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DEVELOPMENT OF AN EFFICIENT VERTICAL TAKEOFF AND LANDING AIRCRAFT

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DEVELOPMENT OF AN EFFICIENT VERTICAL TAKEOFF AND LANDING
AIRCRAFT
BY
ALEXANDER DESILETS

A THESIS SUBMITTED IN PARTIAL FULFILLMENT OF THE
REQUIREMENTS FOR THE DEGREE OF
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OF

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Abstract

Most of the unmanned flight systems that exist today are comprised of either horizontal or vertical capabilities, with very few capable of full Vertical Takeoff and Landing (VTOL) operations. Aircraft with VTOL flight systems have the ability to take off and land vertically, then transition to horizontal flight, allowing an aircraft to cover long distances at high speed while maintaining the highly advantageous ability to take off and land without the use of a runway. These systems, however, are either highly complex and costly, or power inefficient during horizontal flight, highly reducing their practicality to commercial or private applications. With small Unmanned Aerial Systems (UAS) becoming increasingly popular in private, commercial and military markets, simplified, small scale VTOL systems will provide UAS pilots with increased capabilities and significant advantages compared to standard fixed wing or rotor aircraft. A flight system designed for this application will be able to achieve VTOL capabilities and retain the high velocity and long range of conventional fixed wing aircraft while maintaining a comparatively low complexity and cost. To recognize these goals, a design has been established with a “dis-similar” tri-rotor design. This dis-similar thruster design will use powerful vertical lift motors in pods mounted in the wings capable of rotating forward for transition to horizontal flight, with a significantly smaller rear motor, in a similar pod, to provide low power, high efficiency thrust during horizontal flight operations. Several iterations of this design were constructed and tested with progressively more success with each design. The system was not able to achieve a successful

transition to horizontal flight; however, vertical flight capabilities were proven and significant data was collected to aid the design of future iterations.

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Alexander Desilets

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List of Terms

Airfoil:	Cross section shape of a wing
BEC:	Battery Eliminator Circuit, voltage regulator for control electronics
CAD:	Computer Aided Design software
Coefficient of Drag (CD):	Measure of an airfoil's drag at given conditions
Coefficient of lift (CL):	Measure of an airfoil's effectiveness in creating lift
ESC:	Electronic Motor Speed Controller.
GPS:	Global Positioning System
IMU:	Inertial Measurement Unit
PID:	Proportional–Integral–Derivative
Pitch:	Rotation about and axis wingtip to wingtip through the center of gravity
PPM:	Pulse Position Modulation, common transmitter control signal
PWM:	Pulse Width Modulation, common servo control signal
Roll:	Rotation about an axis extending nose to tail through the center of gravity
Stall angle:	Angle at which a wing stops producing lift at a given airspeed
Tri-Copter/Tri-Rotor:	Vertical flight system using 3 vertical thrusters in a triangle
UAS:	Unmanned Aerial System
UAV:	Unmanned Aerial Vehicle
VTOL:	Vertical Take Off and Landing
Wing Loading:	Lift force required by wing per area unit (oz/sqft)
Xfoil:	Airfoil simulation program to find appropriate coefficients
Yaw:	Rotation about an axis top to bottom through the center of gravity

Significance of the Study

In the current market for commercial and private UAVs, there are many systems available with vertical takeoff capabilities. These are commonly referred to as “Drones” or more accurately quadcopters, hexacopters, etc. While these aircraft do offer an inexpensive and simple way of achieving vertical flight, their efficiency decreases rapidly when moving horizontally due to the lack of lifting surfaces and complete reliance on spinning propellers to generate the required lift. This means that these aircraft generate horizontal propulsion by tilting toward the desired direction, which in turn requires more thrust to stay at altitude due to trigonometric thrust losses in the vertical direction. Extended increased battery draw caused by fast horizontal flight will usually reduce the flight time of these UAVs to under 10 minutes. Similarly, the reliance on thrust to maintain flight severely reduces the opportunity to add sizable payload without significantly reducing the flight time.

The complexity and cost of the transitional VTOL systems found in full size aircraft, such as the Bell-Boeing V-22 Osprey, or the McDonnell Douglas AV-8B Harrier II, results in these technologies rarely being implemented in small UAV applications, and when they are, the cost of the UAV balloons beyond practical for anything other than military or extremely high-end commercial use. Most small unmanned aircraft with transitioning VTOL systems are developed by mixing a quadcopter and a fixed wing aircraft into one flight system, in which four vertically oriented propellers will lift the aircraft, then secondary propellers will spin to move the plane forward, allowing the VTOL propellers to shut down to save battery. While this solution offers extreme simplicity, overall practicality, efficiency and scalability are lacking. Disadvantages with this system stem from the inability to use the VTOL thrusters in any meaningful manner during horizontal flight, requiring the aircraft to carry four extra propulsion systems which

are only used during the very beginning and end of a long range flight. This effect will also reduce the overall payload carrying capacity of these types of drones for long duration flights.

Literature Review

Development of VTOL flight systems has a long history with countless stories of success and failure. With each attempt, there are lessons to be learned and applied to future research and development. New technologies, such as highly efficient electric motors and far more powerful microcontrollers, have also made concepts that may have been ruled out in the past more possible in the present. As the literature review proceeded, gained knowledge was used to focus continued research towards work in specific applications of VTOL technologies relating to the developing concept.

Development of a VTOL flight system requires an understanding of the platform's missions and applications, so that reasonable requirements can be developed to align to its use cases. Current use cases range from commercial use, with forestry observation and photography as popular examples, to military use with strike and reconnaissance capabilities unavailable in conventional aircraft, as seen in Yu, Heo, Jeong and Kwon (2016) analysis of multiple countries and their current VTOL UAV developments. This study also explored the capabilities each of the aircraft excelled in, further representing the design goals of these various countries and companies. The most common prioritization was that of extended range and flight endurance, rather than speed, vertical flight endurance or payload capacity. Fredericks, Moore and Busan (2013) also worked under NASA focusing on long endurance flight for VTOL systems to supplement a development effort by Boeing to create a VTOL aircraft with 24 hour flight endurance. This effort was comprised of concept generation and testing in order to explore options to achieve this capability with transitioning VTOL aircraft resulting in a greater efficiency than conventional VTOL aircraft such as the helicopter design Boeing was pursuing.

With many VTOL solutions, a common occurrence is to have the concepts generation run away far past the current technological solutions, often leading to higher

risk development programs. This was also the case for one of the most successful and famous VTOL aircraft commonly used today, the V-22 Osprey. Maisel, Giulianetti and Dugan's (2000) overview of the XV-15 concept, which functioned as a technology development platform and demonstrator for what became the V-22, detail the hardships experienced with developing a flight system before the math and models existed to analyze it. Most notably, the use of large propellers could create dynamic instabilities during flight without the computers and control systems available at the time to counter the effects. Added complexity arose from the need to use constant speed engines in order to achieve the desired thrust and efficiency. The result was a pair of highly complex rotor controls at each wingtip, with the equivalent control system requirements of attaching two helicopters to an aircraft in the middle.

Since the era of the XV-15 experimental aircraft, there has been significant advances in electric motor and battery technology that makes new VTOL concepts possible without significant increases in complexity (Stoll, Bevirt, Pei and Stilson, 2014). Aircraft design variety potential and simplicity is increased with the fact that thrust developed by electric motors is more quickly varied, eliminating the need for complex and heavy variable pitch propellers to achieve the same effects (Moore, 2010). Furthermore, the nearly fixed mass to thrust ratio of electric motors, and fixed energy density of batteries means there is little penalty to using many smaller motors, compared to fewer larger motors, and artificially decreasing the propeller loading during vertical flight, without the disadvantages of maintaining the lower propeller loading during horizontal flight by shutting down unneeded motors (Fredericks et al., 2013). Electric motor propulsion also reduces the number of subsystems and maintenance required with no requirement for fuel flow systems, vibration mitigation or throttle actuators,

significantly reducing the design complexity and both initial and life cycle costs (Moore, 2010).

Fredericks, Moore and Busan (2013) also focused their work around the fact that there is significant difficulty in balancing vertical flight load capabilities and efficiencies with that of horizontal flight capabilities, in which large rotors significantly increase vertical flight efficiency at the cost of horizontal flight endurance. A further analysis effort by Uber (Fast-Forwarding, 2016) found that an increase in propeller loading, or “disk loading” resulted in an exponential increase in the power required to achieve a 1:1 thrust to weight ratio, the minimum requirement for vertical takeoff. Conversely, it was found that the higher disk loading required for efficient horizontal flight would be desirable if vertical flight maneuvers were completed in minimal time. These effects together give a desirable disk loading of less than 50 lbs/ft².

A common and tempting design used to develop a VTOL aircraft is that of a “Tail Sitter” configuration. This layout lands and takes off on its tail, and transitions by rotating the aircraft 90 degrees to horizontal flight and back. The tail sitter has many advantages that makes it appear to be a desirable solution including fixed motor orientation and in stream control surfaces, reducing design complexity (Moore, 2010). However, the tail sitter design has significant disadvantages that preclude the use of this design for a system to operate in a wide variety of environments. Foremost is the inability to accomplish short takeoff and landing maneuvers, limiting the aircraft to vertical takeoff, even when not required. Lacking this capability reduces the overall effectiveness of the aircraft from a mission perspective as maximum takeoff weights and bad weather survivability are diminished (Anderson, 1981). Tail sitters also require a complex transition from horizontal flight with a rapid vertical ascent to reduce aircraft speed, followed by a gradual descent to landing, requiring long duration vertical flight upon

landing. Finally, the reliance of thrust stream control surfaces for propeller driven tail sitter aircraft means that the control effectiveness would be drastically reduced when the thruster was at its lowest power state right before landing, resulting in control authority issues when precision control was most important (Maisel et al., 2000).

After all of its difficulties and problems, the XV-15 concept, and subsequent V-22 operational aircraft became one of the most successful and survivable VTOL aircraft in history (Maisel et al., 2000). This fact focused the research on design elements that could make use of the best aspects of the twin tilt-rotor configuration, while reducing the extreme complexity and high costs associated with the V-22 system. A large driver of the complex control systems for tilt rotor aircraft was the requirement to provide pitch control with the use of variable pitch propellers, similar to helicopter controls (Anderson, 1981). Tri-thruster designs negate the need for these complex controls by combining thrust vectoring and thrust modulation to achieve the required controllability (Mohamed & Lanzon, 2012). The use of a tri-rotor design also maintains the advantages of the V-22 design with a minimal number of propellers with low disk loading used for vertical flight. While the use of tri-thruster designs have experienced many difficulties in the past, commonly pertaining to the inability to rapidly modulate thrust (Anderson, 1981) the recent increased performance of electric systems allows for this capability to be integrated.

Research was further focused towards the mechanisms and controls required for a simplified tilt rotor configuration, with and without 3 thrusters. A study of vertical flight performance was conducted in which a combination of thrust modulation and thrust stream control surfaces provided 3 axis control, (Escareño, Salazar & Lozano, 2006). While a stable vertical flight platform was achieved, transition was never tested and the control system relied heavily on center of gravity position being directly in-line with the

thrusters. A more unusual, but drag reduced, method of twin thruster control was discovered in a patent number US6719244B1, in which manipulation of the left-right angle of the thrusters is used to induce control by differential induced torque on the airframe (Gress, 2004). This represents a significantly more complex mechanical design and control system and represents a high level of technology risk. Yoo, Oh, Won and Tahk (2010) conducted research on different configurations of tri-rotor control implementations to achieve stable and efficient vertical flight. Two options were introduced, thrust vectoring of a single motor and coaxial propellers with speed variation to induce yaw torque. While their simulations showed that both solutions represented a stable design, the thrust vectoring solution had quicker and more stable responses to inputs.

One of the significant technological risks that drove the design of the aircraft was the ability to develop a stable control system with a wide range of stability for variable payload integration. A strong influence to the control system design is the interface between the pilot and the control system and the amount of authority the computer is given. A variety of control interface solutions have been researched and developed in a range of different studies. Theys, De Vos, Leuven, Leuven and Belgium (2016) studied a control interface in which the reference frame would shift with the orientation of the airframe during transition. This system allows an “up” command to result in upwards movement no matter the flight mode of the aircraft, providing extremely intuitive controls. However, in practice, while this system proved successful in normal flight conditions and offered a very simplistic transition control, an unrecoverable state would evolve at the edge of the flight envelope during rapid descents. A different solution was proposed by Casau, Cabecinhas and Silvestre (2011) in which the controller and control scheme would shift to different states depending on the transition state of the aircraft. They found that a stable controller transition from state to state was possible as long as points of

equilibrium were used in which each controller represented a similar control output to the next controller at the moment of switching. This type of control input-output allows for conventional controls in each state where quadcopter type controls can be used during hover, and aircraft controls can be used during horizontal flight. Mode switching control was also explored by Çakici and Leblebicioğlu (2016) with a combined quadcopter, fixed wing aircraft. They chose to implement smoothed control outputs between state transitions in order to negate the need for equilibrium control points. While this controller offers a wider operational envelope during the transition phase, it does not guarantee optimal control.

Computer authority over pilot inputs during complicated maneuvers is another strong driver of control system design. While there is a near industry standard for vertical flight controls and horizontal flight controls, transition flight represents a period of time in which the pilot can quickly become overwhelmed by the inputs required. Muraoka, Okada, Kubo and Sato (2012) studied this effect and the ways in which giving control authority over to the computer running the control system may help reduce the burden to the pilot. The study concluded that the pilot was much more comfortable with reduced control of the transition angle and pitch of aircraft so that they could focus on the more important factors of forward speed and heading. It was also found that the control system was able to provide a more constant and predictable flight profile when compared to a burdened pilot. This study also focused on computer controlled, mixed output control during transition phases with automatic controller switching, and found that this controller design further alleviated the workload for the pilot during transition.

Many versions of control system design were found among the different studies, each with varying levels of success and implementation difficulty. Of the different methodologies, two type of solutions were used, systems that relied heavily on known airframe dynamics, and tuned systems that relied more heavily on sensor data fusion.

Kriel (2008) tested Linear Quadratic Regulator (LQR) and Time Scaled Decoupled (TSD) control for VTOL flight and compared the implementation and results. The study found that while the LQR system was able to use a more simplified model of the aircraft dynamics, its use required significant tuning, made very difficult by the naturally unstable characteristic of VTOL flight. Conversely, the TSD implementation was able to create a very stable system, but was heavily reliant on aircraft dynamics and was very susceptible to instability due to dynamic model inaccuracies. Similar testing was completed between a linear control system and a nonlinear control system implementation by Casau, Cabecinhas & Silvestre (2010). In the case of this study, both solutions were studied using simulations with known dynamic models and were able to show that both designs were viable. However, the control system design was never implemented on a real world prototype, so it is difficult to determine if the success would translate to real world dynamics, where more assumptions of dynamics must be made.

Several VTOL systems relied heavily on high frequency sensor data with a simplified control system. One such study by Çakici and Leblebicioğlu (2016) used a simple multi-loop tuned Proportional–Integral–Derivative (PID) controller, as the control system. While optimal control and closed loop stability are not ensured with a PID controller, requiring tuning to achieve acceptable control, it does not require assumptions to be made of the aircrafts dynamic model. Furthermore, not requiring a dynamic model make it more possible to apply this method to real-world scenarios. The issue of the single input, single output nature of PID controllers, compared to the multi input, multi output control required by an aircraft, was resolved by using a multi-layer loop design. Real world testing showed success with expected responses when compared to simulation testing, validating the use of a PID controller.

