

University At Buffalo

Goose Works
Design • Build • Fly

2023-2024
Design Report



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Abbreviations, Acronyms, and Symbols

AIAA = American Institute of Aeronautics and Astronautics

AOA = Angle of Attack

AR = Aspect Ratio

ASA = Acrylonitrile Styrene Acrylate

AVL = Athena Vortex Lattice

b = Wing Span

BC = Battery Capacity

BLDC = Brushless Direct Current

b_w = Wingspan

c = Chord Length

CAD = Computer-Aided Design

CFD = Computational Fluid Dynamics

CFRP = Carbon Fiber-Reinforced Polymers

CG = Center of Gravity

$C_{D,T}$ = Total Drag Coefficient

$C_{D,i}$ = Induced Drag Coefficient

$C_{D,o}$ = Parasitic Drag Coefficient

C_{HT} = Horizontal Tail Volume Coefficient

C_{VT} = Vertical Tail Volume Coefficient

CL = Coefficient of Lift

CL_{max} = Maximum Coefficient of Lift

CNC = Computerized Numerical Control

C_{VT} = Vertical Tail Volume Coefficient

C_w = Wing Chord

$D_{D,0}$ = Wing Chord

DBF = Design/ Build/ Fly

EMT = Emergency Medical Transport

EPS = Expanded Polystyrene

ESC = Electronic Speed Controller

FDM = Fused Deposition Modeling

FEA = Finite Element Analysis

FPV = First Person View

FT = Flight Test

GM = Ground Mission Score

GSM = Grams Per Square Meter

LiPo = Lithium Polymer

L_{VT} = Vertical Tail Length

L_{HT} = Horizontal Tail Length

M1 = Mission 1

M2 = Mission 2

M3 = Mission 3

MSC = Medical Supply Cabinet

PW = Mission Time

MTOW = Maximum Take-Off Weight

NL = Number of Laps

NP = Number of Passengers

P = Participation Score

P&E = Propulsion and Electronics

PLA = Polylactic Acid

PWM = Pulse Width Modulation

PWT = Payload Weight

Re = Reynolds Number

RPM = Revolutions Per Minute

SD = Selig-Donovan

S_{HT} = Horizontal Tail Area Coefficient

ST = Systems Tests

STOL = Short Takeoff and Landing

S_{VT} = Vertical Tail Area Coefficient

S_w = Wing Planform Area

$t_{min,M}$ = Lowest Mission Time

$t_{min,N}$ = Individual Score for Condor

UAM = Urban Air Mobility

UAV = Unmanned Aerial Vehicle

V1 = Version 1

V2 = Version 2

V3 = Version 3

V_{∞} = Average Velocity

XPS = Extruded Polystyrene Insulation

α = Angle between x-axis and apparent airspeed

1 Executive Summary

This report is intended to document the design, manufacturing, and testing conducted by the University at Buffalo's Design/Build/Fly (UB DBF) Goose Works team during the development of Condor. For the 2023-2024 American Institute of Aeronautics and Astronautics (AIAA) DBF competition, Goose Works developed an Unmanned Aerial Vehicle (UAV) capable of completing three Urban Air Mobility (UAM) missions and one Ground Mission (GM). Mission 1 (M1) involved a staging flight demonstrating Condor's ability to complete 3 laps around the course. Mission 2 (M2) simulated a medical transport flight with a medical payload while completing 3 laps as fast as possible. Mission 3 (M3) entails an urban taxi flight designed to maximize the number of laps and passengers flown in 5 minutes while minimizing battery capacity (BC). For all aerial missions, Condor is required to enter the 2.5 ft staging box in the parked configuration without the propulsion battery. The GM is a configuration demonstration to showcase efficiency by transitioning from the parked to flight configuration and changing between the various mission payloads. The aircraft must adhere to strict operational requirements of a 5-minute staging window and 20 ft takeoff distance.

Goose Works comprises 30 active undergraduate students who designed, manufactured, and tested Condor. The team used an iterative design process to optimize each version developed, encompassing conceptual, preliminary, and detail design phases. In the conceptual design phase, the team identified key system requirements and constraints, performed a sensitivity analysis, and developed the initial configuration. During preliminary design, optimization and trade studies addressing take-off performance, control, static, and dynamic stability were performed. In the detailed design phase, the focus shifted to the integration of system components with particular attention to payload integration and constraints.

