Autonomous Tricopter VTOL Vehicle

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***Abstract*—In this STAR Summer practice project, we designed, developed and manufactured a tricopter Vertical Takeoff and Landing(VTOL) vehicle. The work includes design from first aerodynamics principles to final manufactured vehicle. Apart from the design of the vehicle, we present control methods, simulations, modeling of the actuators, embedded control on the autopilot software and communication scheme between the autopilot and onboard computer running ROS.**

List of Symbols

The next list describes several symbols that will be later used within the body of the document

ρ	Air Density($\frac{kg}{m^3}$)
AR	Aspect Ratio equal to $\frac{b^2}{S}$
b	Wing Span
C_T	Thrust Coefficient
CD	Drag Coefficient
CL	Lift Coefficient
e	Oswald Efficiency Factor
n	Load Factor
q	Dynamic pressure. It is equal to $\frac{1}{2}\rho SV^2$
S	Wing Area
T/W	Total Thrust Available over Total Weight. Called Thrust to Weight Ratio
V	Airspeed
W/S	Wing Loading. Total Mass divided by Total Wing Area
Cruise Speed	The speed in which Total force on the aircraft is zero.

I. Introduction

With the help of Prof. Dr. Kemal Leblebicioğlu, we, METU RAS Team, started working on the VTOL project. Aim of this project is to design, manufacture a VTOL vehicle. Work also includes simulation, modeling and control of the system during all flight phases.

A. VTOL Vehicles

VTOL vehicles are kind of aerial vehicles which can takeoff and land vertically without the need of a runway. Most popular examples are multicopters which became very popular in short period of time. Another design philosophy is to have a fixed wing plane and modify it such that it can takeoff vertically.

This design method has significant advantage over the classic multicopter approach. The fact that without wings, quadcopters has to constantly apply force to keep itself afloat

(a) Commercial VTOL

(b) A Multicopter

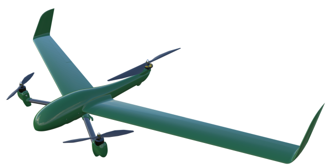
Fig. 1: Two Examples of VTOL vehicles

and to maneuver. A VTOL with wings can rely on the lift wings provide so it can fly with less force or thrust applied from the motors. This results in huge efficiency differences. This results in range and endurance differences. However, multicopter approach also has some advantages as well. These are:

- Multicopters are inherently VTOL vehicles. So after takeoff or for landing, there is no change in the control scheme on the autopilot. But in a fixed wing VTOL, a transition phase must occur so that the vehicle can accelerate to the minimum speed that wings can provide enough lift.
- Multicopters are much more maneuverable than fixed wing counterparts. Fixed wing vehicles needs much larger space to maneuver. This makes them unsuitable for indoor usage.

B. Tricopter VTOL

As mentioned earlier, the purpose of this project is building a tricopter VTOL. This configuration is based on three motors. There are two motors on the front of the centre of gravity and one motor behind. Vehicle takes off with all motors pointing upwards. Vehicle takes off just like a tricopter and then tilts the forward two motors with some angle. This accelerates the vehicle to minimum speed it can sustain flight which is called the stall velocity. After stall velocity is reached, front motors becomes parallel with the flight direction and third motor shuts down. So vehicle essentially transitions into fixed wing flight with two motors. Another very popular setup is five motor setup. In this setup, four fixed engines provide vertical thrust just like an multicopter and a fifth engine, either at front or at the back, provides thrust in the flight direction. Transition is simply done by activating the fifth engine and keep other four operating before reaching stall speed. After transition is done, vehicles are said to be in forward flight.



(a) A tricopter VTOL Example

(b) A Five Engine Setup Example

Fig. 2: Two Examples of VTOL vehicles

There are pros and cons of these different approaches which can be summarized as:

- In tricopter VTOL, only the third engine is not operational during forward flight whereas four engine in the five engine setup is not operational. This increases the dead weight in the vehicle. This decreases efficiency, hence range and endurance of the aircraft is reduced.
- A tricopter VTOL has tilt mechanism in the front engines. Additional mechanism increases complexity of the mechanical design and possibly create reliability issues. Also tilting requires more sophisticated control during transition than five engine setup.
- An important parameter in fixed wing design is Total Thrust Available over Total Mass. This is called Thrust to Weight Ratio or T/W. Since all three engines support the total weight in tricopter, after transition thrust to weight ratio could be close to one. This high ratio might not be necessary so there could be effects on efficiency.
- Fixed wing aircraft traditionally relies on a tail to provide stability. Real advantage in terms of weight and efficiency can be further increased if tricopter VTOL is designed without a tail like a delta wing design. Third engine can provide stability in pitch which further reduces dead weight and increases efficiency.

As the purpose of this project, we chose to implement a tricopter. This is due to its advantages in weight and efficiency. Our implementation only features two tilting motors in the front. This provides an advantage in weight since one more tilt mechanism is not needed. But with fixed third engine in the back, yaw is controlled by changing the speeds of two front motors. This also causes a rolling moment as well. So weight save results in more complicated control scheme. Also the controller tuning is more sensitive since control in one axis disturbs the other axis.

II. Conceptual Design and Aerodynamic Preliminaries

In this section, conceptual design philosophy is explained. Also aerodynamic preliminaries used in design process is also explained.

There are couple of approaches designing such vehicles. Aircraft design is a multiple constraint design problem with almost three digit of variables and almost always conflicting goals. A general principle is that, stability, speed and range can not be maximized at the same time [7]. Due to this fact, we need a method to quantify our needs and direct the design effort to those areas. In aerospace engineering, there are couple of methods to be used for this problem. One method is figure of merit method. In this method, different configuration of the designs are compared in predetermined aspects. A point is assigned to each combination and then they are summed up. Then best design of that configuration is selected. This method is selected since it is easier to use and with correct point assignment, it can provide highly accurate results. Also, it is also widely used in the industry as well.

A. Aircraft Configuration Selection

To create Figure of Merit(FoM) tables, first we must choose the importance of different aspects of the configuration. The following matrix is determined for our vehicle:

Design Factor	Figure of Merit
Simplicity	1
Weight	5
Speed&Drag	3
Stability	2

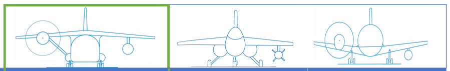
Fig. 3: Figure of Merit Matrix

We used this matrix to determine wing, tail and fuselage configuration.

1) Wing Configuration: First consideration for wing configurations is vertical location of wing.Ground clearance is important for precise take-off&landings. Since motors' vertical location is related with wing location, wing location is key factor for sufficient ground clearance. High wing is selected for this configuration.

1) High Wing: It will provide sufficient ground clearance for motors with landing gears. Also it is tend to be more stable than others
2) Mid Wing: It has structural disadvantages because bending moment produces by the wing must be carried across the mid section of the fuselage,which is weakest point of cross section.
3) Low Wing: Low wing is advantageous for landing gear structure, but providing adequate ground clearance is required longer landing gears.

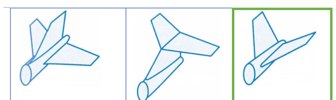
2) Tail Configurations: Aircrafts need an empennage for stability and control, so tail is vital part of an aircraft. There are various types of configurations. Practicable tail arrangement according to previous design selection are conventional tail, V tail, T tail. V tail is selected between them.



Design Factor	FOM Value	High Wing	Mid Wing	Low Wing
Simplicity	1	3	2	3
Weight	5	3	1	3
Speed&Drag	3	3	4	3
Stability	2	4	3	2
Payload Capacity	4	3	2	3
	TOTAL	47	33	43

Fig. 4: Wing Configuration [8]

1) Conventional Tail: It has simple structure and attaching the linkages to this type of tail is easier than others.
2) V Tail: It provide same control and stability with conventional tail.It can be used to lessen wetted area and total tail surface area.
3) T Tail: It works more effectively than conventional tail relatively ,but it necessitate more support to strengthen vertical wing.