With high efficiency horizontal flight as a primary goal of this development study, research was also focused on the design of the aircraft's wing and fuselage, as these elements represent the most significant generation of drag. The most effective way to reduce drag is to create an airframe with the highest possible lift to drag ratio at the designed weight and cruise speed. Boeing recently unveiled and tested an airframe design made possible by modern materials and construction techniques called the "Blended body wing", which makes use of a smooth transition between the conventional wing and a lifting body fuselage. An independent study of this design by Mahamuni, Kulkarni and Parikh (2014) was able to prove that the design offered large increases in lift to drag ratio when compared to aircraft of similar internal volume. The high volume of the fuselage design combined with the inherent strength benefits of smooth transitions between different structures also significantly reduce the takeoff weight of the aircraft when compared to conventional aircraft of comparable payload. The blended design also provides a much higher stall angle for the fuselage area of the aircraft, providing a highly beneficial effect when transitioning between vertical flight and horizontal flight. Finally, in order to ensure that the aerodynamic stability margin falls within a reasonable range for horizontal flight, the aircraft must be designed according to some basic aircraft principals. These design parameters were found from a Massachusetts Institute of Technology (MIT) course and use the ratio between different elements, including stabilizers and wings, to find the aerodynamic neutral point and stability margin compared to the center of gravity (Basic Aircraft Design Rules).

Testing of a VTOL aircraft represents a significant challenge as three different flight modes, vertical, transition and horizontal, must be tested and ensured as stable, preferably before a test flight. Unfortunately, the transition flight can be exceptionally difficult to test in anything short of a real-world flight without the use of tools such as wind tunnels and large test facilities. In the several decades of early VTOL development,

many test practices were developed that can be applied to current prototypes without incurring unrealistic costs beyond the capability of this study. Many of these testing techniques were outlined in the study of past VTOL designs performed by Anderson (1981). The testing of full-size prototypes was frequently accomplished by using large test fixtures or cranes that would hold the aircraft at altitude while the control systems were tested in hover. Horizontal flight testing was generally accomplished by taking off and landing the aircraft conventionally when possible. Unfortunately, many of these aircraft tested their transition stability by outfitting them with ejection seats and having daring pilots trial the flight mode, often leading to crashes. In a study of the same era, a ducted fan quadcopter VTOL system was developed by Niwa and Sugiura (1987). Testing was accomplished by first limiting the aircraft to 1 degree of freedom in each axis to optimize that control output. The prototype was at a much smaller scale than a full size aircraft and this was accomplished by using cables attached to hard points on the airframe to fix each degree of freedom as desired. Once single degree of freedom tuning was satisfied, the testing proceeded to 3 axis testing by hanging the aircraft on a single cable, so that all axis rotations were free. However, this testing was limited to low throttle testing to ensure that each control axis would blend appropriately. Finally, 4 degree of freedom testing, with free vertical motion, was accomplished with a vertical pole in which bearings allowed the airframe to slide up and down, as well as rotate on each axis, with a limitation of 17 degrees in pitch and roll. Recently, a NASA study to develop an electric VTOL aircraft for high endurance flight, as mentioned previously, used small scale prototypes to test the control schemes of several designs (Fredericks et al., 2013). The main purpose of these prototypes was to test transition control and stability, as it was decided it was less expensive and more accurate to create the models to test the aircraft dynamics than it would be to model the dynamics in simulations. The

prototypes also performed 3 levels of testing in order to develop appropriate dynamic models and to test the implemented control system. Initially, the designs were mounted to 3 axis gimbals to test the force and moment. provided by each thruster to develop the dynamic calculations. The system was then flown vertically in a large (70ft x 35ft x 35ft) netted area to ensure that the system would remain out of FAA controlled airspace during testing, reducing the regulation requirements to test the aircraft. Finally, the aircraft was tethered to a 1400 foot long cable tensioned between two points in order to test the transition phase while both avoiding the risk of a crash and maintaining the non-controlled airspace requirement. Though not all of these testing methodologies can be applied to this study realistically, they can serve as a foundation for testing methodology development.

Concept design

To meet the goal of a simplified, low cost VTOL system, a concept was first developed to define the design requirements and overall system architecture. This allowed for initial power and weight estimates and preliminary off-the-shelf component choices. The analysis done during this concept phase lead directly into the detailed prototype design and final control system architecture. Due to the overall financial and time cost in constructing a prototype, and in order to accommodate continued development and testing of the prototype, it was decided that the aircraft must be designed to be highly modular with easy subsystem access and removable and interchangeable components. Furthermore, to truly use this system as a multi-mission, low cost platform, the main cargo compartment had to be located as close to the center of gravity as possible, which precluded the inclusion of any through-airframe thrust devices at the center of the aircraft. Finally, while a final production model would not require testing hard points, the test prototype would need testing points to be included for control system testing in order to safely anchor the UAS.

One of the most significant requirements was the need to reduce the number of components and complexity compared to other existing VTOL systems so that the final price point and user interface could be as accessible as possible. To achieve this goal, and the requirements listed previously, a dis-similar tri-rotor design was conceptualized, with large wings for lift during horizontal flight. The prototype will use two powerful front motors with large propellers, mounted slightly forward of the center of gravity, to provide most of the vertical lifting force. A much smaller and more efficient rear motor will be responsible for pitch control during vertical flight. All three thrusters are capable of active rotation so that the thrust is always directly downward to maximize efficiency. More so, this rotation is used to translate the motors into a horizontal thrust position to transition

between vertical and horizontal flight. During horizontal flight the more power intensive front motors are shut down in order to maintain efficiency. The tri-rotor system significantly simplifies the overall control system due to the separation of each control axis compared to the more common quadcopter and V-22 “Osprey” types of VTOL aircraft. With this dis-similar tri-rotor system, pitch is controlled through manipulation of the rear thruster throttle, yaw is controlled through differential thrust vectoring of the main front main thrusters, and roll is controlled by differential throttle of the front main thrusters.

This system allows for a conventional fuselage with a center of gravity centric cargo bay so that varying flight loads do not affect overall stability characteristics. Furthermore, the main fuselage design accommodates hard points for testing connected to the main aircraft structural components, and close to the center of gravity. Finally, easy subsystem access and modularity in design is being achieved by using 3D printed components fastened using small screws or nylon break-away bolts and plywood ribbed frame construction. Components using rapid prototyping printing are easy to alter in CAD, reprint and replace due to non-permanent fastening techniques, while the wooden structure is easy to repair or modify if required. By employing this construction layout, a single prototype base can be constructed while important components are still accessible for alteration late into the testing phase.

An important aspect of developing an extremely accessible UAS with VTOL capabilities is to have a user control interface that is both affordable and easy enough to use that an average quadcopter or remote controlled aircraft pilot would be able to control the aircraft with little to no extra training. In order to achieve this effect, it was important to consider the basic interface and requirements during the conceptualization phase of the design. It was chosen to use a standard remote-control aircraft controller as the main user interface in which, during vertical flight, the controls would match those of

a standard quadcopter, while in horizontal flight, the controls would match that of a fixed wing aircraft. More so, while the user inputs will be directly coupled to the control surfaces of the aircraft during horizontal flight, the controls will be decoupled during vertical flight so that the pilot inputs a desired angle, and the control system on the UAS will work to achieve that position. This computer assist is similar to the controls on high end camera quadcopters and will be required to ease the pilot burden during vertical flight and the transition between flight modes.

Equally important to the user control input is the data stream the pilot receives back from the UAS. This was also considered during the conceptualization in order to ensure that all the required communication equipment would be included during the detailed prototype design. In order to achieve a safe, and easy to use data stream, a Visual Basic application is used to receive and all relevant aircraft information as well as provide audio cues for unsafe data states. In order to transmit and receive this information, a pair of xBee transceivers are located on the UAS and at the ground station to connect to the laptop running the data application.

Detailed Prototype Design

The design and testing processes have resulted in 3 generations of the flight system. These systems are referred to as the MKI design, developed during the 2015 - 2016 Capstone Senior Design class, the MKII design, developed with assistance from the 2016 - 2017 Capstone Design Team, and the MKIII design, developed in the 2017 - 2018 timeframe. Described in this section is the detailed design of the final generation, the MKIII design (figures 1 and 2).

Figure 1: CAD Render of MKIII Design

Figure 2: Photo of Fight Ready MKIII Prototype

Aeronautical Design

An aircraft capable of vertical takeoff and horizontal flight must have its airframe optimized for both, in a weight efficient manner. While a vertical takeoff system has few requirements for drag reduction and shape, it must be rigid and able to create the most efficient vertical thrust possible. On the other hand, horizontal flight requires comparatively large wings to develop vertical lift while minimizing forward facing drag. Furthermore, stable horizontal flight requires a tail section for maneuvering as thrust modulation is less efficient and less predictable.

Design of the test prototype started with the most important aspect; the wings. A spreadsheet was set up with the mass of each component required for the aircraft to operate as well as an estimated airframe weight of 4500 grams in order to find the final estimated aircraft weight with payload. The airframe weight estimate was gained through observation of weights of other UAS airframes with similar cargo capacity, as well as experience in developing a similar airframe for the 2015-2016 capstone design project. The result of these calculations can be seen in Table 7 in Appendix A with a final estimated weight of 8.22 kg, or 9.22 kg with payload. A desired wing loading, or aircraft mass divided by wing area, of 30 oz/ft² or 9155 g/m² was chosen as it represents an average wing loading for efficient, long flight endurance fixed wing aircraft. While a low wing loading increases overall drag and top speed, it was deemed acceptable in order to reduce the power required to achieve, and maintain cruise speed. A maximum wingspan of 3.5 meters was also chosen in order to maintain a level of practicality in transportation and use. These assumptions yielded a required wing area of at least 1.001 m² and a wing chord of at least 0.29 meters. A final wing chord of 0.325 was chosen with a 3.2 meter wingspan, giving a total of wing area of 1.040 m². Furthermore, A 4-degree dihedral was applied to the wing in order to angle the thrusters inward for vertical flight

stability. Use of a lifting blended body fuselage was also determined to be an achievable option to increase overall lifting area. A simplified diagram of the final wing dimensions and layout can be seen in figure 3.

Figure 3: Wing Shape Layout

Once the size, payload capacity and aspect ratio of the wing is found, an appropriate airfoil can be chosen. Research into airfoils for small scale, high lift applications yielded a lot of interest in the Eppler E197 low Reynolds number airfoil. The small size and low speed of this prototype results in an estimated Reynolds number of 443,000, assuming air temperature at 20 degrees Celsius and an initial transition speed of 20 m/s, given by equation 1, Appendix A. This Reynolds number resulted in a peak Coefficient of lift (C_L) to Coefficient of Drag (C_D) of 110.7 at 6.5 degrees and a C_L of 1.0247, assuming semi-turbulent conditions based on simulation results run on Xfoil on airfoilTools.com In order to work in a margin of safety before stall conditions, an angle of incidence of 2.5 degrees was chosen as it represents a low C_D of .00791, yielding a C_L of 0.5960 and a C_L/C_D of 75.34. The use of this airfoil yields a final transition speed of 15.67 meters per second, or 35 mph, at maximum payload, found by equation 2 in Appendix A.

The tail on an aircraft is responsible for both stabilizing the aircraft during horizontal flight as well as inducing pitch and yaw with the use of control surfaces. The size and position of the tail stabilizers are both contributing factors to the aircraft's

stability, and are calculated about the vertical axis (yaw) and lateral axis (pitch). In a conventional aircraft, the horizontal stabilizer is responsible for pitch stability and control, and the vertical stabilizer is responsible for yaw stability and control. Generally, the greater the distance between the aerodynamic neutral point and the aerodynamic center of the specific stabilizer, the smaller the stabilizer can be, which helps reduce drag forces while also reducing the maximum maneuverability of the aircraft. This is why bomber and cargo aircraft have long bodies and small tail stabilizers compared to their wings, while fighter jets have short fuselages and relatively large stabilizers. This balance of length over size was an important consideration while trying to maintain aircraft practicality while also minimizing drag. It was chosen to limit the aircraft to approximately 1.5 meters to limit airframe mass, with small design adjustments possible to achieve stability. A significant design choice to save further weight was to use an “V” type tail in which the vertical and horizontal stabilizers are combined in a “V” like shape. This design feature allows the center of the stabilizer to be used as a rear landing point to form a 3-point landing gear system when combined with landing points at the inner trailing edge of each wing. This also enhanced the airflow channel for the rear thruster by removing center positioned obstacles and further increasing the efficiency of the rear motor in horizontal flight. Once these design constraints were placed, the length, chord and angle of the stabilizers could be calculated by using an iterative process with calculations listed in Appendix B until a desirable result was achieved. The final tail dimensions resulting from this process were a 90 degree separation angle at 45 degrees off the horizontal, with a chord of 300 mm and a length of 360 mm. The format of this tail can be better visualized in figure 4.

Figure 4: Tail Shape Configuration

Thruster placement has a significant effect on both vertical and horizontal flight. During vertical flight the thrusters must be in a position to provide ample vertical force while also allowing for a controllable flight system, while in horizontal flight, the thrusters must be molded into the airframe to reduce drag to a minimum. The main front thrusters, which provide approximately 90% of the vertical thrust on takeoff, had to be placed both longitudinally compared to the center of gravity, as well as laterally off of the center of rotation about the roll axis. The distance between the center of gravity and the motors has effects on maneuverability, reliability, power draw and maximum vertical thrust. Due to the torque moment induced by the motors on the pitch axis, which passes through the center of gravity during vertical flight, the further forward the motors were placed, the more torque the rear motor would need to equalize for level flight. Therefore, more

forward mounted motors would have to be matched by a more powerful rear motor, and therefore a larger total vertical thrust vector could be achieved. This orientation also creates a more stable system, as disturbances to the aircraft, such as wind gusts, will be at a much smaller comparative level compared to the thrust moment already being induced by the thruster system. However, this would mean that there was a much larger power draw during vertical takeoff, and the wire gauge would need to be increased to handle the extra loads. Similarly, the Electronic Speed Controllers (ESCs), which are responsible for controlling and powering the motors, get very hot under high load operation, so inducing a heavier load under vertical flight conditions, where cooling airflow is minimum, may create a premature failure condition. A high moment thrust operation means that there is significantly more torque that the rear motor must overcome to induce pitch changes under vertical flight, reducing the overall speed at which the aircraft can maneuver. Conversely, main thruster placement too close to the center of gravity could allow for the aircraft to enter an unrecoverable state if it were to pitch nose down too far. This would be possible as the rear motor is unable to reverse directions, so if a motor off state is not enough to correct a nose down pitch, the thrust moment induced by the front motors would be solely responsible for correcting the pitch which, if too minimal, would result in an uncorrectable state. Highly complex VTOL systems use swash plates and variable pitch propellers in order to solve this problem without the use of a rear motor, however, inclusion of these systems is far too complex and costly to meet the user accessible design goals in the scope of this project. With 1.20 size Rimfire motors chosen as the main lift fans, with 70 newtons of estimated maximum thrust according to manufacturer specifications, these considerations lead to the design decision to place the main thrusters 50mm ahead of the center of gravity. At maximum throttle this would induce a thrust moment of approximately 7 newton-meters which the rear motor will be responsible for compensating for. Positioning the motors

laterally, choosing a distance away from the roll axis, was also an important design decision. While placing the motors far out on the wings, far away from the center of gravity would create a far more stable system, due to greater inertia, it would also require a significant increase in the rigidity and strength of the framework between the motor mounts and the main fuselage. While a stable system is more desirable, the weight penalty was too significant and the thrusters were placed as close as reasonably possible to the main fuselage at 480mm away from the center axis on each side. The added instability with closely mounted roll motors is balanced with more complex and optimized control system software as discussed in the control system design section.