Condor is designed as a high-wing, twin-propeller, T-tail, tricycle gear aircraft featuring a streamlined fuselage and wing endplates. Twin propellers and blown flaps provide the necessary thrust and high-lift devices to minimize take-off distance. The high wing design allowed for subsystem integration of a structural skeleton within the wings, tail, and composite ribs, reducing the weight of the fuselage. The tricycle gear allows the passengers to be easily loaded and folding the wings at the fuselage complies with the 2.5 ft parking spot size. The streamlined aerodynamic profile adheres to the rectangular cabin and facilitates easy access to electronic components. Wing endplates are employed to reduce wingtip vortices and enhance lift, primarily during take-off. The landing gear is positioned beneath the fuselage, while carbon fiber spars run through the wings and the fuselage connects to the tail. For M2, a hatch is positioned forward of the wing leading edge. For M3, the hatch is at the side rear of the Condor. To optimize the team's score, Condor has the payload volume to carry 64 passengers, a 17.5 lbf maximum take-off weight (MTOW), a 7 lbf medical supply cabinet (MSC), 82.08 W-h battery, and a maximum cruise velocity of 125 ft/s.

2 Management Summary

Goose Works comprises 40% seniors and 60% underclassmen who worked under five sub-team leads and reported to two captains who led and shaped the design process. The club was assisted by one faculty advisor and was funded by The University at Buffalo Student Association (UB SA) and The School of Engineering and Applied Sciences (SEAS).

2.1 Organization Description

Goose Works includes five executive board members, two captains, and five team leaders. The hierarchical structure of the UB DBF team, depicted in Fig. 1, is organized such that relevant tasks can be distributed to the appropriate authority. The team consisted of one pilot and two captains who oversaw five sub-teams: Propulsion and Electronics (P&E), Flight Dynamics, Structures, Manufacturing, and Mission. The captains worked to break down tasks by providing design constraints and scheduling appropriate deadlines to ensure the competition rules were followed. The captains also supervised the entire design process from its conceptual phase to manufacturing and checked the quality and functionality of all system components.

During bi-weekly meetings, all sub-team leads reported on major developments to ensure seamless and timely integration of all components. At the end of each week, sub-team leads provided a write-up of weekly progress that was made available to all members to further ensure synchronization between members and system components. Conflicts between sub-teams were resolved by reporting differing ideas to captains who would deliberate the topic with explanations as to why each lead believes their methodology should be chosen. After comparing each idea, the captains would determine the method of action that reflected the team's goals for the competition.

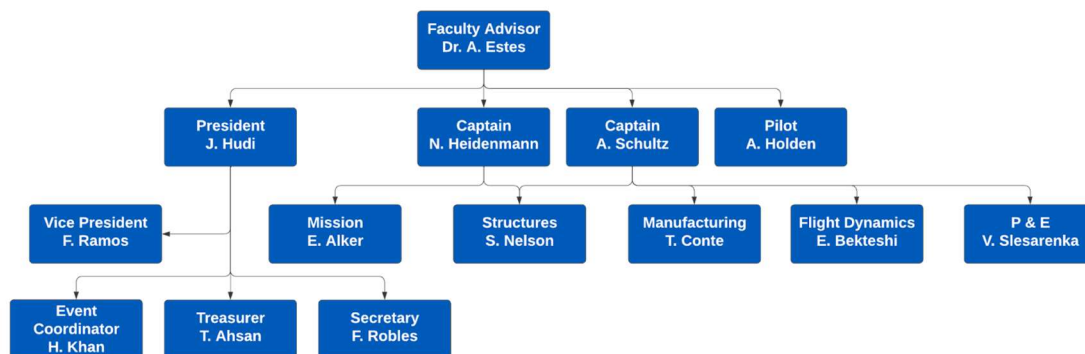


Figure. 1. Organizational Structure

All team members developed in various technical disciplines based on interest and collaborated with sub-teams during weekly meetings to assist in design and manufacturing tasks to help build Condor. The board members handled fundraising, and material procurement, and fostered a well-organized working environment while collaborating with the DBF team leads and university officials. The faculty advisor, Dr. Aaron Estes, offered valuable support by assisting the team with administrative tasks, ensuring that the club met all university requirements, and provided professional guidance to the team.