Design Factor	FOM Value	Conventional Tail	T Tail	V Tail
Simplicity	1	4	2	3
Weight	5	3	2	4
Speed&Drag	3	3	3	4
Stability	2	3	4	3
	TOTAL	34	29	41

Fig. 5: Tail Configuration [8]

3) Fuselage Configuration: Third design configurations is fuselage design of aircraft. Design options are based commercial available and historical aircrafts. Tail Boom fuselage was selected between conventional, flying wing and tail boom aircraft.

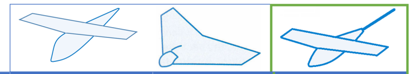
1) Conventional Fuselage: It is easy to design and has lots of example. It offers large volume in it.
2) Flying Wing: Flying wing is the lightest option but it has inadequate volume in it so it requires bigger scales for this project. Also it suffers from controllability issues and usually much harder to design than conventional counterparts.
3) Tail Boom Fuselage: It has less volume than conventional fuselage, while it is lighter than conventional fuselage. It requires more supporting structure to hold boom securely.

B. Conceptual Design Specifications

With the FoM method, we may see that our final configuration is tail boom, high wing, tilt motor tricopter VTOL vehicle.

To start the design, we need to have a prior performance criteria about the vehicle. This may include maximum take-off weight, maximum payload weight, minimum payload, minimum range required or minimum or maximum cruise speeds. To determine these specifications, we decided to use the METU VTOL 2020 competition criteria as our criteria. Advantages of using these criteria can be summed up as follows:

- Specifications are tailored with regards to the law governing the UAV's in Turkey. So if those specifics are followed, this VTOL vehicle can be used without special warrants from the authorities.
- Aircraft design can be heavily influenced by other similar aircrafts with similar performances. Tricopter VTOL is a relatively unexplored concept. Comparing our vehicle with other different configurations will help us validate the claims about tricopters and also gives us insight about the designs competing in the competition.
- Competition is sponsored by Boeing company so if we can successfully complete the design, we can receive financial compensation and the winner earns a spot in AIAA conference to display the work done. This can prove to be very benefical in academy.



Design Factor	FOM Value	Conventional	Flying Wing	Tail-Boom
Simplicity	1	4	2	3
Weight	5	2	4	3
Speed&Drag	3	2	3	4
Stability	2	4	2	3
Payload Capacity	4	4	1	3
	TOTAL	44	40	48

Fig. 6: Fuselage Configuration [8]

1) METU VTOL 2020 Scoring Criteria: According to competition documentation, aircraft should carry at least 0.2 kg payload, imitation of supplies, attached to the end of the 1 meter long rope and drop it to student lost in METU forest. Aircraft capable of dropping payload that weighs more than 0.2 will be awarded with more point. Overall mission score is calculated according 4 main objectives:

$$M_1 = \frac{\frac{W_p}{OEW}}{\left(\frac{W_p}{OEW}\right)_{best}} \qquad M_2 = \frac{N_{lap}}{(N_{lap})_{best}} \tag{1}$$

$$M_3 = \frac{\frac{N_{lap}}{MissionDuration}}{\left(\frac{N_{lap}}{MissionDuration}\right)_{best}} \qquad M_4 = \frac{K}{5} \tag{2}$$

W_p	:	Payload Weight
OEW	:	Operating Empty Weight
N_{lap}	:	Lap Count
$MissionDuration$	:	Time to Complete Mision
K	:	Drop Sensivity Factor[1,5]

According to main 4 objective, payload capacity to aircraft's weight ratio should be as much as possible while carrying out as much lap as possible in lowest mission duration. There is a significant trade in this mission. Increasing payload capacity may seem advantageous for scoring ,but more payload means extra volume attached plane which causes parasite drag. Also more payload capacity require more wing surface area. In addition, payload,extra weight for aircraft, necessitate more durable aircraft structure. Highest possible mission point is aimed and this is scoring equation:

$$score = (A + B)\,(M_1 + M_2 + M_3 + M_4)\,\frac{ReportScore}{OEW} \tag{3}$$

$$A = \begin{cases} 3, \text{ Aircraft Category1} \\ 1, \text{ Aircraft Category2} \end{cases} \qquad B = B_1 + B_2 + B_3 \tag{4}$$

$$B_1 = \begin{cases} 0, \text{ No autonomy in target detection} \\ 0.25, \text{ Autonomy in target detection} \end{cases} \tag{5}$$

$$B_2 = \begin{cases} 0, \text{ No Autonomy in takeofflanding} \\ 0.25, \text{ Autonomy in takeofflanding} \end{cases} \tag{6}$$

$$B_3 = \begin{cases} 0, \text{ no autonomy in transition to forward flight} \\ 0.25, \text{ Autonomy in transition to forward flight} \end{cases} \tag{7}$$

Two types of aircraft is allowed by competition. Both types of aircrafts are capable of vertical takeoff and landing but aircraft category1 can transit to forward flight with it's fixed wing that have sufficient wing area to sustain forward flight. When score is considered, coefficients A and B have great effect on overall point as operational empty weight has a negative inverse effect on overall point.

C. Aerodynamical Preliminaries

In this section, some aerodynamic parameters used in this paper is presented. Explanation of the parameters used here can be found at List of Symbols.

Wings are structures that changes the velocity of the oncoming air. Portion of the air that passes above the wing accelerates and due to Bernoulli's principle, faster air has less pressure. Since the air passing below the wing is not accelerated as much as the air passing above, a pressure difference occurs in two different regions. This difference results in lift generated by the wing. Lift is given by:

$$L = \frac{1}{2}\rho SCLV^2 \tag{8}$$

Every body moving through the air results in a drag. This is the force that slows the plane down. It is due to friction and other effects such as turbulance. It is calculated as:

$$D = \frac{1}{2}\rho SCDV^2 \tag{9}$$

Thrust generated by a propeller can be expressed as [2]:

$$T = \rho n^2 D^4 C_T \tag{10}$$

where,

$$J = \frac{V}{nD} \tag{11}$$

.

D. Conceptual Design Parameters

From the early days of aviation, engineers relied upon the previously built aircraft. Aircraft with similar specifications are used as a baseline for the design. In this work, this method is also followed when applicable. When the author decided that in few instances, historical data may not be sufficient alone, other techniques are employed.

Design process is started by first analyzing the scoring which is stated in the previous chapters. Since the aircraft is compared with other competitors, it is useful to have an estimate of best competitor performance and try to have better performance in terms of points. But aircraft design is a very involved process with a lot of conflicting priorities for different design goals. For example, we want to have a very stable aircraft for ease of operation and seamless transition between modes. But static stability may result in different fuselage design which in turn affect performance. Another example is we need a faster cruise speed for better points. Faster cruise speed means, there might be high G forces during cruise in the competition, which may mean increased mass for better structural integrity.

Instead of inspecting competitor aircraft and start with those estimates, we created an optimization routine in MATLAB environment which given the performance equations and estimated competitor performance, it outputs the performance points it would earn from the competition which are M1, M2, M3. These results are then fed into two different optimizers, one of them is a gradient based nonlinear optimizer and the other one is a genetic algorithm. Different outputs from different optimization methods are compared and we selected a best candidate from each optimizer. After comparing these two outputs, we decided that the genetic algorithm output is more promising and decided to implement this design as a goal.

Analysis is done with XFLR5 program and MATLAB. MATLAB is used to plot performance estimates and constraint analysis. XFLR5 is used for aerodynamic calculations as well as verification of the mission performance.

1) Constraint Analysis: In this section, plots and equations regarding the constraint space of the aircraft is presented. These constraints are in the form of thrust to weight ratio vs wing loading. Constraints of this type include takeoff, landing, rate of climb, cruise and level turn constraints. Since takeoff and landing are done with hover mode, we should only be interested in stall speed for takeoff and landing. Hence we only consider the remaining constraints.