Placement of the rear motor is heavily defined by the position of the front motors and the maximum thrust of the motor. The motor must be capable of generating enough thrust to maintain horizontal flight without the added thrust of the front motors, while also being able to withstand the stress of high frequency throttle modulation during vertical flight in order to maintain pitch control. A Rimfire .46 size motor was chosen, with a maximum thrust output of approximately 31 newtons, or 7 pounds, according to factory specifications, due to the estimated drag of 28 newtons at 27 m/s (60 mph) given the known dimensions of the main wing and tail assemblies. In order to locate the position of the rear motor, thrust output was used to set a distance which would create a direct inverse moment to that created by the front thrusters. While the maximum thrust value was used in the case of the front thrusters, a 75% thrust value was used for calculations pertaining to the rear thruster to create a power overhead for pitch control inputs, yielding an estimated thrust of approximately 23 newtons. Therefore, to cancel out the moment of the front motors, the rear motor would have to be located at a minimum of 304mm away from the center of gravity. A final distance of 340mm was chosen in order to accommodate the blended body airframe shape and provide slightly more overhead for pitch management at high throttle.

Structural Design

The structural design of the system has a significant relationship to the overall mass of the aircraft and must be made to be both rigid and light as possible. While the design styles of current multirotor as well as remotely controlled aircraft of this size were considered as possible solutions, they each held significant disadvantages when applied to this research. Multirotor aircraft have no need for airfoils and often used a highly skeletonized frame structure made of either composite materials or plastic injection molded pieces stemming from a central point. While this creates a highly rigid structure, the front to back non-symmetrical nature of an aircraft reduces the benefits of this design. Similarly, several airfoils are required to produce stability and lift during horizontal flight and thus further parts will be required in order to create the airfoil shapes.

Fixed wing aircraft of this size generally make use of one of three possible construction techniques. Most common is the use of a balsa wood rib design in which ribs provide the shape of the fuselage and airfoil, while balsa runners connect them. Then the shape areas are covered with either a thin plastic film or a thin layer of balsa wood. This provides an easily repairable and modifiable structure. More recent aircraft have begun to use Styrofoam core wings and fuselages which are then covered with thin fiberglass or balsa wood layer to increase durability. Unfortunately, the complexity of the controls required to be within the wings and fuselage, in order to accommodate the VTOL thruster controls, negates the possibility of using this technique on a majority of the aircraft. Finally, the lightest and strongest aircraft use composite wings and fuselage made of fiberglass or carbon fiber. However, the equipment and preparation time required to create such structures are not feasible for a one-off design, in which future modifications may be required, due to both time and economic constraints.

In order to minimize construction time and aircraft mass, while maximizing repairability and modularity, a combination of existing structural designs along with 3D printing was developed. The overall structure can be described as a carbon fiber tube frame with CNC cut ribs and 3D printed complex geometry pieces mounted to it, all covered with a thin balsa wood skin for shape due to the complex shapes in a lifting body airframe. Furthermore, the outer wing sections, which require no control surfaces, are CNC cut closed cell foam, which can be covered with a thin fiberglass layer. This design requires slightly more digital work in CAD software, but the creation of the components is nearly automated, significantly reducing the required man hours needed in order to construct the UAS. The main structure of the fuselage consisted of 3/16" thick plywood runners from the nose to the rear motor mount, with intersecting runners along the rotational tube for the wing. 3/32" plywood ribs attach to the lengthwise runners to provide further structural support and shaping for the fuselage. Every load bearing and load generating subsystem, including the battery mounts, wing mounts, testing hard points and motor mounts, were connected directly to the 3/16" runners. The rotational tube connecting the wings to the fuselage was made up of 0.625" diameter thin wall carbon fiber twill weave tubing and seated in 3D printed bearings installed in the main fuselage runners. The rods connecting the V tail to the fuselage were made up of the same tubing. The inner wings were constructed using 10mm carbon fiber rods to make up the lengthwise rigidity, while 3/32" plywood ribs were used for structural and shaping support. Similarly, the V-tail assembly is a structured rib design with a plywood runner making up the lengthwise rigidity. Finally, the outer wings, which require no control surfaces, were simply CNC cut out of closed cell insulation foam and fiber glassed with 3/4 oz weave. The removable carbon fiber rod used to connect the outer wings to the inner wings also doubles as a structural element of the outer wing to reduce deflection under load. A layout of the top down substructure can be seen in figure 5.

Figure 5: Airframe Structure

3D printers were used to create the further shape forming structures of the aircraft using polylactic acid (PLA) plastic filament. Though Acrylonitrile butadiene styrene (ABS) plastic is 20% lighter than PLA at 1.0g/cm^3 compared to PLA's 1.25g/cm^3 , the significantly increased difficulty of achieving accurate and quality ABS prints using lightweight settings proved the material invalid for this approach. PLA offers the lowest density compared with other 3D printing filaments while still offering acceptable strength and good print qualities. Components requiring little strength, such as the Arduino covers and servo covers, were printed with minimalized filament to reduce weight as much as possible, failure of these parts is both unlikely and non-critical. However, several 3D printed parts which were heavily load bearing and design critical, including the testing points, servo mounts, ball bearings and tail mounts. These parts were highly optimized in mass, design and printing procedure so that the filament strands would only experience tension loads and inter-layer tension loads would be nearly non-existent. Such a technique ensures that the print will not experience common

failure modes in which filament separation occurs, and filament breakage would need to occur for part failure.

Motor nacelle design was a heavy focus during the structural design phase of this aircraft. The nacelles are the assemblies in which both the front and rear thrusters are located and able to rotate to their horizontal flight positions. The first prototype of this system, developed during the 2015-2016 mechanical engineering capstone design class, was unable to fly due to a design failure of the nacelles. The failure stemmed from the mounting placement of the motors, creating a thrust axis that was not in-line with the nacelle rotation axis, resulting in an uncommanded rotation of the nacelle when the throttle was increased. The yaw and forward movement of the UAS is controlled by the angle of the nacelles and thus this motion yielded an impossible to control state. The MKII design prototype featured rotating ducted nacelles located within the wing structure and rear fuselage structure. Though appropriate control was achieved, the aerodynamic and structural mass degradation proved to hinder the aircraft beyond acceptable limits. The current design has the front nacelles blended into the wing structure and rotates the entire wing, connected to the fuselage with a carbon fiber axle. Furthermore, the motors are mounted in line with the axis of rotation to remove any induced torque on the rotation mechanism during throttle modulation. Rear motor rotation is achieved using a hinged motor mount located at the back of the main fuselage, with wing mounted booms connecting the tail to the main aircraft assembly. The angle of the wings is controlled by a linear actuator, while thrust angle manipulation is accomplished with large thrust vector surfaces located at the trailing edge of the airfoil, directly behind the propeller arc. Further details of these controls are discussed in the control system description.

Electronics Design

The electronics and power grid design of the aircraft has gone through several generations to account for system changes and electromagnetic interference issues. What is described here is the current revision of this system and the motivations for the changes made throughout the construction and testing phases. Wiring diagrams of the control box and airframe electronics are located in Appendix C.

Power

The power distribution system on the UAS had to be as lightweight as possible while also handling 6kW of power from a 22.2V, 10Ah battery. In order to accommodate these power needs, 2 pairs of 12 gauge wire are run from the battery location in the nose to the 2 front and 1 rear ESCs located directly behind each motor. The ESCs supply power and control to the thruster motors, while the Battery Eliminator Circuits (BECs) within the ESCs supply 20 amps of 8V and 6V power. The 6V supply is responsible for all 7 of the servos, which it supplies through the control bus in the Pulse Width Modulation (PWM) box. The 8V source connects directly to the Arduino, which requires 7-12V to operate efficiently. Within the Arduino Due, there is an 800mA 5V and 800mA 3.3V regulator. The 5V supply is used to power the 2.4 GHz receiver, PWM to Pulse Position Modulation (PPM) board and lidar range finder, while the 3.3V supply is used to power the remaining sensors, PWM controller and the 900 MHz transceiver.

Sensors

The sensors used in the aircraft were chosen to minimize cost and excess processing in order to achieve high rate control processing, while also providing all data required to control the aircraft and record meaningful flight data. These sensors all connect to a single Arduino Due responsible for final data processing, communication

and overall system control. In order to stabilize the aircraft, fast and accurate orientation data is required. The Adafruit BNO055 chip was chosen for this task for its Inertial Measurement Unit (IMU), which handles the data filtering and processing to return exact orientation data at a rate of up to 100 Hz over an Inter-Integrated Circuit (I²C) or serial interface. Use of IMU chips allows for a slower processing rate on the Arduino compared to the 1-2 kHz required for data collection and filtering for orientation calculation. Control of the motors and servos is handled through an Adafruit I²C PWM board which takes short I²C commands from the Arduino and creates the 1-2 ms PWM signal required by the servos, ESCs and linear actuators. The board also helps to reduce the overall workload of the Arduino, leading to a more efficient and faster control loop. A serial Adafruit Global Positioning System (GPS) module and -2dB antenna were added to track the exact position and ground speed of the UAS during horizontal flight operations. The GPS sensor will also be required for future anti-drift correction during vertical flight. In order to maintain ground station awareness of battery level and possible unsafe power draw situations, two AttoPilot 180 amp, 50V current/voltage sensors were used, with one in each 12 gauge power lane. It was chosen to use two, parallel sensors to add a degree of backup if one were to fail. Since the entire current load of the aircraft passes through these sensors, a failure of a single shunt resistor would result in aircraft loss if only one sensor were used. With a parallel configuration, if one sensor had both shunt resistors fail, the opposing sensor is capable of bearing the entire current load of the aircraft. A summation calculation is used in each control loop to estimate the total mAh usage of the aircraft in order to estimate total battery charge remaining. The use of a Garmin LIDAR Lite V3 allows for system controlled throttle modulation during vertical flight maneuvers. The lidar is capable of measuring the distance to the ground in a 10 to 4000 cm range with a 1 cm resolution and a 3 cm accuracy, communicated to the Arduino over I²C at a rate of up to 100 Hz. As discussed in the control system section, a

PID controller uses this distance data in order to modulate throttle to maintain altitude and provide vertical descent and ascent rate control. Finally, 2 switches were added to the outside of the fuselage to offer the user a direct interface to the Arduino. Switch 1 takes the control system in and out of safe mode as described in the control system section. Switch 2 allows the user to select the wing orientation, vertical flight or horizontal flight, in order to ease storing and moving the aircraft while it is in safe mode; this switch has no function during armed operation of the control system.

In order to allow for control system design and testing while away from the aircraft, and while also providing easy access to the electronic connections, a sensor box design was implemented. This box housed the main input sensors required for aircraft control with a 40 pin connector on one face. This box could then slide into the UAS on a 3D printed rail and plug directly into an opposing connector on the aircraft side in order to route all outgoing and incoming signals and power to the remaining sensors and control interfaces located in fixed locations throughout the aircraft. The equipment located within the box included the Arduino, the 2.4 GHz receiver, the PWM-PPM converter, the level shifter between 5V and 3V signals, the GPS antenna and receiver, and the absolute position sensor. Fixed position sensors include the 900 MHz transceiver, the current sensors, the PWM control board, the lidar range finder, and the two control switches.

Communication

Communication Design

Communication with the pilot was accomplished through 2 separate radio frequency signals. This section will discuss the electronics used to accomplish this communication, while the “Communication Methodology” section will discuss the flow of

user control. For the main pilot-to-UAS control, a commercially available 8 channel JR XP8103 hobby (figure 6) controller was used. While the original design used the stock 72 MHz pulse code modulation (PCM) signal, it was found that this created significant noise in the low voltage power system of the MKII at the same 72 MHz. To remove this power system noise without the extra cost and weight of wire and sensor shielding, a 2.4 GHz module was installed on the transmitter with a matching receiver in the aircraft. The 2.4 GHz module was installed on the transmitter with a matching receiver in the aircraft. The 2.4 GHz module will allow approximately 2 miles of line of sight control before signal transmission failure. The receiver outputs each channel separately as a PWM signal. Each channel is then routed into a PWM to PPM converter in order to convert 8 channels down to a single channel input to the Arduino after going through a level shifter to convert it from 5V to the required 3.3V of the Arduino Due.

Figure 6: JR XP8103 Controller

Two-way ground station communication uses an XBee 900 MHz transceiver, capable of 9600 Baud rate serial communication. On the aircraft, the transceiver is mounted to a breakout board that connects to the Serial 1 lines on the Arduino Due through the 40 pin connector. The transceiver was mounted far aft on the main fuselage in order to negate the interference seen on the analog signal lines during transmission pulses as with the MKII design. At the ground station, a serial to USB converter is used to plug the transceiver directly into a laptop, while a VB.net program is used to interpret, display and record the incoming data stream. The low baud rate transceiver was selected for the 100 mW transmission power and maximum range of 11 miles line of sight and 2000 feet in urban environments. More powerful transceivers exceeded the cost, weight and power limitations of the system in its current state, but may be used in future generations.

Communication Methodology

A main goal of the communication methodology was to develop a robust and user-friendly interface. The communication system had two main aspects, control inputs delivered to the aircraft through the 2.4 GHz controller, and an information stream back to the ground station through the 900 MHz transceiver. The control input from the hand held remote presented several challenges as the control input requirements vary greatly between vertical, transition and horizontal flight. In order to reduce the load on the operator, the control system interprets the input from the remote differently depending on flight mode and flight conditions, rather than providing the user a direct interface to the control surfaces.

During vertical flight, the left joystick vertical motion is responsible for either direct throttle control or commanded decent rate while lateral motion inputs the desired yaw rate. Likewise, the right stick vertical provides a direct thrust vector surface offset and

desired pitch input in order to induce forward and rearward flight. Right stick lateral motion commands the desired roll angle of the aircraft. There are also two dials and two switches located on the top of the remote. The left switch commands the drone to arm or disarm for flight, while the right switch activates or deactivates lidar controlled ascent/descent rate if flight conditions allow. These flight conditions ensure safe control is possible as activating ascent/descent control removes direct throttle control from the user and thus the aircraft must be in vertical flight, while less than 20 meters from the ground with valid distance data. A dial on the right side of the remote provides desired pitch angle to the control system, which is especially important when taking off from uneven surfaces. The flight mode transition is accomplished through the left dial on the remote. As the dial is spun, the control system transitions the wings between vertical flight and horizontal flight. The position of this dial also dictates how the control system responds to each control input through a proportional algorithm to gradually switch control outputs.

Horizontal flight provides the user with a much more common flight control scheme in which the user has a direct control input to each control surface. Like a normal UAS flight system, the left stick provides direct throttle and ruddervator yaw input, while the right stick provides ruddervator pitch input and aileron input. The arm/disarm switch is still active, allowing the user to shut down the entire aircraft should an unsafe situation arise. However, the descent rate switch and vertical flight pitch input dial are deactivated during horizontal flight.