2.2 Project Schedule

Gantt charts were developed to organize the team's workload and project milestones to ensure that all deadlines were met as shown in Fig. 2. A secondary task list was used to issue deadlines and determine the progress on target deliverables to organize the jobs assigned to sub-teams. Through constant tracking, a high-performance aircraft was efficiently developed for competition.

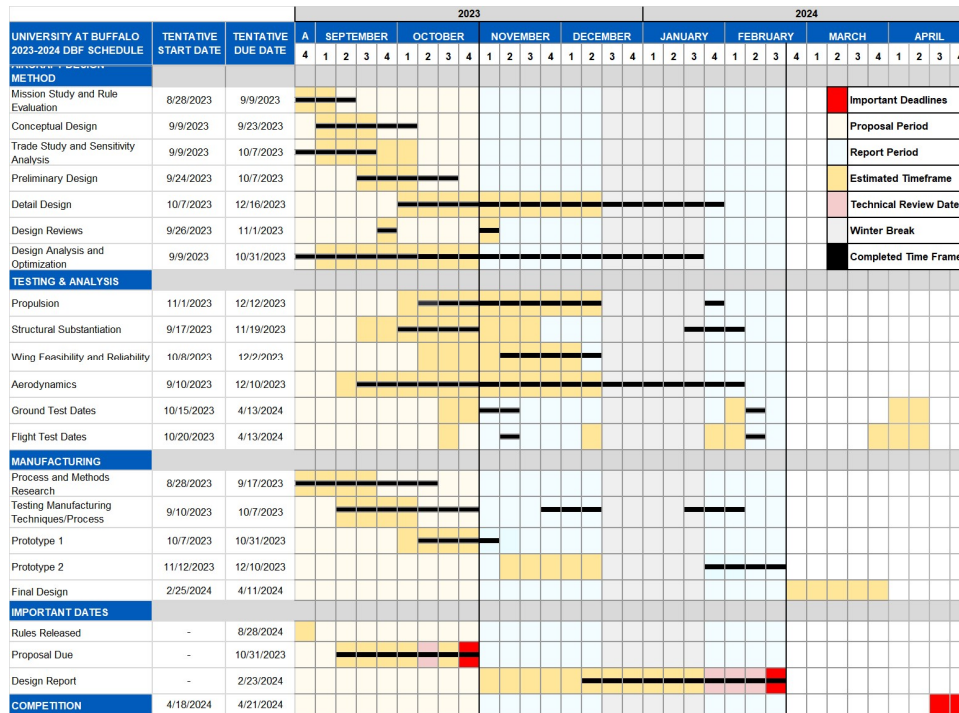


Figure 2. Central Gantt Chart

3 Conceptual Design

An aircraft capable of completing all mission objectives and performance criteria was developed in the conceptual design phase. The initial design required a comprehensive approach consisting of precise system requirements, implementation of decision matrices, and sensitivity analysis as well as optimization of defined parameters while considering the various design constraints.

3.1 Mission and Score Summary

The focus of the 2023-2024 competition is UAM. The score is contingent upon three flight performance missions, one ground mission, and a design report. The individual mission scores will be summed and multiplied by the Design Report Score, then added to the Participation Score (P) to produce the Total Competition Score, as shown in Eq. 1 and 2. Participation Scoring is given a score of 1, 2, or 3, based on attending the fly-off, completing Tech Inspection, and attempting a flight mission. In general, the goal is to design an aircraft capable of achieving the highest mission score of 6 and develop a well-documented design report to scale the competition score [1].

$$\text{Competition Score} = \text{Design Report Score} * \text{Total Mission Score} + P \quad (1)$$

$$\text{Total Mission Score} = M1 + M2 + M3 + GM \quad (2)$$

3.1.1 Staging

Before each mission, Condor will be brought to the staging box in its parking configuration with all components including the gurney, MSC, floor insert, and battery packs removed. The flight attempt will be forfeited if any required material is forgotten after entering the staging box. The assembly crew member, pilot, and observer are the only people allowed in the staging area. The assembly crew member is the only person allowed to prepare the plane for flight. Condor must be configured while upright on its landing gear, prohibiting it from being assembled in any other position. An exception to this rule is allowed when checking balance using the wing tips. The entire assembly must be completed in 5 minutes, except the arming fuse, which must be installed before being called to the flight line. All missions require the plane to take off within 20 ft with all landing gear forward of the start line and follow the flight path shown in Fig. 3. The flight timer starts when the throttle is advanced for the first attempt at take-off and is considered completed only for a successful landing.

