turn radius is at the stall speed. The corresponding turn radius value is 10.95m. The equation used to obtain this plot is:

$$n = \sqrt{\frac{v^4}{r^2 * g^2} + 1}$$

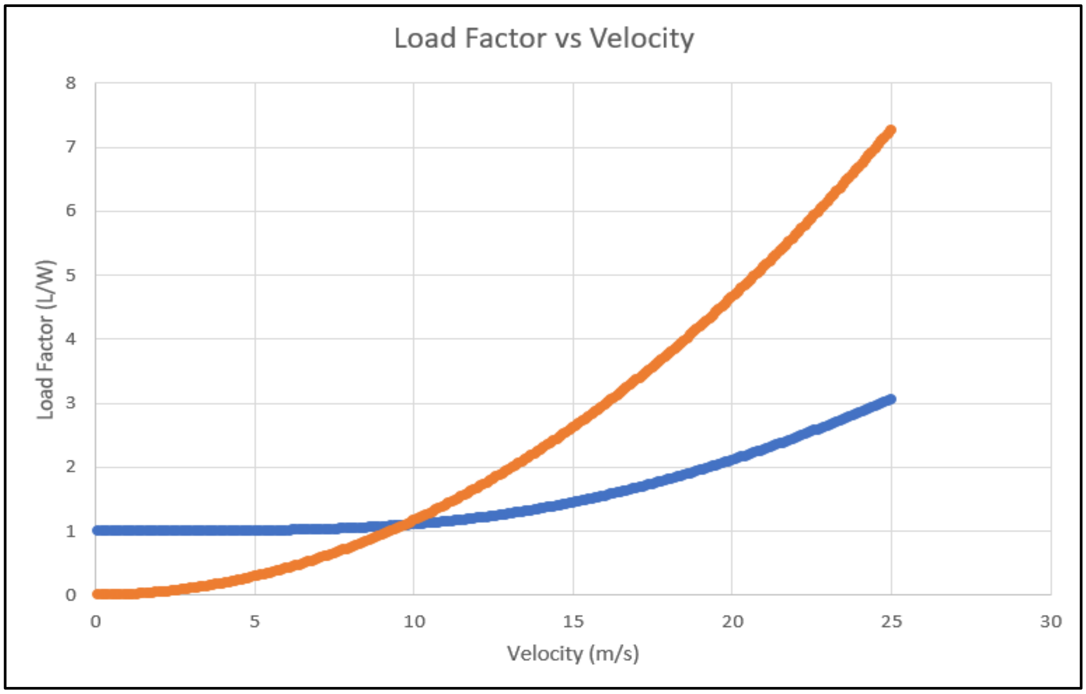

Figure 8: Load Factor Vs. Velocity Graph

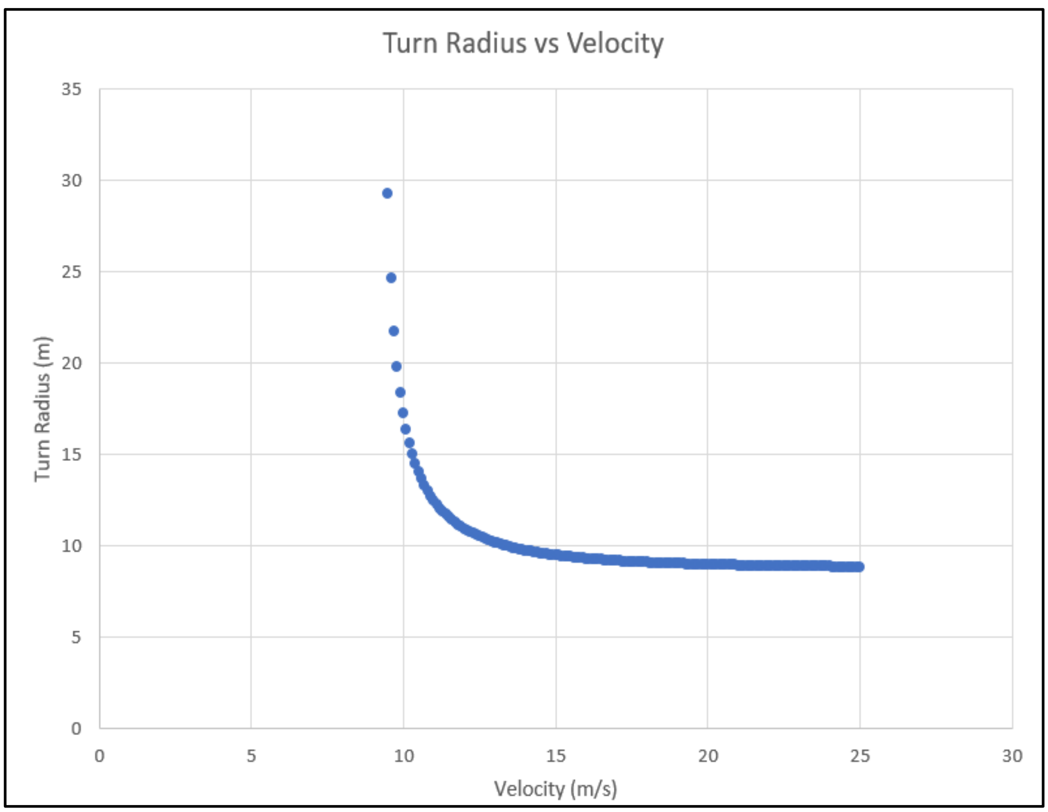

Figure 9: Turn Radius Vs. Velocity Plot

A Thrust required/Thrust available vs. Airspeed graph was then generated as shown in **Figure 10**. On this graph, the stall speed, max speed, max endurance, and max range speeds can all be found. Additionally, this plot visualizes