The laptop interface provides two duties, both of which are always available during the start of a flight. Primarily, the aircraft mounted transceiver will send a 63 byte message containing an array of status and variable updates at a rate of 5 Hz. This is a one-way communication in which the VB.net user interface will display and record all data for viewing at a later time. The secondary interface is a two-way communication

path designed specifically for monitoring and rapidly tuning each of the four PID controllers. Burst communications are used to send a limited number of variables at 40 Hz in order to capture every other control update. A different Visual Basic application is used to interpret this data for live plotting and display, as well as data recording. Furthermore, this application is able to send control gain updates back to the Arduino through the transceiver for live control system tuning while the drone is secured in the test fixture. This secondary communication protocol is incredibly important to thoroughly test the disturbance response and control input response of the aircraft while minimizing risk to user and airframe. Desired data structure mode is switched using header messages upon connection from each VB.net program.

The primary data collection system is designed to be used during entire flights to both monitor the aircraft in real time for possible error states and to provide high detail data for post flight analysis for possible UAS improvements. Message frequency is set at 5 Hz to allow the transceiver to execute its built in “handshake” with the base station transceiver. It was found that switching between transmit and receive takes approximately 90-100ms, requiring approximately 180-200ms between each message to transmit without data losses. The exact message layout can be found in table 1. The base station used the incoming data to provide visual and auditory alarms if values fall outside an expected range in order to inform the pilot or copilot that an unsafe state may exist. Furthermore, the ground station provides a visual representation of the aircraft in 3D orientation corresponding to its current orientation so that the user can quickly be made aware of its state during future upgrades in which it may be outside of visual range. This ground station software also provides a quick data viewer feature to allow for rapid diagnostics and value checks after a flight operation. A screenshot of the base station user interface is seen in figure 7.

Table 1: Full Data Message Byte Layout

Byte(s)	Data	Byte(s)	Data
1-2	header	23	Commanded Transition Pitch
3	message type	24	Commanded Landing Mode
4	Error Status	25	Commanded Armed State
5	Flight Mode	26-27	Timestamp (milliseconds)
6	GPS Signal Quality	28-29	Actual Pitch Angle
7	GPS Satellite Count	30-31	Actual Roll Angle
8	Rear Throttle Output	32-33	Actual Yaw Heading
9	Left Throttle Output	34-35	GPS Altitude
10	Right Throttle Output	36-37	GPS Ground Speed
11	Rear Motor Angle Output	38-39	Up-Down Acceleration
12	Left Wing Angle Output	40-41	LIDAR Height Commanded
13	Right Wing Angle Output	42-43	LIDAR Height Actual
14	Left Thrust Vector Output	44-45	Battery Volts Sensor 1
15	Right Thrust Vector Output	46-47	Battery Volts Sensor 2
16	Left Elevon Output	48-49	Current Draw Sensor 1
17	Right Elevon Output	50-51	Current Draw Sensor 2
18	Commanded Throttle	52-53	mHa Used Sensor 1
19	Commanded Pitch	54-55	mHa Used Sensor 2
20	Commanded Roll	56-59	GPS Latitude
21	Commanded Yaw	60-63	GPS Longitude
22	Commanded Throttle Reducer	64	Checksum

Figure 7: Ground Station User Interface

Control system tuning accomplished through the secondary communication protocol is designed to handle all four PID controllers: roll, yaw rate, pitch and ascent/descent rate. However, due to the limited data transmission rate, and quantity of data required, only one control can be monitored and controlled at a time. When the user selects a specific controller, the Arduino will begin streaming that data to the application. The layout of each message can be seen in detail in table 2, but mostly reflects the five major variable inputs and outputs of the control system, such as: input throttle, PID output, output throttle, current angle and commanded angle for pitch control, as an example. When this data is received it is plotted live on the main window and can be recorded upon request from the user. The user can also choose to allow control input from the remote to represent the most realistic control input scenarios, or to override the remote control inputs with stepped inputs to represent worst case scenarios for control system stability. The data must be sent in burst transmissions in order to allow the transceivers to send data in both directions, in which the switch over between transmit and receive takes approximately 90-100ms, requiring 180-200ms between each message in order to transmit without delays or data losses. Each burst transmission represents the 5 variable collection for every other control output for the last 20 loops, resulting in 53 byte messages with required message headers. The application is also capable of receiving data directly from the wired input at a rate of 250k Baud, thus providing more bandwidth and data from each control loop. The user is also capable of updating any of the gains of each controller. This allows for rapid control system tuning as the effects of gain updates can be seen rapidly and fine tuning can be accomplished without a software update uploaded to the Arduino. Once final gain variables are set, they are updated as the default gains in the control code. A screenshot of the PID tuner user interface is seen in figure 8.

Table 2: PID Tuner Data Message Byte Layout

Byte(s)	Roll Tuning Data	Pitch Tuning Data	Yaw Tuning Data	LIDAR Height Control
1-2	Header	Header	Header	Header
3	Message Type	Message Type	Message Type	Message Type
4-13	Roll PID Output (10 samples)	Pitch PID Output (10 samples)	Yaw PID Output (10 samples)	Controller PID output (10 samples)
14-23	Roll Angle Setpoint (10 samples)	Pitch Angle Setpoint (10 samples)	Yaw Rotation Setpoint (10 samples)	Controller Throttle Output (10 samples)
24-33	Left Throttle Output (10 samples)	Rear Throttle Output (10 samples)	Left Vector Output (10 samples)	LIDAR measured Height (10 samples)
34-43	Right Throttle Output (10 samples)	Input Throttle (10 samples)	Right Vector Output (10 samples)	
44-53	Roll Angle Actual (10 samples)	Pitch Angle Actual (10 samples)	Yaw Rotation Actual (10 samples)	Height Setpoint (10 samples)
54-63				

COM3

9600

Disconnect

Show Comm Ports

Status Connected!

DATA Rate 80

Output control

Roll PIDs

Pitch PIDs

Yaw PIDs

Height PIDs

Disable

Disable

Send Commands

Height controls

Tuning Control OFF

Near PID

Kp 0.00

Ki 0.00

Kd 0.00

Current Values

Height (cm)

Setpoint (cm)

Velocity (cm/s)

Setpoint (cm/s)

Commanded vs actual position

Yaw Stability

Tuning Control OFF

Near PID

Kp 0.00

Ki 0.00

Kd 0.00

Switch over speed

Deg/Sec 0

Far PID

Kp 0.00

Ki 0.00

Kd 0.00

Commanded speed

Deg/Sec 0.0

Current Speed

Deg/Sec

Commanded vs actual position

Pitch Stability

Tuning Control OFF

Near PID

Kp 0.00

Ki 0.00

Kd 0.00

Switch over angle

Deg 0

Trig PID

Kp 0.00

Ki 0.00

Kd 0.00

Commanded position

Deg 0.0

Current Position

Deg

Commanded vs actual position

Roll Stability

Tuning Control ACTIVE

Near PID

Kp 0.00

Ki 0.00

Kd 0.00

Switch over angle

Deg 0

Trig PID

Kp 0.00

Ki 0.00

Kd 0.00

Commanded position

Deg 0.0

Current Position

Deg

Commanded vs actual position

Max/Min Roll 20

Max/Min Pitch 30

Max/Min Yaw 100

Max Height 100

Data Logging

Start Logging Data

Save current Gain Values

Load Recorded Gains

Load Data Viewer

Get Current Gains

ESC 1 Temp (C)

ESC 1 Temp (F)

ESC 2 Temp (C)

ESC 2 Temp (F)

Figure 8: Wireless PID Tuner User Interface

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Control System Design

Overall Design

Achieving stable control on the UAS posed one of the greatest difficulties due to the irregular aircraft and flight design. While most quadcopters use high frequency chipsets and hardware to adjust motor power at a rate of 1-2 kHz, the requirement to remove differential throttle control gradually during transition required the use of software controllers with much more limited refresh rates. Each iteration of this UAS has achieved an increase in the control loop timing, with the MKI at 10 Hz, the MKII at 40 Hz and the MKIII, with the replacement of the Arduino Mega with an Arduino Due, at 80 Hz. Though the processor rate of 86 MHz of the Due should allow for faster calculation time, the limiting factor became the BNO55 position IMU, which failed to provide the advertised update rate of 100 Hz.

The slower update rate of the control system combined with the non-linearity of control in multiple axes required a control algorithm more complex than the PID controller used in most quadcopters and other small hobby aircraft. While the initial control system design used stepped gain PID controllers for each axis, in which the gain variables change at a certain error level, it was found that responsive and stable control was difficult to achieve and would vary with battery level. The final design used a continuously variable gain system in which each gain was calculated based on current position, error, and throttle for each loop and fed into the controller. The specifics and equations used for each control axis are described in each related section.

Three different flight modes, controlled by the angle of the motor nacelles, are also used to affect the outputs of the control system during vertical, transition and horizontal flight. Vertical flight requires stability control to be accomplished by the control system entirely using a mix of thrust vectoring and throttle modulation. In this state, the

user input commanded orientations (pitch angle, roll angle, yaw rate and ascent/descent rate) and the control system works to reorient the aircraft to those positions. However, during transition, the airfoils become increasingly effective as airflow rate increases, while differential throttle becoming increasingly ineffective as the wings achieve lift and the angle of the motors compared to the horizontal decreases. To combat this effect, the input from the user feeds directly into the control surfaces, while the controller output effects are proportionally reduced to zero, at which point the aircraft is in horizontal flight mode with the wings and motors pointed directly forward. Finally, horizontal flight mode passes through the inputs from the user directly to the control surfaces, resulting in control familiar to most aircraft pilots. A chart of the entire control system data flow can be seen in figure 9 with each controller broken down in the following subsections.

Figure 9: Total Controller Data Flow Chart

Vertical Pitch Control

Pitch control during vertical flight is accomplished by modulating the throttle of the rear motor. More throttle input will lift the back of the aircraft and create a negative pitch angle, and vice versa for a positive pitch. In order to calculate the proper throttle setting, the control system calculates a percentage of the throttle input from the pilot input and uses this as the baseline throttle to modulate from. In its current revision, the rear throttle factor is 70% of the throttle input, which was found through observation of the difference required to maintain level flight. This moving baseline was implemented so that rapid changes in throttle input would affect both the main thrusters and the rear thruster. Otherwise, the pitch would be affected by throttle changes. The output from the PID controller is then added to this input to increase or decrease the thrust output. Finally, both the throttle offset and the PID input is multiplied by a variable that proportionally goes to 0 as the motor nacelles approach the horizontal position in order to switch the throttle output from a control system output into a direct input from the user. The algorithm can be seen here, with all pitch control constants and set variables listed in table 3.

$$\text{Pitch_ESC_Output} = \text{PPM_Throttle_Input_Rear} - ((\text{PPM_Throttle_Input_Rear} - \text{REAR_ESCMIN}) * ((\text{REAR_THRUST_MULTIPLIER} * \text{PID_Multiplier}) / 100)) + ((\text{Pitch_PID_Output} * \text{PID_Multiplier}) / 100)$$

The PID controller uses the error value to increase the Proportional (P) and Integral (I) gains sinusoidally as the error grows. Increasing the gains will increase the responsiveness of the system, while reduced gains increases the overall stability, especially during disturbance inputs. The calculation for each gain is formulated so that a “base gain” represents the gain output when there is a 0 error state. A “Trig gain” is then multiplied by the sine of the angle error, constrained between -30 and 30, and added to the base gain. The result is that the trig gain goes to 0 at an error of 0 degrees

Vertical Roll Control

Vertical flight roll authority is achieved with differential throttle between the main front thrusters to achieve the desired control input angle from the operator. The differential is calculated with a PID controller and equally added and subtracted from the current throttle setting in order to maintain a similar overall thrust output during mid-level throttle settings. This controller input is proportionally reduced to 0 as the wings reach the horizontal position, as differential throttle at this point would induce yaw rather than roll. The throttle output is also constrained to the maximum output of the ESC in such a way that any overshoot beyond a specified range is subtracted from the other motor in order to maintain a constant differential thrust during high throttle applications.

The PID controller for roll operation uses continuously variable gains for P, I and D. The PID controller uses the error value to increase the gains sinusoidally as the error grows. Increasing the gains will increase the responsiveness of the system, while reduced gains increases the overall stability, especially during disturbance inputs. The calculation for each gain is formulated so that a “base gain” represents the gain output when there is a 0 error state. A “Trig gain” is then multiplied by the sin of the angle error, constrained between -30 and 30, and added to the base gain. The result is that the trig gain goes to 0 at an error of 0 degrees and all that is left is the base gain. Sinusoidal equations were used to offset the sinusoidal non-linearity of the control scheme. The gains are also reduced proportionally as throttle is increased to counter instability seen at high throttles during testing. This instability is likely induced by the firmware on the off-the-shelf motor controllers, which respond more aggressively to throttle inputs at higher throttles than lower throttles. The result is a higher differential of thrust at higher throttle for the same PID controller output. The final equations for gain manipulation and controller output reduction can be seen here, with all roll control constants and set

variables listed in table 4 and a flow chart of the control system can be found in figure

11.

$$\text{Trig_Roll_Kp} = \text{Trig_Roll_base_Kp} + ((\text{Roll_Kp_Near} - (\text{Throttle_Gain_multiplier} * (\text{Roll_Kp_Near} - \text{ROLL_KP_TRIG_HIGH}))) * \text{fabs}(\sin(0.0175 * \text{Trig_Roll_delta} * \text{Trig_Roll_angle}))) - (\text{Throttle_Gain_multiplier} * (\text{Trig_Roll_base_Kp} - \text{ROLL_KP_BASE_HIGH}));$$

$$\text{Trig_Roll_Ki} = \text{Trig_Roll_Base_Ki} + ((\text{Roll_Ki_Near} - ((\text{Throttle_Gain_multiplier} * (\text{Roll_Ki_Near} - \text{ROLL_KI_TRIG_HIGH}))) * \text{fabs}(\sin(0.0175 * \text{Trig_Roll_delta} * \text{Trig_Roll_angle}))) - (\text{Throttle_Gain_multiplier} * (\text{Trig_Roll_Base_Ki} - \text{ROLL_KI_BASE_HIGH}));$$

$$\text{Trig_Roll_Kd} = \text{Roll_Kd_Near} - (\text{Throttle_Gain_multiplier} * (\text{Roll_Kd_Near} - \text{ROLL_KD_BASE_HIGH}));$$

$$\text{Roll_Left_ESC_Output} = \text{PPM_Throttle_Input} + ((\text{Roll_PID_Output} * \text{PID_Multiplier}) / 100);$$

$$\text{Roll_Right_ESC_Output} = \text{PPM_Throttle_Input} - ((\text{Roll_PID_Output} * \text{PID_Multiplier}) / 100);$$

Table 4: Final Roll Controller Gains

Roll Base Kp	Roll Kp Near	Roll Kp Base High	Roll Kp Trig High	Roll Base Ki	Roll Ki Near	Roll Ki Base High	Roll Ki Trig High	Roll Kd Near	Roll Kp Base High	Trig Roll Delta
0.3	4	0.1	0.9	2.5	5	0.8	1.4	1	0.65	1

Figure 11: Roll Controller Flow Chart

Vertical Yaw Control

Yaw control was a very important system on the aircraft due to the dissimilar tri-rotor design, which induced a rotational moment during normal operation due to an odd number of motors applying torque to the airframe. This effect was further induced by the lack of counter rotating propellers available on the market, resulting in all motors applying torque in the same direction. In order to counteract this, and give the operator proper yaw control, a pair of thrust vectoring surfaces were built into the trailing edge of the wing directly behind the front propellers. The placement was chosen so that as thrust increased, and therefore the induced torque on the aircraft increased, the counter force supplied by the vector surfaces would also increase. Yaw control was achieved by moving the surfaces in opposite directions based on the output of a PID controller. This control loop would work to achieve the yaw rate commanded by the operator. Unlike the pitch and roll axis, this control represented a more linear system, not requiring the use of continuously variable control gains to produce a quick acting, highly stable control output, instead requiring a single step gain change at yaw rate error exceeding 10 degrees per second. The final gains used can be found in table 5 with a flow chart of the control system in figure 12.