For Constant Velocity Level Turn we have:

$$\frac{T}{W} = q[\frac{C_{D_{min}}}{\frac{W}{S}} + k(\frac{n}{q})^2\frac{W}{S}] \quad (12)$$

For Rate of Climb:

$$\frac{T}{W} = \frac{V_v}{V} + \frac{q}{\frac{W}{S}}C_{D_{min}} + k\frac{1}{q}\frac{W}{S} \quad (13)$$

For Desired Cruise Speed:

$$\frac{T}{W} = qC_{\mathrm{D_{min}}}\frac{1}{W/S} + k\frac{1}{q}\frac{W}{S} \quad (14)$$

In the equations above: q is dynamic pressure, n is loading factor.

For constraint analysis, mass is selected as 3.8kg, n as 2, rate of climb as 1 m/s since we don't need high climb performance and minimum induced drag coefficient as 0.08. Battery energy density is also taken as 150 Wh/kg.

Drag coefficient is calculated as:

$$C_D = C_{D_{min}} + kC_L^2 \quad (15)$$

Using historical data, minimum induced drag coefficient can be approximated as 0.03 for UAV's. But this historical data is retrieved from UAV's built with more professional techniques so due to manufacturing differences, our first estimate would be 0.04 for a fixed wing UAV. We are designing a VTOL aircraft so there will be additional drag due to additional components added. In [9] , it is found that %52 of the total drag is due to added VTOL components and %48 is due to conventional fixed wing UAV parts. In this paper, configuration is based on five engine VTOL and our three engine tilt configuration should result in a lesser induced drag. But for worst case analysis, we double the coefficient for a fixed UAV hence end up with 0.08. Thrust to weight ratio and wing loading related plots can be found in the Figures 7 and 8.

2) Determination of Critical Parameters: After calculating the constraints and related performance related plots, we may continue with the initial sizing and determination of the critical wing parameters aspect ratio, wing area and battery,payload mass.

As input to the optimization program, we selected aspect ratio, wing area, cruise speed, battery mass and payload mass. We did not specify total mass as input since it can be inferred from parameters chosen. For this, we need a sizing equation. For historical data, we consulted [5] as data source. This data is retrieved using 400 UAV's constructed for commercial, military or research applications. One shortcoming of this dataset is that these data is collected for fixed wing UAV's only hence not directly applicable to VTOL's. For a quick fix, we added 400g of extra weight to the estimated weight since we believe three motor configuration would result in weight difference close to this value. So our final sizing equation becomes:

$$M_{Empty} = 5.58(AR^{0.71})(Area^{1.59}) + 0.4 \quad (16)$$

$$M_{MTOW} = M_{Empty} + M_{Battery} + M_{Payload} \quad (17)$$

Oswald efficiency factor and k can be estimated as:

$$e = (1.78(1 - (0.045AR^{0.68}))) - 0.64 \quad (18)$$

$$k = \frac{1}{\pi ARe} \quad (19)$$

Using wing area and cruise speed, CL and CD values are calculated. Then given the battery mass, we must find the range aircraft has. Using only fixed wing aircraft range equations may result in dramatic difference between reality and estimation because hovering consumes a lot more energy than cruise. Estimating energy required in transition between modes is also very complicated so it is not used here. So we use a typical quadcopter energy consumption model for hovering 80 secs of estimated time. Remaining energy is then used for calculating cruising range in fixed wing mode.

Energy Required to Hover:

$$E_{Hover} = 22.5\frac{M_{MTOW}^{1.5}}{R_{prop}} * t \quad (20)$$

Range:

$$Range = \frac{(M_{Batt} * e_{Batt} - E_{Hover})\eta_{electrical}}{DragForce} \quad (21)$$

In above equations, we have t as 80 secs, propeller diameter as 15cm, battery energy density as 180 Wh/kg and electrical system efficiency as 0.8.

Using these relations, an optimization program can output different configurations but not all configurations are realizable. CL/CD ratios over 20 or CL over 1.5 is not very realistic assumptions. Hence, we set up nonlinear constraints over the variables. CL which is a nonlinear function of input parameters are limited between 0 and 0.5. Total weight must be below 4kg. CL/CD in cruise must be below 8 for realistic assumptions. Range also must be above 12km as another constraint. Cruise speed is limited between 10-25m/s. Aspect ratio is limited between 5 and 12. Wing area is limited between 0.2 and 0.4. Payload mass must at least be 200g. Battery mass must also be at least 200g.

To evaluate performance, we need an estimate of competitor performance. For this study, we assumed 18km of range, 17m/s cruise speed and 0.4kg/3kg payload to total weight ratio. Using all the relations above, output of two different optimization techniques are shown in Table I

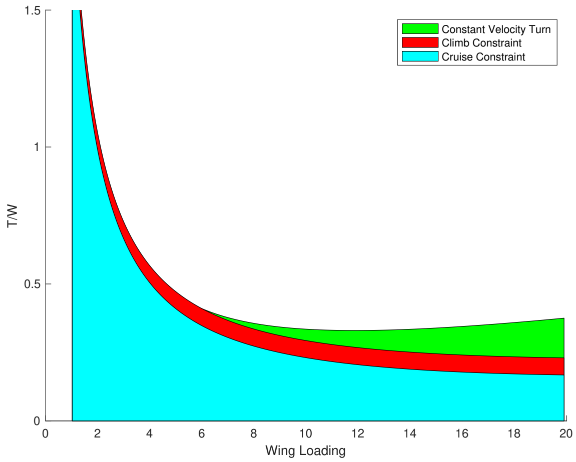
Using the scoring table, genetic algorithm output scores 2.49 points. This configuration is selected for manufacturing.

III. Preliminary Design

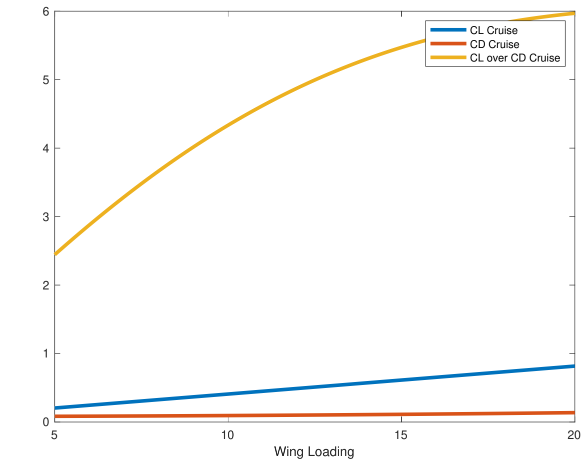
A. Design and Analysis Methodology

For designing the determined configuration, we used XFLR5 software for aerodynamics analysis. Structural design and part designs for manufacturing are done with SolidWorks CAD software.

First step of the design started by designing the wing using aspect ratio and wing area from the conceptual design. We

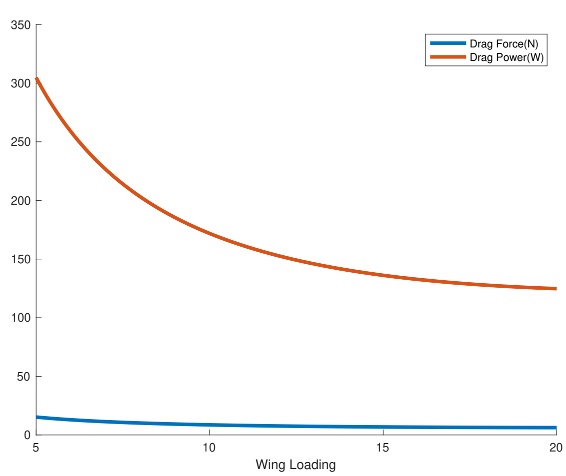


(a) $\frac{T}{W}$ vs Wing Loading Constraints

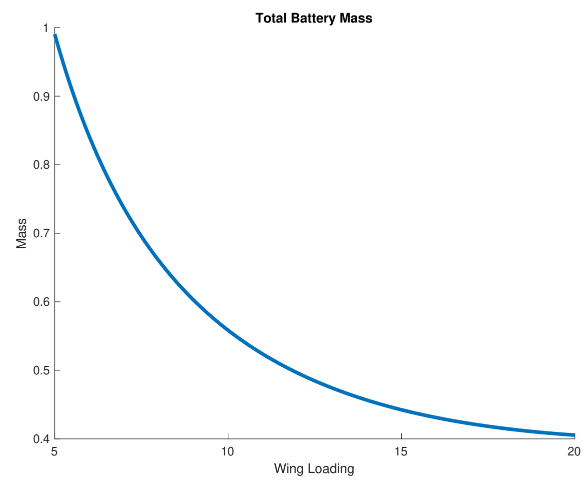


(b) CL, CD, CL/CD in Cruise.