the stability of the aircraft, where any velocity in between the stall speed and the maximum speed will result in stable flight in terms of required thrust versus available thrust. The vertical separation between the available thrust and required thrust lines (vertical separation at a specific velocity) on the graph represents excess thrust at a specific velocity.

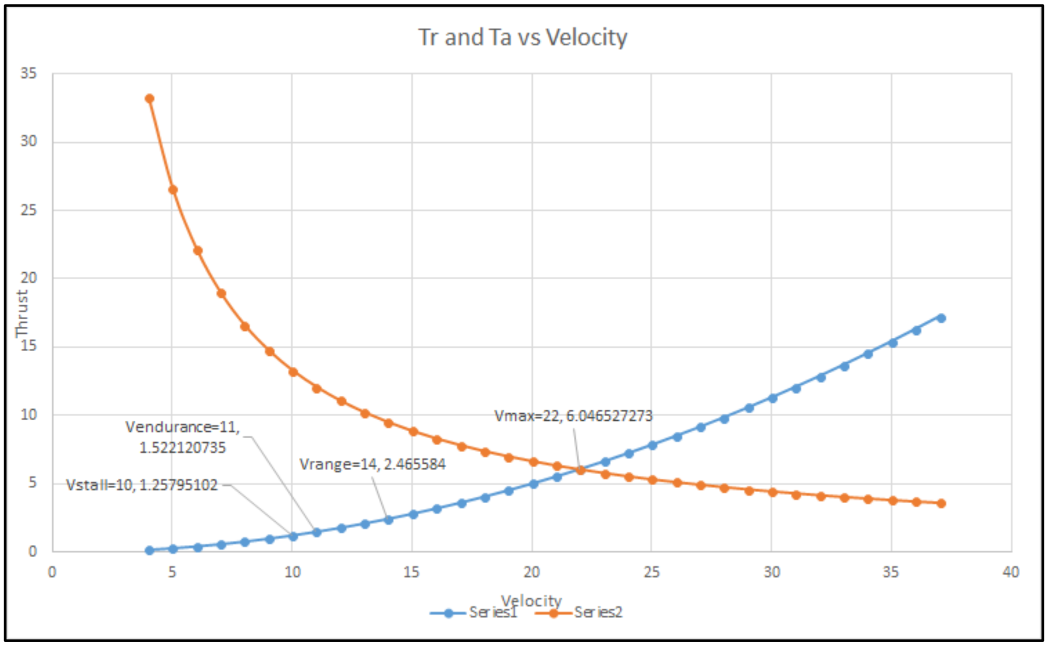

Figure 10: Thrust Required and Thrust Available Vs. Velocity Plot

Thrust available and thrust required were calculated using the following equations:

$$T_a = \frac{\eta * P_a}{V}$$

$$T_r = 0.5 * C_D * S * \rho * V^2$$

The velocity values were found using the following equations:

$$V_{range} = \left(\frac{2}{\rho} \sqrt{\frac{K}{C_{D0}}} \frac{W}{S} \right)^{1/2}$$

$$V_{endurance} = \left(\frac{2}{\rho} \sqrt{\frac{K}{3C_{D0}}} \frac{W}{S} \right)^{1/2}$$

$$V_{stall} = \sqrt{\frac{2W}{\rho S \, C_{Lmax}}}$$

A Power required/Power available vs. Airspeed graph was generated as shown in **Figure 11**. This plot visualizes stability of the aircraft, where any velocity in between the stall speed and the maximum speed will result in stable flight in terms of required power versus available power. The vertical separation between the available power and required power lines (vertical separation at a specific velocity) on the graph represents excess power at a specific velocity.

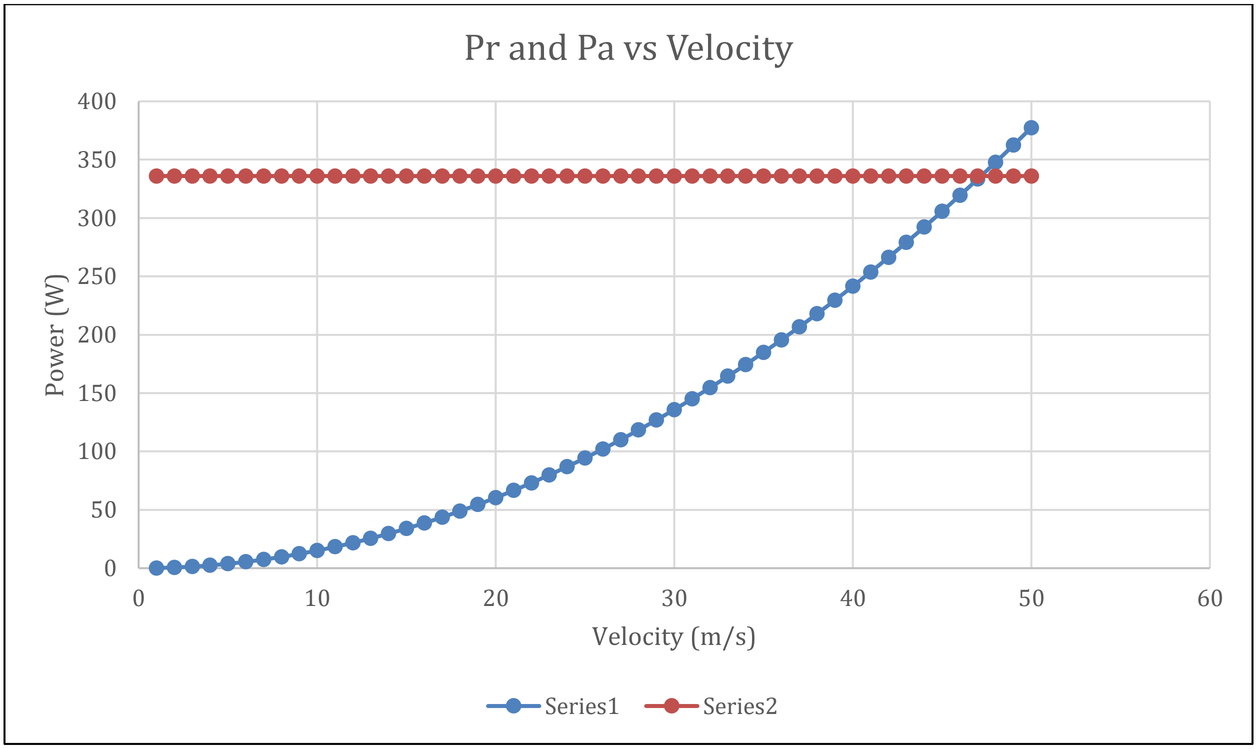

Figure 11: Power Required and Power Available Vs. Velocity Plot

The power available was simple the maximum battery power which was 336W, the power required was calculated with the following equation:

$$P_r = T_r V_{cruise}$$

5.0 CAD Design

Using SolidWorks, FICE has designed a current computer aided design (CAD) prototype of the aircraft. This can be seen below in **Figure XX**. The current model was made by designing each individual component separately, and then mating them together in an assembly. This allowed for a modular model which could be easily reconfigurable to make design modifications.