Table 5: Final Yaw Controller Gains

Yaw Kp Near	Yaw Kp Far	Yaw Ki Near	Yaw Ki Far	Yaw Kd Near	Yaw Kd Far
15	20	6	5	0.1	0.1

Figure 12: Yaw Controller Flow Chart

Vertical Ascent/Descent Control

When vertical ascent/descent control is activated by the operator, it replaces the function of the throttle stick with a vertical rate control function. If the user positions the stick in the middle position, the control system will work to achieve a hover, with no vertical movement. The stick pushed upwards yields an exponential input to the control system to increase the commanded height, in which further stick motion results in a greater rate of increase in height. Moving the stick to the downward position results in a reduction in altitude at a desired rate. The maximum rate of descent is also controlled by the measured distance of the ground, so that only a slow descent is possible to command when the altitude is less than 5 meters. This control is switched on and off by the operator with the landing gear toggle on the transmitter and can be switched on when the aircraft is in vertical flight mode and it detects that it is receiving valid sensor data. Addition of this control was introduced during tethered flight testing when a difficulty in maintaining a fixed altitude, or smooth descent rate, was witnessed. The operator will have the option to both takeoff and land the aircraft in this mode, significantly reducing the workload of the user.

The input sensor data for this control is a Lidar Lite V3 by Garmin. This sensor provides distance measurements up to 40m away with 1cm precision. The incoming distance data is run through a weighted average filter in which each new sample is worth 20% of the average. The output from this filter is then multiplied by the cosine of the pitch and roll angles to account for error induced by the angle of the aircraft creating a longer path to the ground. The control system uses this output from this equation as the aircraft's current altitude. A PID controller then uses this output as the data input. The PID controller output is added to the baseline throttle in order to achieve the commanded altitude. The baseline throttle is simply the last commanded throttle position before the user switched to ascent/descent control input. It is noted that, while this control eases the workload for the operator, it removes their direct control over throttle input and, thus, removes one layer of safety from operation. The averaging filters can be seen here, with control loop variables listed in table 6 with a flow chart representation of the system in figure 13.

$LIDAR_Old = ((4 * LIDAR_Old) + LIDAR_In) / 5;$
 $Lidar_Calc_Height_Old = Lidar_Calc_Height;$
 $Lidar_Calc_Height = LIDAR_Old * \cos(0.0175 * Roll_Current_Angle) * \cos(0.0175 * Pitch_Current_Angle);$

Table 6: Final Descent Controller Gains

Height Kp	Height Ki	Height Kd
1	2.5	0.5

Figure 13: Descent Controller Flow Chart

Vertical Forward-Back Control

Aircraft movement either forward or backward during vertical flight is accomplished through a combination of pitch control and thrust vector planes. Unlike other vertical flight controls, this system represents a direct input-output configuration in which input from the operator translate to outputs from the controller without the inclusion of a separate control system in-between. For small user inputs, of less than $\frac{1}{3}$ of maximum, the thrust vector surfaces directly behind the propellers angle forward or backward together to vector thrust in either direction. Vectoring thrust backwards results in a forward motion of the aircraft. The angle outputs are then fed into the yaw controller in order to output an angle differential between the surfaces for continued yaw control. It

was witnessed during testing that using only thrust vectoring did not yield the control authority desired and required for flight during windy conditions, so user input beyond 33% also modifies the desired pitch angle. A negative pitch angle results in all 3 motors outputting a rearward thrust component, which results in forward motion, and vice versa for a positive pitch angle. This desired pitch angle is fed into the pitch controller to output the required rear motor throttle to achieve the desired effect. During the transition to horizontal phase this control input changes to only output to the rear control surfaces to control the aircraft's pitch.

Transition Design

Transitioning between vertical and horizontal flight represents many challenges in terms of control system authority and stability. The greatest difficulty of control design was that, unlike vertical flight, which went through months of significant testing and tuning, or horizontal flight, which has well established equations and design elements to ensure stable flight, the transition phase is both unknown and untestable prior to full flight conditions. To ensure the greatest chance of success during flight testing, the vertical flight control system was tested for significant disturbance input stability, so that it would represent a safe state the operator could return to during an abort of transition to horizontal.

Transitioning to horizontal flight and transitioning back from horizontal flight to vertical represent two different problems. A transition to horizontal flight requires the aircraft to gain velocity by angling all three thrusters to provide a forward force vector. The greater the angle of the motors, the more forward thrust the aircraft is generating and the faster it will go. However, with total vertical thrust reduced, the throttle setting will need to be higher to maintain altitude until the wings can provide enough vertical lift. Furthermore, the more the motors are angled off the vertical orientation, the differential

throttle used to maintain roll stability will also begin to induce a yaw effect, and the thrust vectoring used to control yaw, will begin to induce a roll moment. To counter these effects, there is an equation to proportionally reduce the effects of the PID controllers until being completely disabled at motor angle of 60 degrees off the vertical.

$$\begin{aligned} \text{PID_Multiplier} &= (100 * (\text{PID_STOP_ANGLE} - \text{PPM_Motor_Pitch_Input}) / \\ &\quad \text{PID_STOP_ANGLE}); \\ \text{Control_Output} &= ((\text{PID_Output} * \text{PID_Multiplier}) / 100); \end{aligned}$$

More so, at 30 degrees off the vertical, the thrust vector surfaces switch over to direct roll control outputs, in which a roll input from the user will result in a proportional deflection of the vector surfaces. Transition to horizontal will have to be carefully and slowly executed by the pilot as rapid motor transition to the horizontal position may result in a time period in which controllability of pitch, roll and yaw using thrust modulation and vectoring will be lost while not yet having achieved enough airspeed to provide adequate lift or the required airflow over the control surfaces to provide authoritative control. The pilot will need to move the motor angle to approximately 30 degrees off the vertical and once the plane achieves enough speed, at an estimated 35 mph, will continue the rotation to the final horizontal position.

Transition from horizontal to vertical flight results in the opposite set of problems. With the high velocity the aircraft will have when entering the transition, there is a risk that increasing the angle of the motors, and therefore the wing, will result in significant angle of attack stall turbulence, which could result in a loss of control. Fortunately, the higher initial speed means that there will likely not be the phase of reduced control authority that may occur during transition to horizontal, due to the increased airflow over all control surfaces. It was decided to use the same PID control proportion and start points as used during horizontal transition so that the flight performance will be as predictable as possible to the pilot. The high stability of the vertical control system

should allow the pilot to transition to vertical flight at any rate, but to minimize the effect of high-speed stall turbulence, the transition should be executed over a few seconds, allowing the aircraft to reduce speed from increased drag before shifting the wings to a stall condition.

Horizontal Control

Horizontal flight represents the simplest of the control systems. Unlike vertical flight, where a computerized control system is required to keep the aircraft stable, the airframe was designed to be inherently stable during horizontal flight. As such, each input from the user translates directly to a control surface deflection on the tail and wing. The surfaces used as thrust vectors during vertical flight are used as inboard ailerons for horizontal flight. Pitch and yaw control come from the V-tail control surfaces at the back of the aircraft. These surfaces are mixed together using a simple equation so that there is no parent control that can override the other. This means that if each control is set to their maximum positions, what will result is one surface in its maximum position and one surface at its center position; providing both pitch and yaw moment. The tail controls remain active during all flight modes, while the aileron controls only activate during the transition to horizontal flight.

Safe Mode and Disarmed State

In order to provide added levels of safety, with an aircraft capable of consuming 6 kilowatts of power under maximum load, two disarming functions were added to the code. A top-level control is available to the pilot at any time by using the disarm/arm switch on the remote controller. This switch sets all the outputs to a known safe state with the motors completely powered off, while still reading all sensor data and maintaining communication with the ground station. This option is meant to be used in

order for pre-flight checks just prior to take off and to allow an operator to approach the aircraft safely after landing.

A secondary safety is on the front of the aircraft just in front of the carrying handle. The status of this switch is checked before the control system completes the setup operation and at the beginning of each control loop. In order for the control system to exit setup and enter the active control loop the switch must be switched from “Disarmed” to “Armed” in such a way that if the switch was initially in the “Armed” position, it must first be switched to “Disarmed” and back to “Armed” before the system will execute any commands. If the switch is set to “Disarmed” after the control system has already entered the main loop, all code execution will cease other than what is required to set the control outputs to the known safe states. This includes stopping all communication with the ground station and all sensor data collection. This switch is intended to be used when physically handling the aircraft after the power system has been connected to ensure that the chance of accidental motor activation is near zero. These two safeties together give both the pilot and the ground crew the direct control required to minimize any risk of injury.

Future Capabilities

One of the biggest motivations for this design of the control system was to allow for future expanded capability. Stabilization for the UAS required a fixed rate controller to manage various control loops in known intervals; a use best suited for a microcontroller like the Arduino Due running commands at a low level. However, more advanced features such as autopilot, waypoint navigation, automated transition, etc., require more complex sensor fusion, floating point operations, vector math and the overall greater processing power seen in a full computer, such as a raspberry pi. While these calculations are important, they do not require the same rigid time factor and can be

communicated to the main controller when ready. A future system may use a combination of a microcontroller and a more powerful microprocessor to handle stabilization and sensor data collection separately from flight path planning, automation and ground communication. The control system was designed specifically so that it could receive the desired position data the user supplies, but from an onboard computer instead, then execute those commands with the same system currently in use. The computer would use current position, orientation, flight speed and other added sensor data to compute the required motions to achieve the desired flight path, whether generated by the user, or generated by the control code. While this level of automation is far outside the scope of this project, it was decided it was important to integrate these functionalities into the base level programming to leave a clear path for future efforts with this UAS.

System Testing and Results

To ensure the safety and successful operation of the UAS, control system and mechanical design elements were tested at each stage of development. Satisfactory results from each of these tests were required in order to continue to the next phase of development. While there have been three generations of the flight system, much of the testing was designed to be universally applicable to each iteration of the design.

General design testing

At the onset of this project, the Dissimilar Tri-rotor transitioning aircraft design was non-existent in any examples found by the capstone team. Since that time, a few similar examples have appeared on the market horizon, but lack data required to use as design reference. As such, significant testing was done in order to confirm that the flight system would be controllable and overall possible. Unfortunately, due to the simplicity of the test fixtures, there was little data to be collected during the initial control system checks until the first full aircraft prototypes were developed. Rather, these fixtures were used to confirm control system code elements with visual confirmation of the expected results.

The most significant concern for control system stability was the use of a small rear motor for pitch control during dynamic flight events at the slow update rate achievable with an Arduino microcontroller. In order to test the feasibility of this design element, a simple test fixture was created with a much smaller motor mounted at the end of a hard pendulum arm, so that a throttle input would result in the pendulum swinging. An Arduino mega and a BNO55 absolute position sensor were used to control the motor ESC and to sense the current angle of the system. The control code implemented a PID loop running at 10 Hz to modulate the motor throttle in order to maintain 0 degrees

angle, or parallel to the ground. Figure 14 depicts a solid model of the test fixture. Once the PID was tuned to achieve the desired angle, the control axis was moved up and down rapidly and the pendulum jounced in order to ensure system stability. Initial testing quickly resulted in success in which proper orientation was achieved rapidly, and maintained with only minor oscillation. Though a more stable and refined PID control loop with less oscillation and faster response may have been possible at this stage, the desired confidence in design implementation was achieved and the design moved forward, making use of the components in the next test fixture.

Figure 14: Pitch Control Test Fixture

Roll axis control was the next control element to be tested. The system for controlling roll is the most equivalent to existing control systems for quadcopters, in which differential throttle is used to induce a roll motion. Unfortunately, while multiple test fixtures were prototyped, a solution to accurately represent the flight dynamics was not found. While systems using a pivoting center axle would seem like the simple solution, the reality is that tendency to fall to one side under low power is both unrealistic and

near impossible to tune safely at lower throttle settings. During regular flight, the low center of gravity compared to the lifting areas will result in a pendulum effect with slight self righting characteristics, the opposite of which was possible with a test fixture.

In order to accurately represent the roll axis, a 3-motor free-flight system was developed that would also allow for the testing of the yaw axis and pitch axis in a single system. The yaw axis control would be achieved by using thrust vectoring techniques on the front main lifting motors. This control layout disconnects the control of each axis from each other, overall simplifying the control system required. Prior to testing, the team had concerns that the unequal induced yaw torque by the 3 motor layout would result in yaw control complications, resulting in a test fixture focused on ensuring a realistic flight model. The test fixture was designed as a ridged “T” shape with the center of gravity and motor positions set close to the expected ratios compared to the early design of the full size prototype. A model of this design can be seen in figure 15. Dissimilar motors were also used between the front and back to apply the unequal, and unpredictable torque. Finally, servos were used to rotate the main motors along the pitch axis in order to ensure yaw control could be achieved with thrust vectoring. An Arduino Mega and BNO55 position sensor were maintained as the control elements in the system. The ability for the control system to read inputs from a wireless user-operated handheld controller was also added in this fixture. This control input scheme is the same system used throughout the remainder of the UAS iterations. Due to the lack of a USB tether or any other form of communication, as well as the lack of on-board data recording capabilities, it was once again impossible to record data of the testing done with this test fixture. While it was not expected for the fixture to hover more than a few inches off the ground, the test was considered a success if complete user input yaw control was achieved with stable pitch and yaw during hover in ground effect. Though several challenges were faced in achieving a stable 10 Hz control update rate, it was eventually

accomplished and testing resulted in success. The quick modulation available with thrust vectoring, compared to throttle manipulation, resulted in a very stable system with fast response to yaw commands from the pilot. Furthermore, pitch stability remained strong through user inputs of yaw, throttle and forward-back control accomplished by vectoring throttle in the same direction. Roll authority continued to be somewhat problematic, in which the high center of gravity caused by the battery and Arduino mount meant that the motors would saturate the PID before they could recover from a disturbance. This resulted in a skip off the ground, in its ground effect flight, and a bounce to the other side. Initially this action would escalate until the throttle had to be reduced, but some tuning of the PID loops, and the addition of gain stepping, resulted in stable enough response to allow the team to move forward with full size prototype design and construction.

Figure 15: Three Motor Control Test Fixture

Small scale winged model

During construction of the full size prototype, it was decided by the design team to develop a $\frac{1}{4}$ scale model of the UAS with all functioning controls and sensors on board. This would result in the ability for full system testing in a small scale, durable chassis, reducing the risk for system failure during the initial operational testing of the much larger, more fragile prototype. This scale test also incorporated the controls

required for horizontal flight to allow for control system development and validation, as well as simple transition tests. Due to the mass of the $\frac{1}{4}$ scale model compared the available motors, it was unexpected that this system would be able to hover out of ground effect or achieve horizontal flight, but only a few inches of hover would be required to validate the UAS. While the transceiver would be added to the test model at a later time to test the data acquisition software on the computer, it was decided to be left out for initial control testing in order to reduce the risk of damage to long lead time parts. As such, the only data available for the test system is in the form of video recordings and pictures. Along with maintaining the Arduino Mega, the BNO55 and the input devices for the hand held remote, this system saw the addition of the altitude sensor, the GPS, the transceiver at a later date, and 4 micro servos to control the ailerons and ruddervators. A model of this test fixture can be seen in figure 16.