Fig. 7: Constraints and CL/CD Plots vs Wing Loading



(a) Drag vs Wing Loading Constraints



(b) Battery Mass Required for 13km Cruise.

Fig. 8: Drag and Required Battery Mass vs Wing Loading

TABLE I: Optimization Techniques

Optimizer Type	AR	Wing Area	Cruise Speed	Battery Mass	Payload Mass
Nonlinear Gradient Based Optimizer	5.33	0.2	18.6209	0.4	0.6
Genetic Algorithm	5.9826	0.2877	20.54	0.4467	0.2132

will be using a rectangular wing with no twist or sweep since it will be much easier to manufacture. During pandemic outbreak, it is increasingly hard to find proper tools for manufacturing in campus environment. And using machining shops, CNC machines outside campus will require a lot of human interaction when travelling to the location and during manufacturing process. So considering pandemic conditions as well, we decided to use a rectangular wing. So for determined aspect ratio and wing area, we have 22cm of root and tip chords, 66 cm of half span.

Other important feature of the wing is the airfoil. For the determined wing parameters, we need target CL of 0.5. Reynolds Number for cruising at 20.5 m/s is 300000 for root and tip of the wing. High camber airfoils are avoided because they are relatively difficult to manufacture and may also result in increased drag when not operated in optimal conditions. We are more geared towards NACA 4-series airfoils since this team has experience building fixed wing aircrafts using NACA 4-series airfoils in wings and tail parts of the aircraft. Using the databases about NACA 4-series airfoils [8], we see that NACA4412 airfoil has 0.5 target CL. Its another desirable characteristic is that best CL/CD is near 0.5 CL value and this value is attained at low AoA. This means better cruising performance at low AoA angles. We target 20m as minimum cruising altitude so low AoA at cruising means aircraft will stay below 50m maximum altitude without much correction from the autopilot. Due to overwhelming advantages, NACA4412 airfoil is selected for wing. For tail surfaces, one of the most widely used airfoils are NACA0009 and NACA0012. We decided to use NACA0012 in the tail section.

Airfoil performance plots(Brown-NACA4412, Blue-NACA0012, Reynolds Number:300000) is given in Figure 9.

After the determination of both airfoils, we proceeded with the drawing of the wing in XFLR5 software. Since all the parameters regarding the wing is fixed by conceptual design, we continued with the tail. There is two important parameters regarding the tail design. These are vertical and horizontal tail volume coefficients and the calculation is given below:

$$V_{HT} = \frac{l_{HT} S_{HT}}{\bar{c} S} \tag{22}$$

$$V_{VT} = \frac{l_{VT} S_{VT}}{b S} \tag{23}$$

Due to [8], horizontal tail volume coefficient may be taken between 0.4 - 0.5, and vertical coefficient up to 0.05 [1]. Our design has a strong emphasis on stability so maneuverability is not so important. Hence we selected these coefficients as the maximum recommended values, namely 0.5 and 0.05. For now, we selected distance from the start of the wing to the start of the tail as 1m. So using these results, we have 0.033 meter square as horizontal tail area and 0.009 as vertical tail area. Since we are using a V tail configuration, we can use these areas as projections along different planes.

$$\Theta_{dihedral} = \arctan \frac{HorizontalArea}{VerticalArea} = 15deg \tag{24}$$

$$TailArea = \frac{HorizontalArea}{\cos \Theta_{dihedral}} = 0.036 \tag{25}$$

Aspect ratio of the tail is selected as 4 since it is a recommended value. After stability analysis, it is found that plane is not very stable in spiral mode. So for dynamic stability, 2 degrees of dihedral is also added to the wing So final design in XFLR5 software looks like the Figure 10

1) Static Stability and Performance Analysis: In this section, plots regarding static stability and performance analysis regarding the aircraft is presented.

For static margin, a value between 10-15 percent is satisfactory in most cases. From the analysis, we see that neutral point is 0.111 meters from the leading edge from the wing. We have MAC equal to root chord since we don't have sweep or twist. This results in a CG range between 7.8-8.9 cm. After analysis we selected CG position as 8.5cm from the leading edge of the wing.

Three different kinds of analysis is carried out in the XFLR5 program. First one is fixed lift analysis. Using 2-D panel methods, aircraft is subjected to different velocity and AoA pairs which always results in lift being equal to weight. Second analysis is fixed speed analysis. For this analysis, airspeed of the aircraft is fixed at 20.5 m/s and AoA of the aircraft is swept. Third analysis is the stability analysis. The velocity and AoA pair which the aircraft is trimmed is found and then eigenvalues and the eigenvectors of the dynamic modes are found. All other parameters such as frequency or damping ratio is found using eigenvalues and eigenvectors.

In all plots, a dot can be observed. This dot is generated by stability analysis and it is the point where the aircraft is trimmed for cruise. Starting from the fixed speed analysis, we can see that aircraft has a negative sloped Cm vs AoA graph and Cm is positive for zero AoA. This is a requirement for static stability and it can be verified that aircraft is statically stable in pitch. We can see verify from the fixed lift analysis is that aircraft is trimmed near 20.5 m/s cruise speed that we are aiming for. Another important take is that when we observe CL/CD plot for cruise conditions, we can see that trim point is at roughly 2 degrees of AoA and we are almost at the peak value of CL/Cd. This ensures really efficient cruise at the cost of minimum altitude change. This is a very desired property and we can see that it is satisfied.

2) Dynamic Stability: A fixed wing aircraft has two axis which we can inspect its behaviour. One of them is lateral axis and the other is longitudinal axis. Along longitudinal axis, we have two main modes: Phugoid and short mode. Along lateral axis, we have roll, dutch roll and spiral mode. Effect of these modes on the handling characteristics of the aircraft is complicated and one combination of eigenvalues may be more desirable as the operating condition of the aircraft changes. Due to these reasons, US Air Force developed a guideline for fixed wing aircraft in which we can quantify handling characteristics based on pilot experience. These specifications are called MIL-F-8785C. There are different classes of aircraft and different kinds of mission phases. For most small UAV designs, class 1 and flight phases of B and C should be assumed. For manual operation, flying quality of level 1 must also be assumed. This can be relaxed to level 2 or 3 if an autopilot is used since autopilot decreases pilot load significantly. [4]

Aircraft Classes:

- Class 1: Light Aircraft
- Class 2: Medium-weight, Medium-maneuverability Aircraft
- Class 3: Heavy, Low-maneuverability Aircraft
- Class 4: High-maneuverability Fighter

Flying characteristics are ordered from best to worst. B and C flight phases correspond to cruising and maneuvering.

For this specifications, for longitudinal axis, a summary can be in the following Table II .

Same analysis is carried out for lateral axis as well which resulted as Table III.