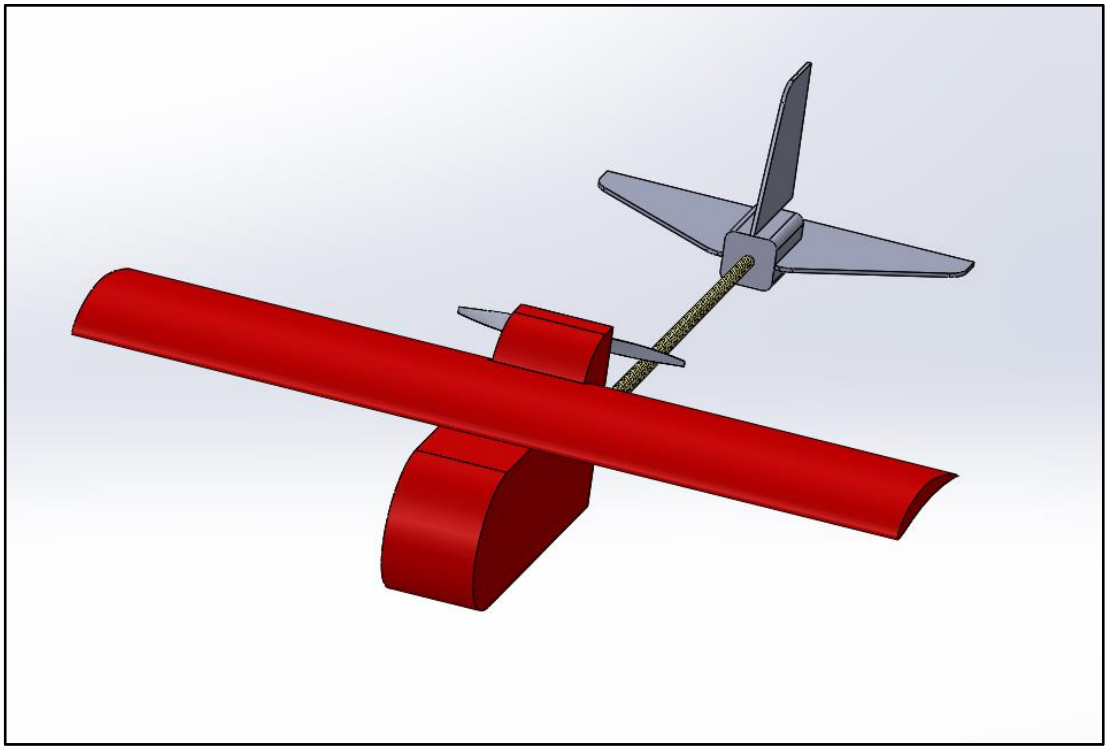

Figure 12: Current Prototype CAD Model

The SolidWorks model proved useful in performing appropriate sizing and positioning as well as weight analysis of individual components and overall design. There were several major system modifications from the conceptual design seen in the previous report as a result of analyzing the CAD model. Changes included the tail boom being modified from a boxed foam configuration to a tapered carbon fiber rod configuration which reduced system weight, manufacturing time, and improving structural integrity. As well, the engine mounting was changed from front to back in order to protect the propeller and electrical components during landing and crashing scenarios. Additional, benefits of switching from a front mounted propeller to a rear mounted propeller was that it brought the weight of the motor back which will help keep the plane from nose diving during flight. It also allowed the front of the fuselage to take on a more aerodynamic shape. Although these changes are against the initial scoring matrices, the team found them necessary to create a successful aircraft.

As previously mentioned, the current CAD model is broken down into many individual components. The current breakdown of those components is as follows:

The fuselage is currently modelled with an aerodynamic nose with a top, rear mounted motor and propeller. The interior of the fuselage is hollow to allow for the necessary electronics and a place to store the payload. The rear mounted motor and propeller, in conjunction with the payload being loaded in the fuselage, allows for the two weights to offset about the tipping point of the fuselage, adding overall stability to aircraft. The fuselage is to be fabricated from buoyancy foam.

No changes were made to the airfoil during this design iteration. As in the preliminary model, it remains an E423 airfoil with a 36” span. Similar to the fuselage, the airfoil will be made from buoyancy foam.

The tail boom has been modified since the preliminary model. In this design iteration, the tail boom was modified to be a 36” tapered carbon fibre tube rather than using foam. There were several reasons as to why this modification was made. Firstly, for ease of manufacturing. The original design would have to be hand cut by FICE which may pose problems with getting it accurate to the design. Secondly, the carbon fibre tube adds a significant amount of strength and integrity to the aircraft, especially being tapered so that there is added strength in the fuselage and less in the tail, adding to weight saving over its constant diameter counterpart.