Figure 16: MKI Quarter Scale Test System Render

Due to the success of the 3 motor test fixture, success of this test only required that previous control authority was maintained, roll stability increased to a level in which ground strikes no longer occurred, and all user inputs resulted in the expected outputs. Also required was the implementation of transition controls in which different control

loops and outputs are required for user inputs. The development of this control system ran parallel to the construction of the full size prototype, so code updates were required regularly as details were finalized in the design. Control system optimizations were implemented in order to increase update rate to 15 Hz to achieve more stable control. This change, along with a more realistic flight dynamics and further PID tuning resulted in a ground effect hover stable in all axes. Furthermore, all control outputs were mapped as required. The code layout for mapping developed in this testing remains the same layout implemented in the current revision of the UAS. While it was not expected that the test model would be able to achieve flight during testing of transition controls, due to the excessively high wing loading, during a high throttle test the drone surprisingly lifted off the ground and took flight. Unfortunately, this testing was being done inside a warehouse building, so throttle was immediately cut to avoid a wall and the resulting fall caused structural failure of several components. While this ended the use of the test model for flight testing, the transition to flight in a non-optimized design was a very promising outcome towards the success of the overall design. It was at this point that the transceiver was added to the assembly in order to begin testing of data collection. Due to the time constraints of the capstone design class, full data collection was not realized before the completion of the MKI prototype. This capability was expanded much further during the design and testing phases of the MKII Prototype. A picture of the scale test system prior to flight testing is available in figure 17.

Figure 17: MKI Quarter Scale Test System Completed

MKI design and construction

The MKI Prototype design tested many unknown factors in both its design and construction. While many of its design elements were tested using test fixtures and scale models, its overall construction of modular, 3D printed sections could only be tested in small sections previous to final construction. Furthermore, there were a few design elements that worked well during small scale testing, but failed to scale well to the full size prototype. While the MKI overall failed to accomplish the goals set for it, many lessons were learned and applied to the next generations drone.

The MKI design implemented a 3D printed exoskeleton for the fuselage, in order to maximize cargo volume capacity, and sheeted rib construction for the wing and tail elements. The motivation for using 3D printing was to achieve simplified, fast construction, precise dimension control and structural durability due to the plastic nature of the material. While these goals were accomplished, it was not feasible to print the

structure as light as expected. The result was an assembly that was far heavier than was required or feasible for the given thrust output of the motors. In an attempt to reduce the mass of the system the durability and rigidity of the system was sacrificed, requiring post-construction reinforcement of the structural design with carbon fiber tubes. Finally, the initial design included features to allow for disassembly of the fuselage and wings to ease transportation. However, wire routing and structural rigidity requirements made disassembly difficult and impractical for most situations.

Several design elements tested in small scale models did not translate to the larger size with the same reliability. Most notably, the design of the wingtip motor nacelles created several significant issues. Foremost was the off axis thrust line, which oriented the rotation point of the nacelle above the centerline axis of the motor. Under small scale testing, this created the desired effect of moving the thrust line ahead of the center of gravity during vertical flight. However, the induced torque load on the rotation pin, and therefore the servo controlling it, was too large for the servo to maintain location and move accurately. Due to the necessity for accurate and rapid motor angle control, this alone negated any possibility of flight testing the first generation prototype. The MKI continued to be ground tested in order to highlight any other design modifications required in a second generation design, and several other design issues were noted. With efficiency as a key element of the project goals, it was noted that the direct propeller wash onto the main wing, with approximately 25% of the propeller arc directly over the wing in vertical flight, would reduce overall thrust and therefore increase energy use. More so, while having the main thrusters at the wingtips increased the stability of the aircraft in vertical flight, the required structural reinforcement to support the heavy motors increased the overall weight of the aircraft significantly. Each of these design failures were analyzed and solved in the conceptual design of the second generation

airframe. A model of and a picture of the final MKI design can be seen in figures 18 and 19 respectively.

Figure 18: MKI Full Model CAD Render

Figure 19: MKI Full Assembly Photograph

MKII design and construction

With the lessons learned from the first generation UAS, the MKII design saw a much greater level of success, reaching the first stages of free flight testing. While still not completely successful, the MKII provided a platform for control system, ground station and data recording code to be completed. Likewise, more structural design techniques were tested in this design, resulting in a weight improvement and significantly

faster construction time. Finally, each of the design failures seen in the MK1 design were solved, while later stage testing revealed other design elements to be reconsidered during the MKIII conceptual design phase.

In order to address issues seen in the MKI system, several significant changes were made to the construction and basic design of the aircraft. A focus on rigidity was implemented with a carbon fiber subframe, running the length of the drone and across the inner wings. This subframe was surrounded by printed rib and spar construction to create the fuselage shape of the aircraft in an effort to reduce weight. While the final aircraft design was heavier than desired at 22-24 lbs, it remained a significant improvement over the MKI design. The disassembly point on the rear fuselage was also removed to reduce the weight and complexity required to make it structurally sound. Likewise, the wings were split into inner and outer sections at the point in which the complex mechanical systems for nacelle control ended, with the outer sections removable. Resulting was a much more manageable and practical wing disassembly than seen in the MKI design, with only 1 servo wire running into each outer wing section. Unfortunately, the aircraft testing ended with a failure of the forward fuselage to aft fuselage interconnect. As this is the location in which the forward carbon fiber subframe and the aft carbon fiber subframe meet, it represents the weakest part of the airframe. The reason for this failure is still unknown, but is likely the result of a non-mission related force applied on this weaker area.

The wing split could be accomplished due to the inboard mounted motors affixed as close to the fuselage as the 15" propeller diameter would allow. These motors were mounted within in-wing ducts, which pivoted on an axis directly intersecting the motor centerline axis. Mounting the propellers in ducts ensured that all propeller wash would have a "clean" exit towards the ground, maximizing thrust efficiency. The on-axis rotation point significantly reduced the load on the control servo and ensured that an increase in

throttle would not induce an excessive load on the servo. These changes to the front propeller layout resulted in an increased thrust and faster thrust vectoring control, represented in the data by precise and quick yaw control.

The rear motor mount was also modified to increase efficiency and control authority. Rather than a fixed, vertical facing mount at the far back of the aircraft, the motor was moved up to 420mm back from the center of gravity and was built into a nacelle similar to the front motor mounts, allowing it to rotate to the horizontal position. This rotation results in the possibility of front motor shutdown during horizontal flight, with the much lower power draw rear motor providing required thrust to maintain speed. Though this possibility was not realized during MKII testing, it is an element carried forward into the MKIII generation. A CAD model render and a final picture of the MKII design can be seen in figure 20 and 21 respectively.

Figure 20: MKII Full Model CAD Render

Figure 21: MKII Vertical Flight Test Ready Photograph

From a control systems perspective, many elements were optimized and improved throughout the development of the MKII drone. While these are discussed in the control systems section, a system design level change to a control update rate of 40 Hz, rather than the 15Hz the MKI ran at, presented the most significant improvement. During control system testing, it became clear that this change was required for successful control, and a further increase in refresh rate may be beneficial for stability.

In order to verify the stability of the control system before risking the aircraft in free flight, a 3 dimensional test rig was developed with help from the 2017 Mechanical Engineering Capstone team. This test rig allowed for freedom of movement in one axis at a time while anchoring the aircraft to the ground. The disconnected layout of the controls during vertical flight meant that each control could be tuned completely independently of the others. Once each axis was validated, the aircraft was to be tested in a tethered free flight fashion to further ensure that all systems remain stable without the influence of the test fixture. Ensuring the test rig would replicate real-world dynamics

required the axis of motions to be as close to the rotation point for pitch, roll and yaw as possible, which was accomplished by using 3D printed mounts that secured directly to the carbon fiber subframe along the centerline of the fuselage. The pitch axis was oriented ahead of the center of gravity, just aft of the front motor thrust line. This was required to replicate the moment about the CG that would be applied by gravity during vertical flight, which would not manifest during anchored tests if the aircraft were affixed at the center of gravity. Each axis would be analyzed based on four criteria and be considered satisfactory only if all were within acceptable tolerances. First of these criteria was the ability to respond to a commanded step response of 10 degrees in under 1 second with minimal oscillation. Overshoots of less than 5 degrees were considered acceptable if saturation of the control PID loop was required to achieve rapid response, as was the case with the pitch stabilization. Similarly, the second criteria required that each control axis must be able to maintain various angles, or turn rates, other than level flight. For pitch and roll this included testing -10, -5, 5, and 10 degrees. Thirdly, the aircraft must respond quickly, and without oscillation, to disturbances, such as what may be experienced during windy and turbulent conditions under normal flight. Testing for this criterion simply required pushing the aircraft in each direction against the axis being tested using varying force, speed and longevity and ensuring the response is as expected. Finally, the hand held remote was used as the input device to apply a proportional and varying commanded position; a much better representation of real world conditions. The aircraft must respond quickly and reliably to the commanded inputs to pass this test. Though the step response is generally the worst case scenario when testing a control system, there was concern it would be possible to induce an oscillation into the control system during proportional control, so this was an important test.

A laptop driven test interface was developed alongside the test fixture to provide rapid tuning and recording capabilities. This program provides a tool for the user to run the entire control loop on the drone, while having access to modify the control gains in real time, allowing for rapid control system tweaks wirelessly without the need to reflash the Arduino code between each run. The user can also command specific positions from the interface in order to apply the required step responses for testing. Furthermore, this program would notify the UAS it is in use, and lock out the output control to axes that were hard mounted, ensuring the aircraft would not be fighting against the test fixture. Once the user selects the appropriate axis, the Arduino system sends burst transmissions, every 200ms, with data about that axis from every control update comprising of PID output, required servo or throttle outputs, commanded position and actual position. This data is then recorded and displayed live on the laptop screen so that the user can see overshoots and oscillations that may not be clear while looking at the drone directly. This recorded data allows for confirmation that the UAS is passing each of the required criteria and to analyze the data afterward to look for possible overdamped or underdamped situations.

During this control system testing, a continuously variable gain equation was developed in order to address the non-linearity of polar control system requirements. Initially, gain scheduling techniques were used to counter the non-linearity. However, though successful for pitch and yaw, these had little success during roll testing. It was found that it was very difficult to achieve a fast response at mid band throttle without losing stability at higher throttle, due to the faster motor response times at high throttle. The variable gains were based upon polar angles and throttle settings; further discussion of which can be seen in the control system section. Initial testing of the gain algorithms was done along the pitch axis so that the results could be compared directly to a known stable system using gain scheduling. The results were immediately improved with minor

tuning and can be seen in figure 22 vs figure 23. This system was then applied to the problematic roll axis, resulting in a satisfactory state after some tuning, seen in figure 24. It was chosen to leave the Yaw axis as a gain scheduled system as it already showed highly precise response and more closely resembled a linear system.

Figure 22: Pitch Control with Stepped Gain Mapping

Figure 23: Pitch Control with Continuously Variable Gains

Figure 24: Roll Control with Continuously Variable Gains

With all stability requirements met on each axis, testing moved to tethered free flight. This was accomplished with a test assistant as a hands-on tether using an elongated handle. This allowed the assistant maintain positive control of the aircraft if anything went wrong, while allowing it to fly “hands-off” if it were stable enough. During the initial test, however, the sustained current load caused a wire to fail and the test needed to be ceased immediately before relevant data could be recorded. While a repair was accomplished with double power line runs to distribute the load, the structural failure of the aircraft was found when preparing for the next test. It was decided that the failure damage was beyond the capability for reasonable repair and testing ceased. While the fuselage could have been replaced, thanks to the aircraft’s modular layout, consideration of other design faults resulted in the conclusion that lessons learned would be best applied to a third generation of the UAS.

The extended testing of the MKII design revealed several design faults that were either unrecognized in the MKI design or a result of modifications made to solve MKI design issues. The most impactful issue was the choice to shroud the propellers in the wing section for added safety and thrust efficiency. While the shrouds were appropriately sized for the originally selected motors, the 15” diameter created a limitation when the motors were sized up, and propellers had to be cut down to fit in the shroud. This requirement reduced the maximum thrust and efficiency the system could achieve, and also resulted in excessive noise under high load. Similarly, while the ducts were designed to maximize thrust efficiency, the resulting structure required to support longer wings, as well as maintain rigidity with a 15” hold cut through each wing, added enough mass to the aircraft that it more than canceled out any added thrust. Overall the final aircraft weighed in at near 27 lbs, with an extra battery required to support sustained high throttle due to the high weight of the aircraft. This was much higher than the predicted 19-20 lbs as many late changes had to be made to the design to support

larger motors required due to lower than expected battery performance under high current load. Much of the weight of this system came from the use of 3D printing to create many of the complex surface shapes required by the in-wing duct. While this created highly accurate and durable components, possible weight savings were limited due to the nature of 3D printing.

MKIII Design and Testing

The MK III prototype represents the current generation of the UAS and lessons learned from each previous system have been applied to this design. With high weight as the most significant issue of the previous generations, the main focus of the MKIII design was to reduce complexity even further and remove any components not completely necessary for flight or testing. As 3D printed components contributed the most mass to the first two generations, care was taken to ensure all profiles and surfaces were simple enough to be created with plywood rib structure and balsa sheeting skin. This move drastically reduced construction time as all the fuselage and wing ribs were cut in a few days' time, rather than the weeks required with previous generations. More so, ribbed structure creates an increased internal volume for component placement, allowing for more flexibility and an overall lower profile airframe. The use of aircraft plywood also increases repairability of the airframe in the event of any damage as wood can be glued back together with similar strength far easier than plastics can. Finally, while the use of a carbon fiber subframe was maintained from the MKII design, new Commercial-Off-The-Shelf (COTS) availability resulted in lighter weight tubes being paired alongside the plywood frame. As a result, the aircraft saw a 20-30% reduction in overall mass, at only 18-20 pounds depending on whether the outer wings are affixed or not. The outer wings were also able to be reduced in weight by

moving to a CNC cut foam rather than ribbed construction. This was made possible by removing control surfaces from the outer wing sections and replacing the aileron function with rotatable full length wings with inboard thrust vector/aileron surfaces.

Analysis of options available to increase roll stability and authority during vertical flight resulted in a 4 degree dihedral being added to the wing on either side. This is a dihedral consistent with most fixed low wing aircraft that provides self-righting characteristics in both vertical and horizontal flight. This allows the motors to be retained inboard, reducing the required aircraft structure as much as possible. Furthermore, the motors are mounted directly to the wing, with the entire wing rotating to a vertical position. This change is reflective of the need to reduce structure in the propeller wash, while also maintaining lifting surfaces to reduce overall required wing area and span. Mounting the motors directly to the wing also provides clearance for larger propellers, reducing the propeller loading in vertical flight and increasing maximum thrust and thrust efficiency. While the control system remained relatively unchanged from the successes of the MKII design, the Arduino Mega was replaced with an Arduino Due to make use of faster calculation speeds, resulting in a control update rate increase from 40 Hz to 80 Hz.

MKIII Test Fixture Stability Testing

The new airframe structure required the creation of a new test stand for control system modifications and tuning. The new test stand was designed to allow for more configurations of degrees of freedom. Whereas the MKII test fixture was limited to pitch, roll or yaw, the current test fixture allows for Pitch, roll, yaw-pitch, yaw-roll or all three; yaw, pitch, roll. Furthermore, the new system sets the roll axis at the same plane the propellers are on during vertical flight, reducing or eliminating the “inverted pendulum” effect witnessed during MKII testing, in which small roll disturbances yielded large

upsets inconsistent with real world dynamics. A render of the test stand system can be seen in figures 25 and 26.