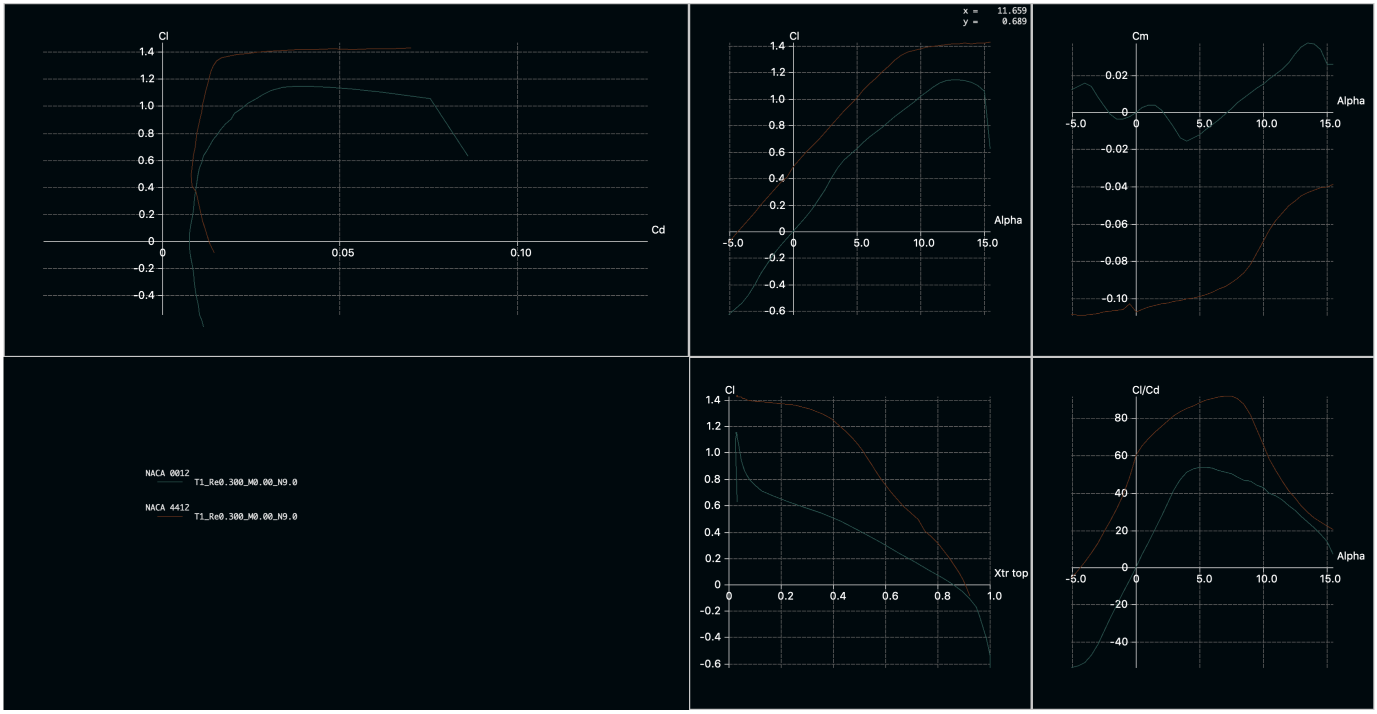


Fig. 9: Airfoil Performance Plots

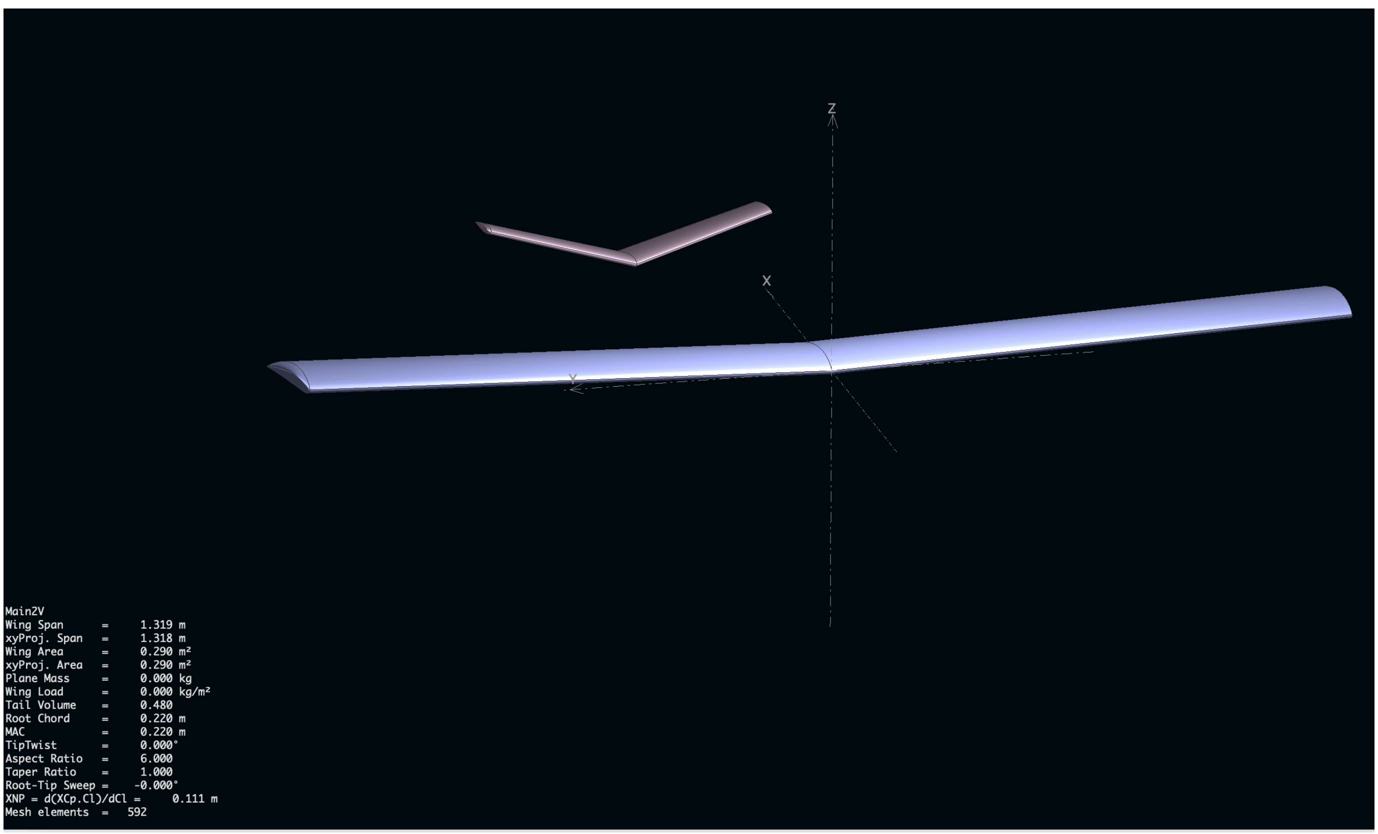


Fig. 10: Aircraft Preliminary Design

From dynamic stability analysis, we can see that we have desired performance for almost all required modes. Only mode that is problematic is phugoid mode. Its damping ratio is not high enough for best handling but XFLR5 underestimates drag