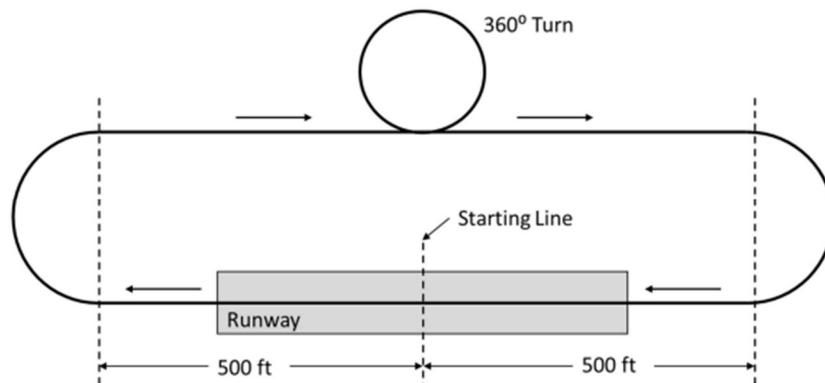


Figure 3. Course Layout

3.1.2 Ground Mission

The GM is timed and can be completed at any point throughout the competition. Condor starts in the parking spot with only the assembly crew member permitted to touch it. Condor must be removed from the parking spot and loaded sequentially with the different mission payloads set up in the terminal. The time starts every time the GM judge says “GO” and the assembly crew member configures Condor for the flight configuration with the respective payload; the time then stops when the assembly crew member says “STOP.” Doors are checked to ensure security; then flight controls are tested by the pilot after the completion of each configuration. The first configuration includes preparing Condor for flight and loading crew members and materials for M2: MSC, gurney, patient, and EMTs. The second configuration includes unloading the M2 payload while leaving the crew members loaded and inserting the M3 payload: The maximum number of passengers declared at Tech Inspection. Finally, all crew members and passengers are removed, and Condor must be returned to the parking configuration. Points are rewarded based on Eq. 3 where $t_{min,M}$ is the lowest mission time for all teams and N is the individual score for Condor.

$$GM = \frac{t_{min,M}}{t_{min,N}} \quad (3)$$

3.1.3 Mission 1

M1 payload consists solely of pilots. To successfully complete this mission, there is a 5-minute staging window to change from the parked configuration to the flight configuration followed by the completion of 3 laps within a 5-minute window. A lap is considered complete when the airplane passes over the start line. Landing is not a part of the time constraint, and one point is awarded for a successful flight and landing.

3.1.4 Mission 2

M2 simulates a medical transport flight. The payload includes EMTs, a patient on a gurney, and an MSC. The patient is represented by a wooden peg doll and must fit on a gurney of at least the same size and be placed at least 1.5 in. above the floor. An MSC of minimum length, width, and height of 3.00 in., 3.00 in., and 3.50 in. respectively, must be used. The max weight of the MSC must be declared at Tech Inspection and cannot change. EMTs will sit next to the patient with the cabinet either near the bulkhead or at the tail of Condor. The patient and EMTs must also not touch each other. After staging, there will be 5 minutes to complete 3 laps excluding the landing. The score is a function of the MSC weight and the time it takes to fly 3 laps as shown in Eq. 4. Payload weight (PWT) is the MSC weight, $(PWT/t)_{Max}$ is the highest ratio of payload weight to time for all teams, and $(PWT/t)_i$ is the individual score for Condor.

$$M2 = 1 + \frac{\left(\frac{PWT}{t}\right)_i}{\left(\frac{PWT}{t}\right)_{Max}} \quad (4)$$