The tail of the aircraft is made up of a small foam block connected to the end of the tail boom to allow a place to mount the rudder and elevator fins. The two fins are being constructed of foam board as they were in the FT Explorer aircraft. This overall design should keep the tail of the aircraft relatively light weight.

By comparing this report to the previous, it is evident that there have been quite a few changes made by FICE. The team agreed these changes were necessary to ensure a successful flight.

Using the current CAD model, FICE was able to get a rough estimate of the weight of the airframe (without any electronics, motor, battery or payload) and was found to be approximately 350g. This is only a preliminary estimate of the weight as the exact density of some materials are only estimates and also depends on the manufacturing process of some of the parts as errors in this process could potentially add a lot of extra material to the final aircraft. In addition to neglecting the aforementioned parts in this weight estimation, there will also be added weight due to means of fastening on adhesion (such as hot glue). This will be accounted for during fabrication as well as weight to reinforce the bottom of the aircraft to help it survive multiple landings.

An assembly drawing of the initial prototype design which details the dimensions of the aircraft is shown in **Appendix B**.

6.0 Conclusion

This report discusses the detailed design process which FICE used to determine an optimal aircraft design following the conceptual design from Report 2. This process involved finding the basic sizing and performance specifications, performing XFLR5 design analysis and refining the aircraft size using these results and, thus, creating a prototype CAD model. Through this process, FICE has obtained a design that can reach the design specifications from Report 1. In the coming weeks, FICE will procure and manufacture the aircraft to perform initial testing and refine the detailed conceptual design.

7.0 References

[1] FSDeveloper (2013). “Finding Real World e and CD induced”. [Online], Available: https://www.fsdeveloper.com/forum/threads/finding-real-world-oswald-e-and-cd-induced.424644/