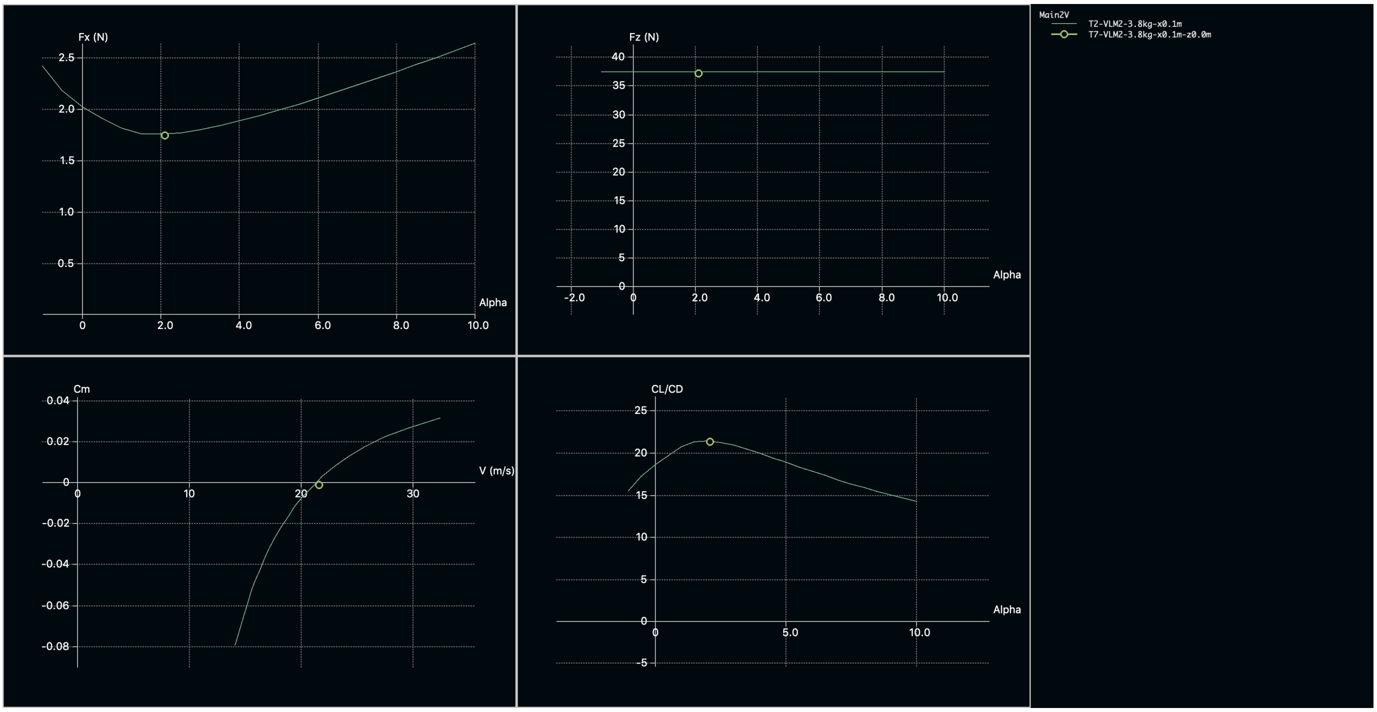


Fig. 11: Fixed Lift Analysis Plots

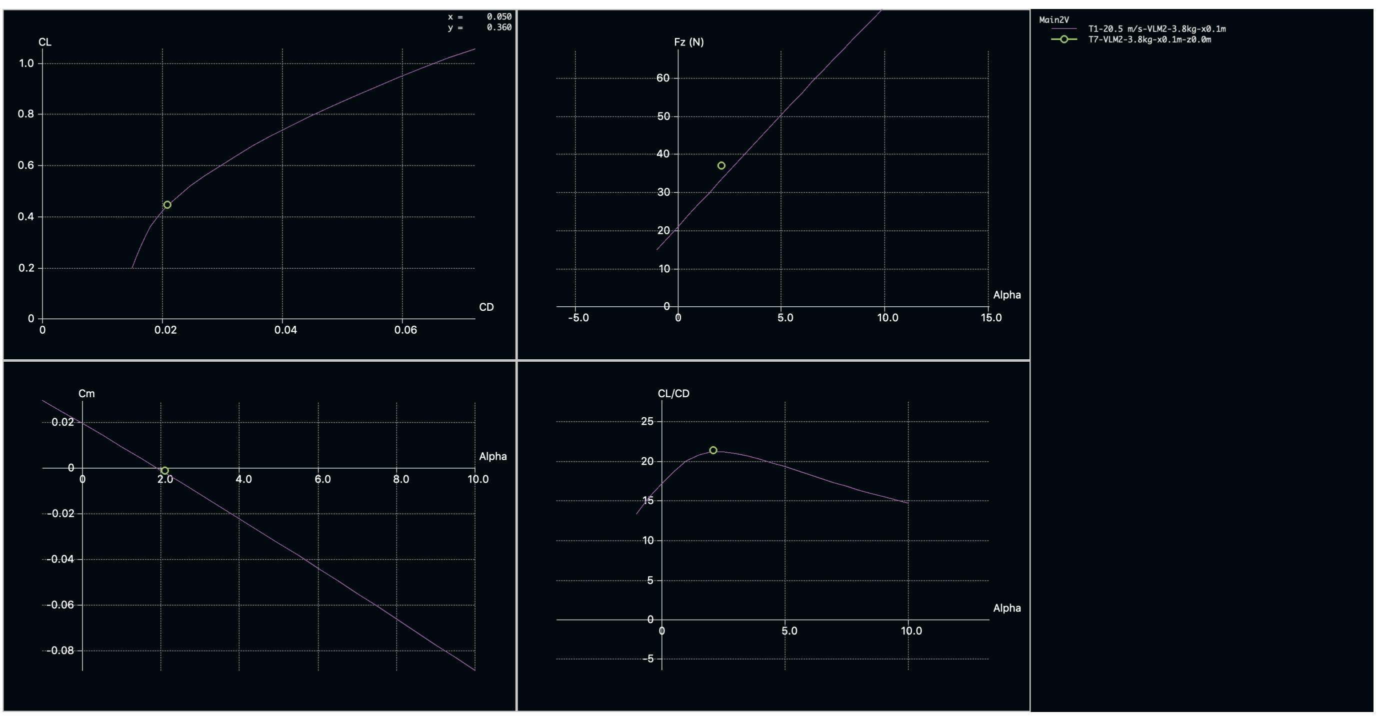


Fig. 12: Fixed Speed Analysis Plots

TABLE II: Longitudinal Analysis

Mode Name	B Phase	C Phase	Analysis Result
Short	0.3-2.0	0.35-1.3	0,729
Phugoid	>0.04	>0.04	0.029

and fuselage and other body parts are not included in the drag estimation. With the added induced drag, phugoid mode will have much higher damping ratio which we assume will be satisfactory.

TABLE III: Lateral Analysis

Mode Name	B Phase	C Phase	Analysis Result
Roll(Maximum Time Constant)	<1.4	<1	0,06
Dutch Roll(Damping-Frequency)	0.08 - 0.4	0.08 - 1.0	0.238 - 1.4
Spiral(Time To Double)	>20	>12	16

B. *Battery And Motor Selection*

Battery plays a very important role in the overall design. It effects very crucial mission performance. Heavier battery provides more range but aircraft becomes heavier hence results in a less efficient vehicle. Also voltage of the battery is important. To provide same power, a lower voltage battery outputs a larger current which increases losses in the electrical system. Higher currents might also mean heavier motor controller electronics. But higher voltage usually means that electric motors becomes heavier and more powerful. More powerful engine might not be necessary if a smaller engine is sufficient enough to provide enough lift. To effectively determine battery size and battery technology, research has been done on the commercially available Lithium Ion and Lithium Polymer batteries. Lithium Ion batteries is more energy dense but their current output is very low compared to Li-Po counterparts. This means more cells in parallel are needed to supply enough current. Li-Po batteries are capable of outputting enormous amount of currents but they lack the capacity to sustain longer duration flights. With these aspects in mind, following table is created to compare two technologies with different capacities and voltages.

First of all Li-Ion battery table is given in Table IV. While making this table LG-HG2 18500 type battery choose as a reference. Since each battery have high capacity and maximum discharge ampere rate (3000 mAh capacity and 20A max. discharge rate), this battery type is chosen as a reference.

Secondly Li-Po battery is given in Table V. Since it cannot be modified not all S and P combinations listed. Two brand is used which are Gens and Leopard because of the safety reasons.

With these tables in mind, we chose Li-Ion technology since we can get almost twice as much energy density from these batteries. Since optimization routine determined that the battery should be roughly 500g, we searched for engines capable of running in 4S configuration. Current requirements of these motors were higher than the current battery could provide. So we increased the battery to 4S4P configuration and used Sony VTC5A cells. These cells can output 30A continuously and has 2800mAh capacity [3]. Features of the motors is given in the Table VI.

C. *Modeling Of The System*

In this section, modeling techniques used in this project is explained. Firstly, aircraft is modeled generically meaning forces present in all flight phases are used. Secondly, based on the flight phase such as takeoff, landing or forward flight, forces that are dominant are simplified. This allows for one general model to predict behaviour in all flight phases.

General model used in this work can be found in [6]. PID and LQR based controllers also proposed in this paper are used in this work. These controllers are used for controlling the vehicle in hover mode i.e. in takeoff or landing period of the flight.