Figure 25: MKIII Controller Test Fixture

Figure 26: MKIII Test Fixture with 3 Degree of Freedom Mount

Pitch Testing

Pitch stability success during MKII testing resulted in pitch being the first control tested with the new aircraft and test stand. Limited changes in the code, pitch control forces and test methodology meant that the results of this testing would give an appropriate baseline of the effects that universal changes, such as doubling the refresh rate of the control loop, on the control of the system. With minor tuning of the control gains, extremely stable and responsive pitch control was achieved, exceeding the results recorded during MKII system testing. In order to validate the stability and response of the system, three methodologies were used to simulate both worst case and normal operating conditions. Disturbance testing was accomplished by commanding the aircraft to a fixed position, then inducing several disturbances and recording the time to return to stable motion at the commanded positions. These disturbances were introduced by rapidly pushing upward or downward on the tail of the aircraft with varying magnitude of force. Figure 27 details the results of these tests, where the average time to return was approximately 1.8 seconds, well within the desired operating range. Furthermore, it is unlikely that a significantly better response time would be possible, as the control output saturates in several cases. The noise seen in the plot was confirmed to be the result of a processing error in the ground station recorder, not actual sensor readings.

Figure 27: MKIII Pitch Controller Disturbance Testing

Step response was also tested with instantaneous position command inputs of varying magnitude in order to measure the response time to achieve stability within 1 degree of the commanded position. These step inputs were in both the negative and positive direction, with an average response time of 1.2 seconds, as can be seen in figure 28.

Most significantly, the ability of the system to follow a continuously varying command input was tested in order to best simulate real world flight scenarios. In order to most accurately represent the type of input that would be seen during operational vertical flight, the command input was accomplished by simply releasing position control back to the remote control that is used during flight, then having the test operator input random and varying pitch positions. The significant data point observed during this test was the delay between the control system getting a command update, and the aircraft achieving that position. The average delay time observed was approximately 0.4 seconds, with figure 29 representing a plot of desired position versus actual position over several seconds. It can be seen that the system operated rapidly and predictably and executed the commanded positions near perfectly.

Figure 29: MKIII Pitch Controller Variable Command Testing

Roll Testing

Creating a stable and responsive control loop for the roll axis proved to be significantly more challenging than pitch. The gains used during testing of the MKII design were used as a base point as the distance between the main rotors was relatively similar. While initial testing appeared to result in a stable system, the responsiveness to command inputs was too slow for active control. Further testing and tuning revealed that further algorithm changes were required for the variable gain functions in order to handle the force differences when off from the level position. System stability was achieved using the same three tests executed during pitch control testing at a fixed throttle of approximately 60%. The results of these tests are plotted in figures 30, 31 and 32. The noise seen in the plots was confirmed to be the result of a processing error in the ground station recorder, not actual sensor readings.

Figure 30: MKIII Roll Controller Disturbance Testing

Figure 31: MKIII Roll Controller Stepped Command Testing

Figure 32: MKIII Roll Controller Variable Command Testing

While initial roll testing was concluded at this point, it was discovered, during initial tethered flight testing, that under very high throttle the system destabilizes. In order to handle this control system degradation with throttle increase, the system was re-tuned at 80% throttle, during which less aggressive gains proved to be necessary. This new control point was then combined with the initial data point at 60% throttle in order to extrapolate a linear gain reduction equation which proved to provide adequate stability at all throttle settings under approximately 90% throttle, as displayed in figure 33. The result of the roll tuning is clearly not as stable or responsive as pitch control, but it was suspected that under positive lift, the inverted pendulum effect still existed enough to create stability issues that would not exist in true flight. The final system gains resulted in a disturbance response of 2.5 seconds, a step response of 2.0 seconds and an average control delay of only 0.4 seconds. These response times, combined with the difficulty in accurately representing flight conditions, motivated concluding roll testing and proceeding with the next tests.

Figure 33: MKIII Roll Controller High Throttle Testing

Yaw Rate Testing

The fast acting nature of the thrust vector surfaces located directly behind the main thrusters resulted in a very fast acting and easy to tune control loop. Upon initial testing, it was found that the aircraft was immediately responsive to control inputs and was able to quickly counteract any induced torque from the motors. It was noted that the high update rate of the control system resulted in a fluttering effect of the control surfaces, and the gains were reduced slightly as such to minimize this effect. It was also noted, as figure 34 reveals, that while system saturation occurred nearly instantly during maximum yaw rate commands, the system was able to achieve the commanded yaw rate and desaturate the control outputs. This observation validated that the maximum yaw rate input command chosen was appropriate for control surface method used, which has significantly less authority capability when compared to the thruster rotation used in the MKII design. Due to the fact that precise yaw control stability is not required for aircraft survival, it was not tested to the same scrutiny as pitch and roll. Rather, a more significant emphasis was placed on predictable and responsive control from pilot inputs, resulting in a control lag of only 0.2 seconds on average.

Figure 34: MKIII Yaw Rate Controller Variable Command Testing

Mixed control Testing

Mixed axis testing was completed to validate the stability of each axis when extra degrees of freedom are added. Since the recording software can only record a single axis at a time, it was chosen to do this validation under user control using the remote controller, while recording video, with an aircraft fixed action camera, to check the response visually both during and post test. The control system proved to show no difference in stability or response when extra degrees of freedom were added, rather showing to be more stable for roll in which roll momentum could be transferred to the yaw axis. In total, three types of control freedom were tested, yaw-roll, yaw-pitch and yaw-pitch-roll. Each of these tests were recorded and reviewed to come to the conclusion that the aircraft has extremely satisfactory control authority and stationary stability. These tests concluded operations on the test fixture and provided the required guidance to move forward to tethered free flight.

MKIII Tethered free flight testing

A series of tethered free flight tests were conducted to validate the control system when not connected to the test fixture. These tests were conducted in the same format at the MKII tethered testing, in which a significantly elongated handle was attached to the aircraft, allowing for a test assistant to be hands-off while still maintaining full control if something were to go wrong. Several control issues became apparent during this testing and allowed for appropriate software changes to be made without risking the aircraft. As discussed in the “Roll Testing” section, moments of instability would happen during takeoff, when the thrusters were at their highest power level. After a second round of test fixture testing was completed to alleviate this condition, flight control testing proceeded to ensure stability and control was possible on all axes. While

stable and responsive control was confirmed in all axes, it was also found that forward and back control was not adequate to sustain position in light winds. This control was accomplished through forward and aft vectoring of thrust using the same control surfaces responsible for yaw control. In order to ensure lateral control remained possible in adverse flight conditions, a pitch input was also mapped to the forward and back input from the user. As such, when the user commanded a forward motion, the aircraft would pitch downwards as well as vector the thrust. These combined motions yield a more significant change in the thrust vector, allowing the aircraft to maneuver appropriately. Most significantly, it was discovered that the large thrust overhead of approximately 15 lbs., or approximately 40% of the total thrust, made maintaining altitude and controlling descent rate nearly impossible with the limited resolution of throttle control available to the pilot. As detailed in the section "LIDAR Testing" it was decided to incorporate a high frequency LIDAR into the nose of the aircraft in order to modulate the aircraft's throttle automatically to maintain altitude or a desired descent or ascent rate. The added fidelity of control provided with the LIDAR controller represented the last step required to conclude tethered flight testing to move towards full free flight testing of vertical flight.

LIDAR Testing

In order to maintain altitude and controlled descent rates, a LIDAR control loop was added during tethered free flight testing. This control loop would be impossible to test in a test fixture, as it required the true response of the aircraft in vertical flight. Further complicating testing and tuning of this control loop was the fact that the user would be giving over direct control of the motor throttle to the control system, presenting a less controlled situation and a more dangerous scenario for the test assistant. In order to ensure a safe operating envelope, the LIDAR controller was implemented over several phases including fixed altitude hold, controlled descent velocity, and finally

controlled altitude hold. Explanation of the coding framework for the final flight mode is detailed in the “Control System” section of “Detailed Design”.

Initial LIDAR control loop testing was accomplished with a simple “hold altitude” function activated by a switch on the hand held controller. As figure 35 highlights, this test was immediately successful and resulted in very stable control at a fixed altitude with minor variations. While this controller could likely have been further improved, the goal was for controlled ascent/descent rate control, so the test was concluded as a successful proof of concept and testing proceeded to ascent/descent velocity control.

Figure 35: MKIII Altitude Hold Controller Tethered Testing

Ascent/descent velocity control relied on a calculated vertical velocity of the aircraft based upon the LIDAR measured distance. Due to the limited resolution of 1cm of the sensor, the resulting measurement had to be averaged and filtered in order to provide controller input data. Achieving reliable ascent/descent rate control proved to be more difficult, especially as the aircraft entered and exited ground effect when within approximately 0.5 meters of the ground. Control gains were eventually found during low altitude tethered flight testing that accomplished the goal of smooth landing characteristics, but resulted in data showing significant variability compared to the commanded rate. This variability became significantly more problematic when the aircraft was not limited to approximately 2 meters of altitude. During the first several seconds of the full free flight testing, the ascent/descent rate control proved to be unstable, resulting in growing oscillations of throttle and altitude of the aircraft, requiring the pilot to immediately disable the control for the remainder of the flight and manually control altitude. Furthermore, the lack of this control functionality resulted in wing rotation control linkage damage during landing due to excessive descent rate.

The final iteration of LIDAR testing represented an updated control loop design in which the pilot simply varied the commanded height, and the control system worked to achieve that altitude. This functionality works off direct sensor data and is most like the original LIDAR testing that proved to be far more stable. The testing once again started with a tethered scenario until control gains were tuned to a stable condition. This system design instantly proved to be much more robust and resulted in the very stable operation seen in figure 36 with in only 3 iterations of system tuning. This functionality was again tested during the takeoff and climb phases of the flight transition testing. The data collected during this test proved that the system worked flawlessly and maintained the commanded altitude with an average error of only 15cm, all the way up to the transition altitude of 15 meters. This data can be seen in figure 37. While the flight transition

testing was overall a failure, the data collected provided enough evidence that the LIDAR controller was extremely functional.

Figure 36: MKIII Ascent/Descent Controller Tethered Testing

Figure 37: MKIII Ascent/Descent Controller Free Flight Testing

Vertical Free Flight Testing

Vertical free flight testing validated the data collected during tethered testing and proved to be highly successful. The flight plan called for an ascent/descent-controlled ascent to approximately 10 meters, followed by maneuvers about and along each axis to confirm a fully controlled state, finally followed by a descent-controlled landing.

Unfortunately, as evidenced below in Figure 38, a plot of the throttle outputs during the initial moments of the flight, the ascent/descent controller proved to be unstable and had to be disengaged to recover the aircraft.

Figure 38: MKIII Descent Controller Free Flight Instability

Due to the robust control system, the aircraft was able to correct itself and level out, even with the motors momentarily entering an off state just as the ascent/descent controller was deactivated. Once stable flight was achieved, the pilot lowered the aircraft altitude and continued to adjust throttles manually for the remainder of the flight to maintain a low flight level. Each control test was then executed in sequence, starting with a counterclockwise, then clockwise yaw rotation, followed by rolling to the left then right, simultaneously inducing lateral motion left then right. Finally, forward and rearward motion were commanded, inducing a simultaneous negative and positive pitch command respectively. With these maneuvers all proving stable and responsive, the flight was concluded with a user controlled landing 110 seconds after takeoff. The data recorded during this operation for each axis is plotted in figures 39, and 40. Yaw is not able to be plotted as heading is recorded rather than yaw rate, as it is more useful data for the user ground station interface. Unfortunately, the manual throttle operation required the pilot to land the aircraft at a higher rate of descent than the landing gear was designed for, resulting in a nose over effect. The nose over resulted in light damage to the nose of the aircraft and significant damage to the control linkage between the linear actuators and the wing rotation rod. Though the damage was repairable, it was decided that, with the controllability and recoverability of the system, the best path forward was to conclude vertical flight testing and move into transition and horizontal flight testing, to avoid another lengthy repair period before final testing could be completed. Furthermore, an extra landing gear point was added to the nose to minimize the risk of this damage upon a future landing.

Figure 39: MKIII Free flight Pitch Angle Actual vs Commanded

Figure 40: MKIII Free Flight Roll Angle Actual vs Commanded

Flight Transition Testing

The most significant design issue with the current generation of the aircraft, and the limited resources available, was the inability to test horizontal flight or transition to or from horizontal flight in any meaningful way prior to full-up flight conditions.

Unfortunately, these limited test conditions created a scenario in which the test would either be successful and result in a flying aircraft, or would be unsuccessful and result in a crashed aircraft. In this case, the result was the latter and the aircraft ended up in a state far beyond reasonable repair. Fortunately, as the control system actively sends a significant amount of data to the ground station during flight, a large amount of data was able to be collected and analyzed. Discussed here are the main points of the flight and the corresponding data as well as conclusions based on analysis of video and data of what caused the failure to transition that ultimately ended the flight prematurely.

The flight plan called for a LIDAR controlled takeoff, followed by a climb to approximately 15 meters. At this point, the aircraft would be rotated to point directly into the wind and the wings would be rotated gradually as the aircraft increased in horizontal velocity until full horizontal flight was achieved. The transition would have been followed by a main thruster throttle reduction and a period of horizontal flight before a gradual transition to vertical and a LIDAR controlled landing. The LIDAR ascent/descent controller was updated prior to this flight and was being tested during full free flight operations for the first time. This new controller automatically raises the aircraft to a flight level of 125 cm when the pilot arms the aircraft with ascent/descent control activated. At this point, the aircraft maintains altitude until further ascent/descent commands are given. This functionality worked as expected as can be seen in figure 41, a detailed plot of commanded height vs actual height during the first phase of the flight.

Figure 41: MKIII Automated Takeoff Height Response

The aircraft was then commanded to a flight level of approximately 15 meters, using the automatic ascent/descent controller, and oriented away from observers and directly into oncoming wind. The transition control was gradually incremented towards horizontal flight as the aircraft was commanded to continue to increase altitude in order to ensure the motor throttles were at a high setting. At a wing angle of approximately 45 degrees off the horizontal, the first indications of problems arose when the aircraft suddenly pitched upwards, then downwards before the pitch controller returned the system to level flight. This action prompted the pilot to expedite the transition to horizontal flight and increase throttles to maximum in order to restore the required flight speed. Upon completion of transition, the aircraft entered a steep dive at approximately 70 degrees off the horizontal (Figure 42). Full elevator input was required to recover from the dive before the aircraft impacted the ground, subjecting the aircraft to a significant g-force beyond the expected design limits (Figure 43). During the recovery maneuver it was witnessed that the outer wings suffered from a severe upward deflection (Figure 44), followed by a severe negative deflection upon leveling out (figure 45). Following the negative deflection, the aircraft returned to a dive and impacted the ground at significant velocity (Figure 46).