APPENDIX A - Complete Sizing Sheet

ENGI 7926 - Aero Design - General Sizing Sheet

					Notes/Equations Used
Weights			**lbm**	**N**	
Airframe (W_e)	0.458	kg	1.0097	4.493	Airframe Only
Batteries and Instruments (W_f)	0.327	kg	0.7209	3.2079	
Payload (W_pl)	0.315	kg	0.6945	3.0902	No Payload Assumed
Total (W_o)	1.1	kg	2.4251	10.791	
Aircraft Aerodynamics					Calculate using the provided drag polar
Max Lift Coefficient (C_lmax)	1.1	-			http://airfoiltools.com/polar/details?pola
Max Lift to Drag Ratio (L/D_max)	6	-			http://airfoiltools.com/polar/details?pola
Zero Lift Drag Coefficient (C_do)	0.07	-			
K	0.09	-			
Oswald Efficiency (e)	43.23%	-			
Environmental Conditions					
Air Density (ρ)	1.225	kg/m^3			Assume Sea Level
Gravity (g)	9.81	m/s^2			
Air Speed (V _cruise)	12	m/s			
Propulsion					
Motor Power (P_max)	336	W	0.4506	HP	
Battery Voltage (V_bat)	11.1	V			
Battery Capacity (E_bat)	2200	mAh			
Propeller Diameter (Φ_prop)	204	mm			
Propeller Pitch (p_prop)	101.6	mm			
Propulsion Efficiency (η_p)	0.3959	-			
Max Thrust (T_max)	11.0853	N			
					Coordinate System Origin: y wing tip x
Main Wing			**Ft**	**m**	pointing forward (nose) and z pointing
Airfoil	Flat Bottom				
Aspect Ratio (AR)	8.18182	-			
Taper Ratio	1	-			
Wing Area (S)	990	cm^2	1.0656	0.099	
Wingspan (b)	90	cm	2.9528	0.9	
Semispan (b/2)	45	cm	1.4764	0.45	
Root Chord (c_root)	11	cm	0.3609	0.11	
Tip Chord (c_tip)	11	cm	0.3609	0.11	
Average Chord (c)	11	cm	0.3609	0.11	
Sweep	0	deg			
Wing Rise	9.5	cm	0.3117	0.095	
Dihedral	8	deg			

Horizontal Stabilizer			Ft	m	
Airfoil	Over/Under Cambered				
Aspect Ratio (AR)	3.15433	-			
Taper Ratio	0.52941	-			
Wing Area (S)	520	cm^2	0.5597	0.052	
Wingspan (b)	40.5	cm	1.3287	0.405	
Semispan (b/2)	20.25	cm	0.6644	0.2025	
Root Chord (c_root)	17	cm	0.5577	0.17	
Tip Chord (c_tip)	9	cm	0.2953	0.09	
Average Chord (c)	12.9	cm	0.4232	0.129	
Sweep (this is an angle)	25.4	deg			
Volume Ratio	3.28999	-			
Lever Arm (x)	68.9	cm			
Vertical Offset (z)	11.5	cm			
Vertical Stabilizer			**Ft**	**m**	
Airfoil	Over/Under Cambered				
Aspect Ratio (AR)	4.25163	-			
Taper Ratio	0.70968	-			
Wing Area (S)	288.125	cm^2	0.3101	0.0288	Total Height of the Wing
Semispan (b/2)	17.5	cm	0.5741	0.175	
Root Chord (c_root)	15.5	cm	0.5085	0.155	
Tip Chord (c_tip)	11	cm	0.3609	0.11	
Average Chord (c)	13.25	cm	0.4347	0.1325	
Sweep	58	deg			
Volume Ratio	0.2228	-			(Surface area of the stabilizer *lever arm)/(main wing area *main wingspan)
Lever Arm (x)	68.9	cm			
Vertical Offset (z)	5	cm			

				Notes/Equations Used
Performance				
Wing Loading (W/S)	2.275734929	lb/ft^2		
Power Loading (max)	5.382099663	lb/hp		
Payload Fraction	0.286363636	-		
Lift Coefficient	1.235827664	-		
Drag Coefficient	0.207454301	-		
Lift to Drag Ratio	5.957107931	-		
Thrust Required	1.811449469	N		
Power Required	21.73739363	W		
Current Draw (Cruise)	1.958323751	A		
Stall Speed	10.54630219	m/s		
Max Range Air Speed	14.20516622	m/s		
Max Endurance Air Speed	10.79359221	m/s		
Rate of Climb	2.0144003	m/s		
Estimated Flight Time	3235.032199	s		
Estimated Range	38820.38638	m		