1) Body Frame and Forces on the Body Frame: Before expressing the forces, a suitable coordinate frame must be specified. For this model, we use body frame. Body frame can be expressed as in Figure 13:

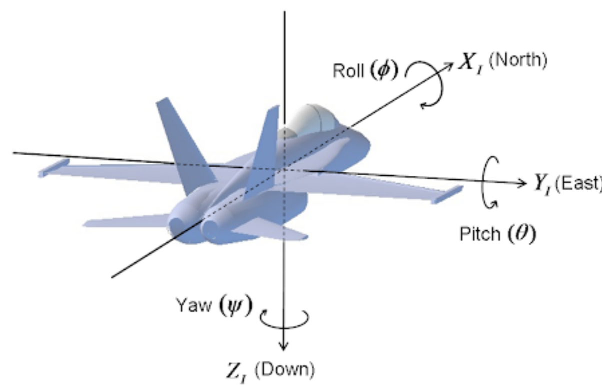
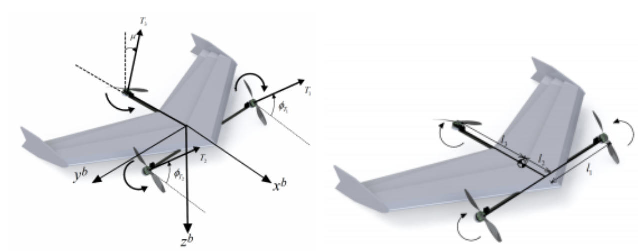


Fig. 13: Body Coordinate Frame

Using this frame, forces applied on the body can be expressed as:



(a) Forces on the Body

(b) Moment Arm of the Motors

Fig. 14: Forces On The Body**??**

$$\mathrm{F}_T = F_{T_1} + F_{T_2} + F_{T_3} \tag{26}$$

$$\begin{bmatrix} F_{T_x} \\ F_{T_y} \\ F_{Tz} \end{bmatrix} = (T_1 + T_2) \begin{bmatrix} \cos\phi_T \\ 0 \\ -\sin\phi_T \end{bmatrix} + T_3 \begin{bmatrix} 0 \\ -\sin\mu \\ -\cos\mu \end{bmatrix} \tag{27}$$

S Value	P Value	CDR Rate (A)	Capacity (mAh)	Weight (g)	Energy Density (Wh/kg)
4S	4S1P	20	3000	188	225
	4S2P	40	6000	376	225
	4S3P	60	9000	564	225
	4S4P	80	12000	752	225
	4S5P	100	15000	940	225
	4S6P	120	18000	1128	225
5S	5S1P	20	3000	235	225
	5S2P	40	6000	470	225
	5S3P	60	9000	705	225
	5S4P	80	12000	940	225
	5S5P	100	15000	1175	225
	5S6P	120	18000	1410	225
6S	6S1P	20	3000	280	225
	6S2P	40	6000	564	225
	6S3P	60	9000	846	225
	6S4p	80	12000	1128	225
	6S5P	100	15000	1410	225
	6S6P	120	18000	1692	225

TABLE IV: Li-Ion Battery Comparison Table

S Value	P Value	CDR Rate (A)	Capacity (mAh)	Weight (g)	Energy Density (Wh/kg)
4S	4S1P	66	2200	267	122
	4S1P	100	4000	439	134
	4S1P	250	10000	942	157
5S	5S1P	123	1300	201	120
	5S1P	112,5	4500	600	139
	5S1P	280	7000	915	142
6S	6S1P	200	5000	795	140
	6S1P	400	10000	1250	178
	6S1P	480	12000	1570	170

TABLE V: Li-Po Battery Comparison Table

Internal Resistance	38 $m\Omega$
Motor KV	1000 KV
Voltage supply	14.8 V
Idle current	1 A
Maximum continuous current (180s)	40 A
Compatible Propeller	APC 11x5.5
Max Thrust	2330g
Weight (incl. cable)	108 g

TABLE VI: BLDC Motor Parameters

$$\begin{bmatrix} L_T \\ M_T \\ N_T \end{bmatrix} = \begin{bmatrix} (T_1 - T_2)\, l_1 \sin\phi_T \\ -T_3 l_3 \cos\mu + (T_1 + T_2)\, l_2 \sin\phi_T \\ T_3 l_3 \sin\mu \end{bmatrix} \tag{28}$$

With these forces and moments in mind, dynamics of the system is found as:

$$\begin{aligned} m\ddot{x} &= u_1 \sin\theta \\ m\ddot{z} &= u_1 \cos\theta + gm \\ \ddot{\theta} &= u_2 \end{aligned} \tag{29}$$

where u_1 is F_{Tz} and u_2 is $\frac{1}{I_{yy}} M_T$.

Based on this model, a nested controller structure is proposed. First controller controls the total thrust exerted on the body. Second controller adjusts the difference between the front and back motors to balance the pitch angle. Also, the second controller is an LQR controller.

$$u_1 = \frac{k_p e_z + k_d \dot{e}_z - mg}{\cos\theta} \tag{30}$$

For large T, altitude error tends go to zero and pitch angle stabilized leaving it close to zero. With these observations, state space representation becomes,

$$\begin{aligned} \ddot{x} &= -g\theta \\ \ddot{\theta} &= u_2 \end{aligned} \tag{31}$$

If we define our state as $\begin{bmatrix} x & \dot{x} & \theta & \dot{\theta} \end{bmatrix}$, our system matrices becomes,

$$A = \begin{bmatrix} 0 & 1 & 0 & 0 \\ 0 & 0 & -g & 0 \\ 0 & 0 & 0 & 1 \\ 0 & 0 & 0 & 0 \end{bmatrix}, B = \begin{bmatrix} 0 \\ 0 \\ 0 \\ 1 \end{bmatrix} \tag{32}$$

An LQR controller is used to find the K matrix such that $u_2 = -Kx$.

2) Propeller Model: For accurate simulation, also the model of the propeller is also incorporated into the model. With this addition, it can be determined if the thrust required can be supplied with the motors we currently have. Also, propeller model incorporates saturation in the RPM of the engine which also allows for more accurate simulation.

A linear model of APC 10.5 propeller's coefficient of thrust can be estimated as:

$$C_T = \begin{cases} 0.095 & 0 < J < 0.07 \\ -0.16379J + 0.10647 & 0.07 < J < 0.65 \\ 0 & J > 0.65 \end{cases} \tag{33}$$

where the thrust and J is defined in the aerodynamics preliminaries section. Our motors have a maximum RPM of 16800. So RPM output of the motors are also saturated at this value.

3) Simulation Results: In these tests, vehicle's initial state is $x = -2$, $z = 0$ and $\theta = 0$. Subsequent plots show the trajectory of different aspects of the system. We can see that this controller structure is capable of both altitude and pitch control. Pitch control also results in control in the x direction.

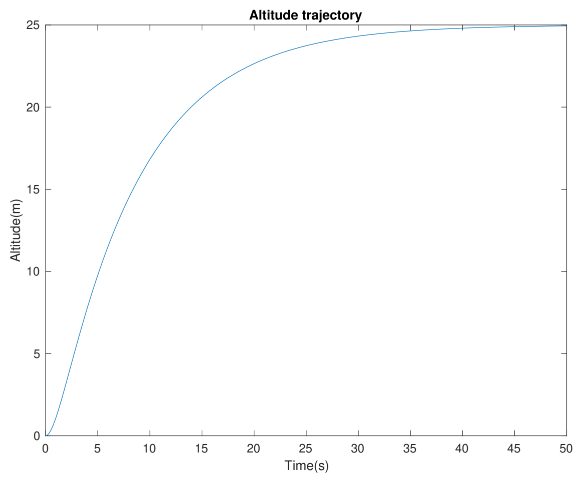


Fig. 15: Altitude vs Time

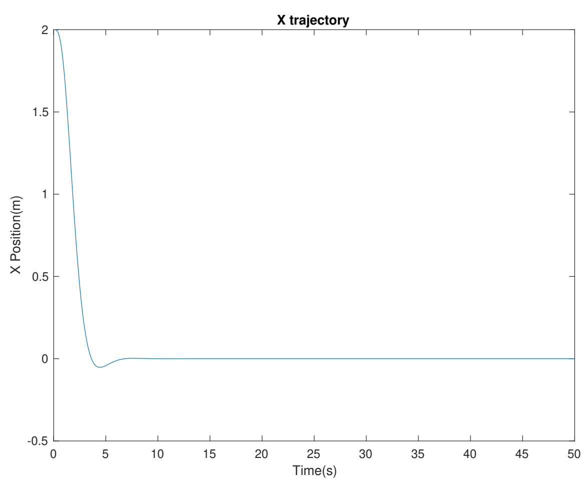


Fig. 16: X Position vs Time

D. Future Work and Conclusion

In this work, I presented our Tricopter VTOL project. This type of VTOL vehicle offers weight savings and potential