3.1.5 Mission 3

M3 is an urban taxi flight carrying passengers only. Each passenger is represented by a wood peg doll 3.5 in. tall x 1.5 in. diameter. Passengers must sit on the floor of a compartment without contacting each other or the inner surface of the fuselage and must be accessible through hatches that are no more than 6 in. wide. After staging, there is a maximum 5-minute flight time within which Condor must be flown for as many laps as possible with landing not included in the flight time. Points are a function of the number of laps flown, passengers carried, and battery size as shown in Eq. 5. Here $Max_{NL*NP/BC}$ is the highest ratio of the number of laps (NL) times the number of passengers (NP) divided by the BC between all the teams. N represents the individual score for Condor.

$$M3 = 2 + \frac{N_{NL*NP/BC}}{Max_{NL*NP/BC}} \quad (5)$$

3.2 System and Subsystem Requirements

According to the 2024 AIAA DBF Rules, the major system and subsystem components were identified to develop Table 1 regarding major aircraft design constraints and subsystem components. Table 1 ensured that all rules and regulations were satisfied (S) or not satisfied (NS). Goose Works aimed to identify further

compliance with the design requirements regarding both the system and subsystem components outlined in Table 2 and 3 based on Table 1.

Table 1. System Constraint Fulfillments

Category	Subcategory	Label	Requirement	S/NS
Aircraft	General	AR-1	Wingspan cannot exceed 5 ft	S
		AR-2	Must take-off in 20 ft with ground contact points forward of the start line	S
		AR-3	The aircraft must use ground rolling takeoff and landing	S
		AR-4	All missions must be flown in the same configuration excluding payloads and optional passenger inserts	S
		AR-5	Aircraft must successfully land to receive a score	S
		AR-6	Aircraft has a maximum gross weight of 55 lb	S
	Electronics	AR-7	Propeller driven with an unmodified over-the-counter electric motor	S
		AR-8	Commercial brushed or brushless electric motors	S
		AR-9	Commercial propeller/blades that can be folded, clipped or painted	S
		AR-10	Commercial ducted fan units are allowed	S
		AR-11	Must have an externally accessible switch for radio control system that is not an arming plug	S
		AR-12	Radio control system must be separate from the propulsion system and arming system	S
	Propulsion	AR-13	Aircraft must be propeller driven and electrically powered	S
		AR-14	A propulsion system consists of one battery, one externally accessible arming fuse, and one or more motors and electronic speed controllers	S
		AR-15	The battery eliminator circuit must be disabled for the ESCs	S
		AR-16	Total stored energy cannot exceed 100 Wh	S
		AR-17	The maximum fuse current rating should equal battery discharge load up to 100 A	S
		AR-18	The battery C-rating should be at most 100 C	S
		AR-19	Battery can only be a single type. (Lithium-based or NiMh/NiCd)	S
		AR-20	Battery packs must be un-altered and commercially procured	S
Staging	General	SR-1	All components must be restrained according to Technical Inspection standards for taxi, take-off, flight, and landing	S
		SR-2	For all flight missions, the aircraft assembly and payload installation from flight configuration must not exceed 5 minutes	S
		SR-3	In parking configuration, with its landing gear in the upright orientation, the plane must fit inside a 2.5 ft wide parking spot	S
		SR-4	Cannot remove any components to fit into parking configuration other than fasteners or retention devices temporarily	S
Subsystems	Medical Payload	SSR-1	The gurney must be at minimum 5.5 x 1-11/16 in x 1.5 in	S
		SSR-2	The Medical Supply Cabinet must be at minimum 3x3x3.5 in	S
		SSR-3	Crew can see forward of the cockpit	S
	Crew	SSR-4	Crew and pilot must sit on a horizontal plane, side-by-side, without touching	S
		SSR-5	Crew is loaded in separate hatch forward of bulkhead	S
	Hatch Configuration	SSR-6	Passenger compartment is accessible from hinged hatches on the side of the aircraft	S
		SSR-7	Hatch opening cannot exceed 6 in. or extend past the fuselage vertical centerline, including the hinge connection	S
		SSR-8	Separate hatch or hinged canopy on side or top for crew members	S
	General	SSR-9	All passengers, EMTs, and crew members must not touch each other, only the tray and floor	S
		SSR-10	A single plane horizontal floor to carry EMTs, gurney with Patient, Medical Supply Cabinet and passengers all mounted upright and perpendicular to compartment floor	S
		SSR-11	A solid bulkhead must separate the crew and passenger compartments	S