APPENDIX B - CAD Drawings

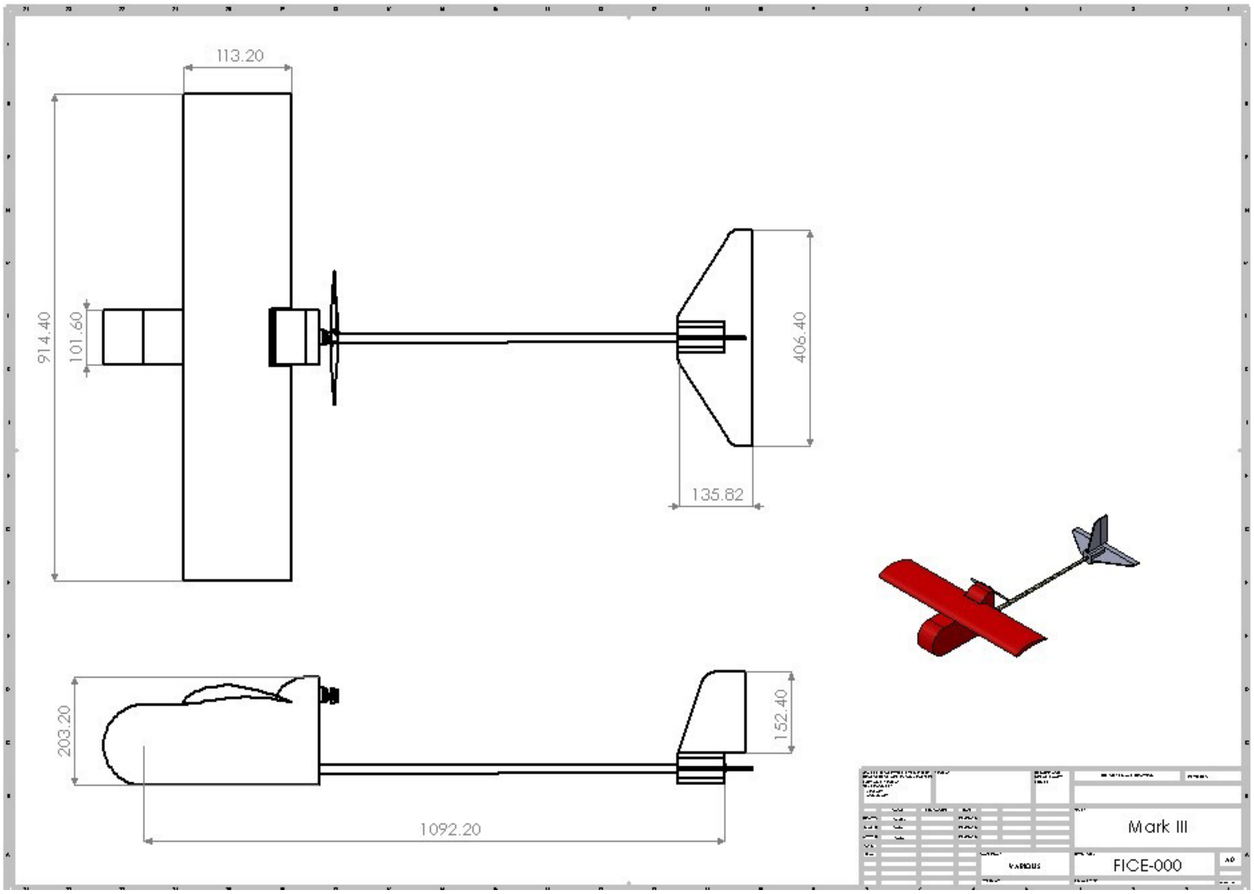

Figure 13: CAD Drawings