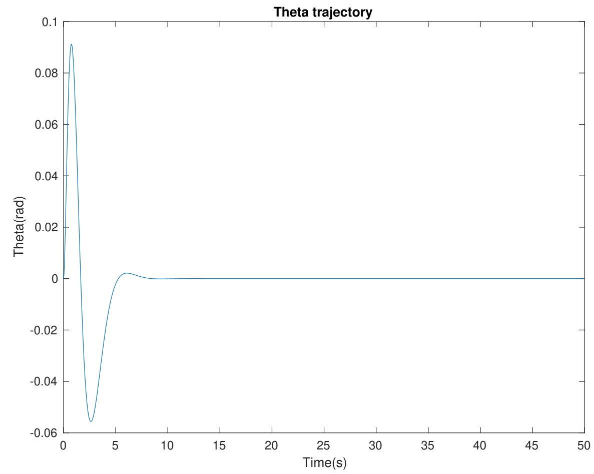


Fig. 17: Pitch Angle vs Time

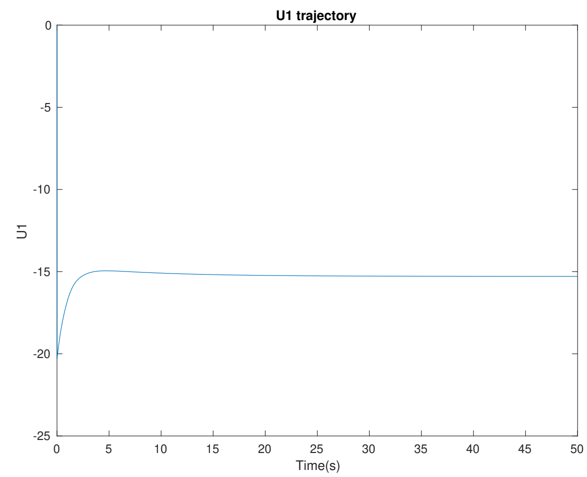


Fig. 18: U1 Output vs Time

efficiency increase with the cost of increased complexity of design and control. All stages of theoretical design is presented and simulation results are shown. We as a team, keep on building this vehicle and carry on real life flight tests.

Project still has room for improvements. Future work includes:

- Optimization techniques are run separately. By using the output of the genetic algorithm as initial condition in gradient descent solver, we can avoid local minimas in that solver more easily.
- Optimization method relies on a handful of equations. By incorporating tail equations, fuselage estimates and engine data or estimate, optimization output will converge to a significantly better solution. So more dynamic models should be integrated.
- Tilt mechanisms are supposed to be light so that total weight of whole setup is lighter than that of the five engine setup. Real tilt mechanism is heavy and almost

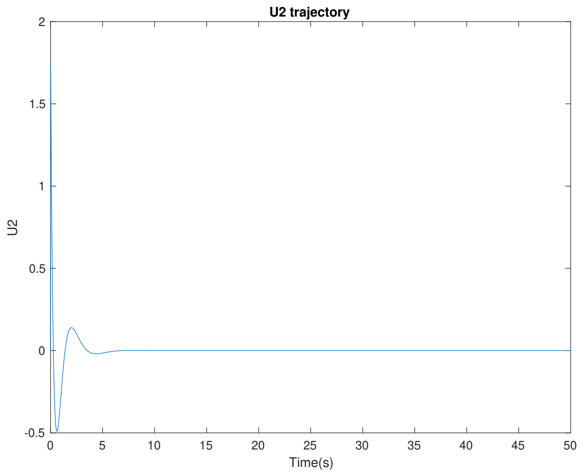


Fig. 19: U2 output vs Time

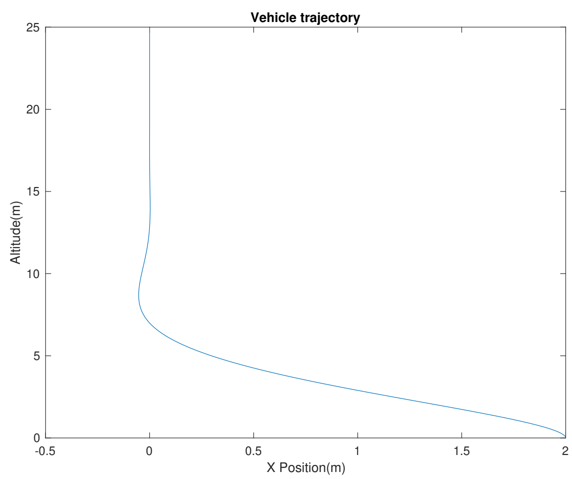


Fig. 20: Trajectory of the Vehicle in X-Z Plane

equalizes both systems in weight.

- Controller simulation is sufficient for hover but other flight regions must also be simulated and controlled. Especially, transition phase is really tedious and PX4 flight stack only offers an open loop solution. This controller must be improved.
- Further flight tests are needed for data acquisition. Simulated data and real system may diverge since aerodynamics simulation that I used is a simple, linear solver. Also, complex interactions between mountings,wings, propellers are can not be simulated in this program. System identification methods are useful and might be necessary for better controller design.
- Aircraft has $2.5\,\mathrm{deg}$ of pitch angle when in cruise. This results in altitude gain. With a better trajectory planning, this altitude gain can be exchanged for kinetic energy in turns. This can significantly reduce energy consumption in turns.
- Aircraft carries a load attached with a one meter rope. Hence load can alter the dynamics of the system significantly if its torque is high enough. Simulations must be further advanced to incorporate slung load models. This will also allow the user to analyze the maximum amount of moment that the load must have for a stable system.
- Slung load can swing freely in flight. In relatively tight spaces an aircraft can be, this may be a problem. It may also be the case that load is sensitive to swing so it must stay as still as possible. This results in a problem such that how can we change our course so that swinging of load is minimum. A controller can be proposed to overcome this problem.
- Inertia of the system is crucial for accurate simulation. Right now, estimation methods are used to determine inertia by historical data. Using detailed CAD design, inertia can be calculated more precisely.

REFERENCES

[1] Ae 451 aeronautical engineering design i geometry sizing configuration. [Online]. Available: http://www.ae.metu.edu.tr/~ae451/geometry_sizing&configuration.pdf

[2] Performance of propellers. [Online]. Available: https://web.mit.edu/16.unified/www/FALL/thermodynamics/notes/node86.html

[3] Sony vtc5a specification sheet. [Online]. Available: https://voltaplex.com/media/whitepapers/specification-sheet/Sony_VTC5A_Specification_Sheet.pdf

[4] J. P. S. Andrew J. Keane, András Sóbester, “Small unmanned fixed-wing aircraft design: A practical approach,” 2017.

[5] J. J. A. Emilio Botero, “Conceptual design and optimization of small transitioning uavs using suave.”

[6] R. L. Jose A. Bautista, Antonio Osorio, “Modeling and analysis of a tricopter/flying-wing convertible uav with tilt-rotors,” 2017.

[7] M. W. Karl Nickel, *Tailless Aircraft In Theory And Practice*, 1994.

[8] D. P. Raymer, “Aircraft design: A conceptual approach,” 2018.

[9] C. T. Watcharapol Saengphet, “Conceptual design of fixed wing-vtol uav for aed transport,” 2016.