Table 2. System/Subsystem Traceability

Label	System/Subsystem Description
AR-1	Wingspan is 5 ft
AR-2	Incorporated blown flaps and high thrust-to-weight ratio
AR-3	Tricycle landing gear and wing mounted props, no VTOL
AR-6	Heaviest aircraft configuration of 17.5 lbs
AR-8	Hobbystar C4130-435 KV motors
AR-9	APC 18x10E and 18x10EP propellers
AR-11	Externally accessible switch towards rear of fuselage
AR-12	Radio control system and propulsion system are powered by different batteries
AR-14	Propulsion system consists of one battery, one externally accessible arming fuse, 2 motors and 2 electronic speed controllers
AR-16	Total stored energy is 82.08 Wh
AR-17	Maximum fuse current rating is 100 A
AR-18	Battery has 75 C rating
AR-19	HCL-HVP Flight 22.8V-3600mAh-75C Battery
SR-1	Friction fit inserts attached with velcro, constrained by cabin, and blocked from forward/aft movement by wood dowels obstructing path to move
SR-3	Wings hinged at fuselage and tail width is 29 in
SR-4	Wing spar slides out of fuselage and rests inside aircraft wing in parking configuration
SSR-1	Gurney is 5.5 x 1-11/16 in x 1.5 in
SSR-2	Medical Supply Cabinet is 3 in x 3 in x 3.5 in
SSR-3	Developed a bubble canopy out of plastic to see ahead of cockpit
SSR-4	Crew rest on cabin floor friction fit into the plane separated by insert
SSR-5	Bubble canopy opens ahead of bulkhead separate from cabin cargo
SSR-6	Passenger compartment is on rear side of fuselage and hinged shut
SSR-7	M2 hatch opening is 6 in; M3 hatch opening for passengers is less than 6 in wide. Both hatch openings are to the left of the centerline and hinges do not cross
SSR-9	3D printed inserts separate passengers, EMTs, and crew members by 0.1 in. Inserts friction fit to sides of fuselage and touch the floor preventing components from touching the sides and top of the fuselage
SSR-10	Horizontal floor is the bottom of carbon fiber fuselage allowing all components to be perpendicular to compartment floor
SSR-11	3 pieces of plywood separate the main cabin from the crew member location

Table 3. High Scoring Requirements

Mission	Subsystem Requirements
M1	Parking configuration compliance High lift devices High thrust to weight ratio Stable controlled flight
M2	Mass optimized MTOW High thrust-to-weight ratio High airspeed Secure restraint system
M3	Volume optimized cargo bay Secured passenger area Low drag airframe High thrust-to weight ratio
GM	Low cargo load/unload time Low configuration change time

3.3 Sensitivity Analysis

The contribution of individual parameters to the score of each independent mission and the resultant score was assessed through a sensitivity analysis shown in Fig. 4. A system of equations was created to determine the effect of each mission parameter on the total score. The slope of the line for each parameter directly corresponds to the parameter's relative impact on the total score. The analysis showed that M3 had to be prioritized as it had the greatest impact on the overall score, followed by M2 and then GM. M1 was neglected since a constant score of 1 is rewarded for completion. To analyze how M3 could be maximized without neglecting other mission outcomes, independent parameters such as payload weight, time, number of laps, payload capacity, and BC were also analyzed. From the parameter analysis shown in Fig. 4, decreasing the time needed to complete M2 and GM positively impacted the score and increasing payload weight while decreasing the BC was also favorable.

The slopes of the parameters showed that the optimal aircraft should be able to attain high speeds. This resulted in a decrease of M2 time and an increase in the number of laps for M3. Ideally, Condor also carried the heaviest payload with the smallest BC allowed. This conveyed that Goose Works had to focus on having a strong structural skeleton, a high wing area, and a combination of AR, top speed, efficiency, and reduced aircraft weight. In doing so, MTOW would have a higher mass fraction and would satisfy AR-2 and AR-6 requirements while maintaining a high cruise velocity. It would also ensure the minimized BC provided enough power to land each mission, satisfying AR-5.