Figure 42: MKIII Free Flight Pitch Error During Transition

Figure 43: MKIII Vertical Acceleration During Dive Recovery

Figure 44: MKIII Wing Bending Upwards During Dive Recovery

Figure 45: MKIII Wings Bending Downward Following Dive Recovery

Figure 46: MKIII Secondary Sudden Dive Resulting in Crash

Upon reviewing all the data, it is believed that multiple design and control issues, as well as pilot error, compiled to create an unrecoverable situation. It is theorized that the initial pitch stability issues during transition were a result of the rear thruster moving into the position in which its airflow stream would pass over the tail at an angle (figure 47). This effect would create a sudden downward force on the tail and pitch the aircraft upwards, as was witnessed. As the control system attempted to compensate by increasing the throttle of the rear motor, the effect would worsen, but also be counteracted momentarily by the increased thrust of the motor, creating an overall unstable and unpredictable result. Review of the data also shows that this pitch motion occurred just as the control system entered "flight mode 2" (figure 48), which begins to reduce the effect of the PID controls and transitions the controls to conventional horizontal flight controls. It is also possible that this controller transition was not as smooth as expected and created a controller instability. With the aircraft re-stabilized, the pilot considered the two options were rapidly continuing the transition to horizontal flight, or returning to vertical flight and abandoning the transition test. With the aircraft momentarily stable, it was decided to continue the transition, whereas in reality, the best flight action would have been to reduce the transition angle to reestablish horizontal velocity, as the pitching motions had bled off much of the flight speed. This decision likely led directly to the dive following the transition, likely an effect of the aircraft stalling due to very low velocity and total removal of vertical lift.

Figure 47: Rear Thruster Angled Airflow Over Tail Assembly

Figure 48: Pitch Error vs Controller Flight Mode

The next issue to compound the flight state of the aircraft was the large flex seen in the wing during the maneuver to recover from the dive. Unfortunately, due to the low altitude once the aircraft recovered enough horizontal velocity, maximum controller inputs were required, far exceeding the maximum 2g force the airframe and wing were designed to at a peak of 4g (figure 43). This extreme load subjected each wing to a distributed load of approximately 18 kg each. Limited video evidence, along with a review of the mechanical design leads to the theory that most of the bending occurred along the outer wing structure and in the connection between the inner wing and the fuselage. The choice to use a lighter weight, thin wall carbon fiber tube as the main wing structure resulted in a twill weave with minimal strand count in the longitudinal direction, and a lower resistance to bending moment under high loads, whereas the double layer tube used in past generations used a longitudinal wrap in conjunction with the lateral wrap, resulting in higher bending moment stiffness.

The upward wing deflection was quickly followed by a severe downward wing deflection, signifying a negative lift of similar magnitude, ultimately resulting in rapid altitude loss and impact. Based upon design limitations and video evidence, it is suspected one, or both, of the following situations occurred to result in negative lift. Foremost, after the near ground recovery from the dive, the pilot rapidly reduced the pitch up input to reduce the risk of a high speed stall while only a few meters off the ground. This sudden control surface change would have rapidly changed the vertical velocity of the aircraft and resulted in the part of the aircraft with the highest inertia, the fuselage, to continue upward as the wings were pulled downward by vertical drag effects. The downward deflection of the wings may have induced a twisting effect in which the wingtips were at a negative angle of incidence, creating negative lift and worsening the effect further. This twisting is not visible in the video, nor was it witnessed in person, but the viewing angles may have obscured such from being noticeable.

Similarly, failure of the linear actuators ability to hold position was considered as a possibility. During the dive recovery, when the wings experienced a load of approximately 18kg, the actuator would have experienced a similar back drive force as an effect of the position of the wing's axis of rotation compared to the aerodynamic center lift. This load of approximately 175 newtons of force far exceeds the 100 newton holding force specification of the linear actuator. If the linear actuator was back driven it would result in the entire wing entering a negative incidence, and negative lift scenario. As soon as the aircraft's vertical inertia, and therefore induced higher angle of attack, was spent, the airflow over the wings would have created negative lift and resulted in the witnessed downward bending of the wings. While the resulting crash was not the desired outcome, the data recorded from the flight, especially that of the transition and witnessed mechanical design issues, yields a wealth of useful knowledge for future work towards this effort.

Conclusions

Development of a high efficiency, transitioning VTOL flight system represents an industry goal with a long history and many possible solution paths. While the latest iteration of the UAS did not transition to horizontal flight during testing, the data collected from the flight will provide the information needed to make future iterations successful. Each prototype generation resulted in significant improvements to the overall design with increasing headway towards the final goal of low cost VTOL capability with high efficiency horizontal flight. Though the MK I airframe was never subjected to flight testing, it provided a base to develop the control system foundation and structural design requirements. The MK I design resulted in many lessons learned about modularity of design, limitations of mechanical actuators, and possibilities for weight reduction. The MK II, developed from required changes to the MK I layout, was able to achieve basic controlled hover functionality, but little more due to its inefficient thrusters and overweight airframe. The test stand testing accomplished on the MK II, however, provided important insight into the control system design required to create a stable aircraft. Furthermore, it revealed the limitations of additive manufacturing for strength and weight reduction requirements. In a similar fashion, the MK III prototype achieved much more than the MK II with fully controlled vertical flight and a flight ready weight reduction of approximately 25%. The MK III also presented many more lessons to be incorporated into future interactions of the design.

Most importantly, the MK III's success in vertical flight, and its ability to recover from the large disturbances witnessed in test flights, provide evidence that the dis-similar control format is stable enough to be used as the VTOL control for payload capable airframes. Furthermore, the use of thrust vectoring for yaw control resulted in far more stable and faster responding control around that axis. This is a likely an effect of the

faster actuation speed to shift a control surface angle compared to that of rotating an entire motor, therefore outweighing the reduction in total thrust vector deflection capability. Another successful feature to be brought forward into future iterations is the use of maximum size propellers, and the controls required to minimize thruster airstream interference. The large thrust output overhead compared to the reduced weight of the aircraft yielded a vertical flight power draw reduction from approximately 150 amps on the MK II system to approximately 60 amps on the MK III system, extending the maximum flight time by over a factor of 2. Similarly, combination of “built up” construction and a boom style tail technique used in the design of the MK III aircraft created an airframe with more internal volume while weighing over 40% less than the MK II airframe. It was concluded, however, that the main airframe was likely overbuilt and weighed more than necessary evidenced by the comparably intact status of the fuselage after an extremely high velocity impact with the ground (figure 49).

Figure 49: Damage to MKIII Following Crash

Data collected during the failed transition test has led to several significant design changes to be incorporated into future iterations. As discussed in the testing section, it is theorized that the pitching instability during the transition was induced by angled rear thruster wash over the tail. In order to resolve this issue, future designs should ensure that no large surfaces are in a propeller thrust stream during any point in the rotation arc between vertical and horizontal flight. This may be accomplished by changing the orientation or position of the stabilizers, or by deactivating and rotating the rear motor only after horizontal flight has been achieved. A structural weakness of the design emanated from the single point of connection required to rotate the wings to the vertical position combined with an incorrect assumption of the loads the aircraft wings would be subjected to under abnormal flight conditions. Future designs should make use of a stronger axle connection material or develop a solution to add to the connection rigidity. Similarly, the use of CNC foam outer wings provided an extremely low weight wing, but were unsuccessful due to the flexibility of the carbon rod used to stiffen the wing. The results of the testing should not be defined as the failure of the foam cut outer wing, but of the substructure supporting it, which will require reinforcement if this design solution is used again. The expected failure of the linear actuators to maintain position is an extremely important aspect to be revised in any design that implements rotating airfoils, which experience varying and hard to predict loads. A proposed solution to this design flaw is the use of a worm gear type actuator with position control and feedback. The use of a worm gear would preclude the possibility of a back-drive event, while feedback control would inform the control system and the pilot of a potential system fault. Finally, it became clear during vertical flight testing, that the implemented landing gear design was not capable of absorbing a reasonably harsh landing load. Though the integrated landing gear into the wing tips reduced the aircraft weight a great deal, it subjected key

components to unnecessary stress, and should be substituted for a more survivable solution in later iterations.

VTOL flight is a difficult problem, made more possible by recent advances in battery, controller, and manufacturing technologies. The successes of each iteration have served to prove that the dis-similar tri-rotor design has is a possible solution to the efficiency goals, while the failures of each prototype has paved a path forward with more information and lessons to apply to the next design. Though the system has not yet achieved a full flight profile, the design is significantly closer to success than at the onset of the project, and has provided a wealth of information for future design work.

Future work

The work accomplished towards the goal of an efficient, transitioning VTOL aircraft has gone through several iterations of design and resulted in significant progress. The conclusion of MK III testing also serves to show there is still work to be completed before the design goals are achieved. As evidenced, the most difficult part of testing the MK III was the inability to test horizontal or transitioning flight in any capacity prior to actual flight, in which an error may, and did, lead to the aircraft's destruction. Future work on this topic should focus on developing an aircraft, or test methodology so that horizontal flight characteristics can be catalogued prior to the transition test. Furthermore, flight transition stability is a complex element of the flight with variable airflow and forces. The requirement to witness these effects for the first time while in flight represents an extremely high risk scenario with extensive design rework if a problem formulates during testing. Future development may focus on the creation of transition flight testing methodology in a controlled environment, while still subjecting the system to the variability of the real world dynamics. Throughout the development effort, the disadvantages of an electric power system were also exposed. Most notably, is the Specific energy density gap between gasoline (46 MJ/kg) and lithium polymer batteries (0.36 - 0.95 MJ/kg), resulting in a massively increased dry weight when compared to a gasoline engine system with similar flight power and endurance. Similarly, while gasoline provides a flat power curve as the fuel tank is drained, a battery will taper off power output once the battery is approximately 75% discharged. While a gas engine creates a more complex control environment, as power output cannot be modulated rapidly to control pitch and roll, use of an internal combustion engine may result in a more scalable system with significantly increased flight endurance if the control system concerns can be addressed.

Appendix 1: Lift Calculations

Table 7: Component Weights

Total Weight (oz)	277.4793 316		
Total Weight (grams)	8238.7		
Name of Item	Quantity	"Weight" (grams, metric)	Total Weight (gram)
Great Planes Rimfire 1.20 50-65-450 Outrunner Brushless	2	400	800
Castle Creations 90A Talon ESC	2	186	372
22.2V 10000mAh 6S Cell 25C-50C LiPo Battery Pack w/ XT150 Connector Plug	1	1200	1200
Great Planes Rimfire .46 42-60-800 Outrunner Brushless	1	268	268
Castle Creations Talon 60 Amp esc	1	57	57
Xoar 19x6 PJP-N-M Multi Rotor Prec Prop Coated	2	86	172
Xoar 10x6 PJA Series Beechwood Propeller	1	14	14
Hitec HS-7954SH High-Voltage Ultra-Torque Dual BB Servo	2	65.2	130.4
Traxxas High Torque Waterproof Servo E-Maxx	2	45	90
Arduino due	1	59	59
Adafruit 16-Channel 12-bit PWM/Servo Shield - I2C interface	1	28	28
Adafruit Ultimate GPS Breakout - 66 channel w/10 Hz updates - Version 3	1	10	10
GPS Antenna - External Active Antenna - 3-5V 28dB 5 Meter SMA	1	28	28
Adafruit 9-DOF Absolute orientation IMU BNO055	1	3	3
180A atto pilot current sensor	2	3	6
RMILEC High-Precision PWM/PPM/SBus Signal Converter V2	1	10	10
EMS Heavy Duty Extension 36" Futaba J	7	8.5	59.5
Airframe construction	1	4500	4500

Tactic Servo Extension 6" Futaba J	8	2.8	22.4
Receiver 2.4GHz	1	12.4	12.4
Level Shifter	1	3	3
L16-R linear actuator	3	84	252
L12-R linear actuator	3	40	120
Lidar Lite V3	1	22	22

Equation 1: Reynolds number estimation

Re = Reynolds number

ρ = Density of air at sea level at 20 degrees C = 1.2401 kg/m³

v = Velocity = 20 m/s

l = characteristic length of airfoil chord = 0.325 m

μ = Dynamic Viscosity of air at sea level at 20 degrees C = 1.8205 x 10⁻⁵ kg/ms

$$Re = \frac{\rho * v * l}{\mu} = \frac{1.2401 * 20 * 0.325}{0.000018205} = 442,771$$

Equation 2: Transition velocity based on lift force

V = Transition Velocity

L = Lift force = 9.22 Kg = 90.42 Newtons

C_L = Coefficient of Lift = 0.596

ρ = Density of air at sea level at 20 degrees C = 1.2401 kg/m³

A = Area of lifting surface = 1.040 m²

$$V = \sqrt{\frac{2 * L}{C_L * \rho * A}} = \sqrt{\frac{2 * 90.42}{0.596 * 1.2401 * 1.040}} = 15.338 \frac{m}{s} = 34.31 \text{ mph}$$

Appendix 2: Stability Calculations

Stability Margin Calculations

b = Wing Span = 3.2 meters

c = Average Wing Chord = 0.325 meters

S = Wing area = $b * c = 3.2 * 0.325 = 1.040$ meters

AR = Wing Aspect Ratio = $\frac{b}{c} = \frac{3.2}{0.325} = 9.846$

C_L = Coefficient of Lift = 0.596

Y = Wing Dihedral Angle = 4 degrees

$T\alpha$ = V Tail Separation Angle = 45 degrees

T_c = V Tail Chord = 0.3 meters

T_b = V Tail Total Width = 0.5 meters

S_h = Horizontal Tail Area = $T_c * T_b = 0.3 * 0.5 = 0.15 \text{ m}^2$

S_v = Vertical Tail Area

$$S_v = 2 * \left(\sin(T\alpha) * \frac{\frac{T_b}{2}}{\cos(T\alpha)} \right) * T_c = 2 * \left(\sin(45) * \frac{\frac{0.5}{2}}{\cos(45)} \right) * 0.3 = 0.15 \text{ m}^2$$

L_h = Horizontal Tail Moment Arm = 0.9 meters

L_v = Vertical Tail Moment Arm = $L_h = 0.9$ meters

AR_h = Horizontal Tail Aspect Ratio = $\frac{T_b}{T_c} = \frac{0.5}{0.3} = 1.66$

V_h = Horizontal Tail Volume Coefficient (*acceptable range: 0.3 to 0.6*)

$$V_h = \frac{S_h * L_h}{S * c} = \frac{0.15 * 0.9}{1.040 * 0.325} = 0.3994$$

V_v = Vertical Tail Volume Coefficient (*acceptable range: 0.02 to 0.05*)

$$V_v = \frac{S_v * L_v}{S * b} = \frac{0.15 * 0.9}{1.040 * 3.2} = 0.0407$$

$$X_{np} = \text{Neutral Point} = c * \left(0.25 + \left(\left(\frac{1 + \left(\frac{2}{AR} \right)}{1 + \left(\frac{2}{AR_h} \right)} \right) * \left(1 - \frac{4}{AR + 2} \right) * V_h \right) \right)$$

$$X_{np} = 0.325 * \left(0.25 + \left(\left(\frac{1 + \left(\frac{2}{9.846} \right)}{1 + \left(\frac{2}{1.66} \right)} \right) * \left(1 - \frac{4}{9.846 + 2} \right) * 0.3994 \right) \right)$$

$X_{np} = 0.1283$ meters back from leading edge

$S.M.$ = Desired Stability Margin (*acceptable range: 0.05 to 0.15*) = 0.05

X_{cg} = Center of gravity (*acceptable range: 30% to 40% of c*)

X_{cg} range = 0.0975 to 0.13 meters back from leading edge

X_{cg} actual = $X_{np} - (S.M. * c) = 0.1283 - (0.05 * 0.325)$

X_{cg} actual = 0.112 meters back from leading edge

Appendix 3: Wiring Schematics

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