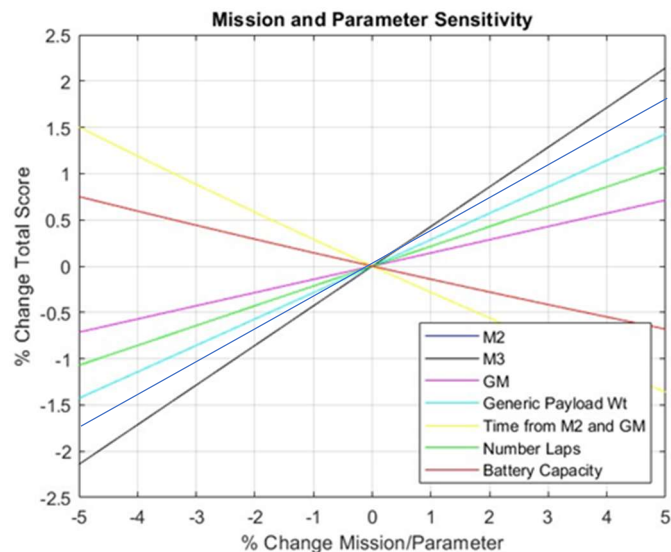


Figure 4. Sensitivity Analysis

3.4 Aircraft Conceptual Design Selection

During the conceptual design process, Goose Works used decision matrices to quantify the process of choosing an aircraft configuration based on the team's prior knowledge while referencing Raymer's Aircraft Design [2]. All qualitative selections pertaining to the aircraft type, landing gear, engine, fuselage, wing, and tail configuration were made using this approach with specific sizing conducted later. The decision matrices were developed to reflect options for each aircraft parameter and weighted. Table 4 outlines the reasoning for each criterion.

All weights ranged from 1-5, where 5 meant the criteria is strongly satisfied and 1 is weakly satisfied. When ranking each parameter, the design rules and constraints for the 2023-2024 DBF competition were also

considered. For each parameter, the rank is multiplied by the weight of the given criteria and summed. The parameter that had the highest score was determined to be the best option from that decision matrix.

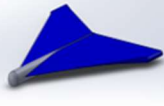
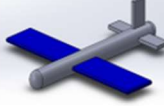
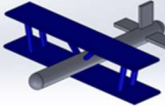
Table 4. Criteria Reasoning

Criteria	Reasoning
Safe Flight Characteristics	Ensure all passengers remain in their place during flight missions
Manufacturability	Save time and maintain reliable quality and precision
Folding Capability	To satisfy parking configuration requirement
Weight	Minimize takeoff distance and increase flight time
Take Off Distance	Necessary to meet 20 ft constraint to fly each mission
Drag	Reduce to improve flight time and performance
Volumetric Efficiency	Increase overall passenger capacity
Stability and Control	Improve flight performance and ensure safe flight
Structures	Increase g-load capabilities and reliability
Propeller Clearance	Prevent prop strike on ground
Hatch Placement	Reduce loading time and optimize passenger placement
Take Off/STOL Capability	High lift generation to reduce takeoff distance
Crash Resistance	Ease of repairability and robust structures makes repair easier
Efficiency	Number, size and type of motors affects battery size and flight time
Stall Recovery	Stall conditions are likely to occur in Kansas winds, size tail correctly
Durability	Dynamic loading during landing requires rigid structures

3.4.1 Aerodynamic Surface Configuration

A variety of aerodynamic configurations such as flying wing, conventional, biplane, and 3-surface designs were considered as shown in Table 5. The biplane was a poor choice because of the complexities involved with folding two sets of wings, high drag, and manufacturability. The flying wing and 3 surface had similar scores to the conventional aircraft, but they had shortcomings in terms of safe flight characteristics and takeoff distance rather than manufacturability, weight, and drag. Three-surface almost won because it could fulfill the 2.5 ft parking configuration requirement while keeping a high lift generating area.

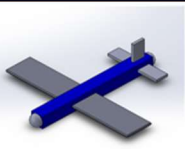
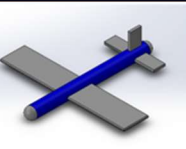
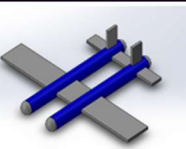
Table 5. Aircraft Configuration Selection

		Score			
					
Criteria	Weight	Flying Wing	Conventional	Biplane	3 Surface
Safe Flight Characteristics	5	2	5	5	5
Manufacturability	2	5	4	2	3
Folding Capability	4	5	5	1	5
Weight	3	4	4	4	3
Take off distance	5	1	5	5	5
Drag	4	4	3	1	3
Total		73	77	49	72
Rank		3	1	4	2

3.4.2 Fuselage Configuration

A blended body fuselage scored highest due to its volumetric efficiency and drag characteristics; however, after deliberation between the close score of blended body and rectangular fuselage, the team opted for the rectangular fuselage due to manufacturability and superior lift generation for a fixed span. In a fixed span application, the blended body produces considerably less lift as it has less pure airfoil span. Therefore, there is less efficient lift generation as a higher percentage of the allowed span as compared to a conventional configuration. For these reasons, a conventional design was ultimately selected. Table 6 documents the complete decision matrix for the fuselage configuration.

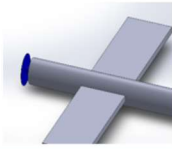
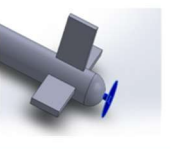
Table 6. Fuselage Configuration

		Score			
					
Criteria	Weight	Rectangular	Blended Body	Circular	Double Fuselage
Drag	4	1	5	4	2
Manufacturability	3	5	3	3	1
Volumetric Efficiency	4	5	3	2	1
Weight	4	4	4	4	0
Total		55	57	49	15
Rank		2	1	3	4

3.4.3 Engine Configuration

Single tractor, single pusher, double, and double wingtip configurations were considered as shown in Table 7. A double engine configuration was chosen for its improved aerodynamic efficiency and contributions to takeoff distance. By mounting the propellers on the wings, the higher velocity airflow behind the propeller results in a higher Re and lift generated by the airfoil span behind the propeller. As a result, the short takeoff capability is increased through the effect of blown flaps [2]. The single tractor and single pusher were discarded due to concerns regarding P-factor and angular impulse of increasingly large propellers at sufficient acceleration rates. These factors were theorized to lead to instability at and during takeoff which would contribute to poor real world short take-off and landing (STOL) capabilities. The pusher configuration was eliminated due to the increasing complexity in designing a pusher with an adequate center of gravity (CG) with increasing motor sizing.

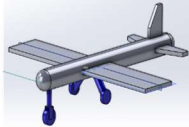
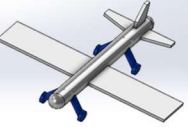
Table 7. Engine Configuration

		Score			
					
Criteria	Weight	Single	Double	Double Wingtip	Single Pusher
STOL Capabilities	4	3	5	4	2
Manufacturability	3	5	4	1	3
Crash Resistance	1	1	3	1	5
Efficiency	3	2	2	5	2
Total		34	41	35	28
Rank		3	1	2	4

3.4.4 Landing gear

Goose Works opted for tricycle landing gear that marginally beat taildragger as seen in Table 8 for the V1 and V2 aircraft due to the perceived take-off distance effect. Although the score was close, tricycle gear was preferred for ground handling, landing, and reduced distance between the nose and main gear to better fulfill AR-2. A quadracycle and skid were considered but ultimately deemed unfit due to added complexity, drag, weight, and low ranked scores.

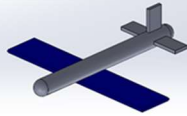
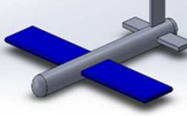
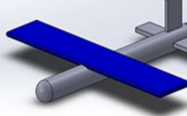
Table 8. Landing Gear Configuration

		Score			
					
Criteria	Weight	Tricycle	Tail dragger	Quadracycle	Skid
Manufacturability	3	3	4	1	0
Durability	5	3	5	1	5
Take off distance	5	5	1	1	0
Drag	3	3	5	1	0
Total		58	57	16	25
Rank		1	2	4	3

3.4.5 Wing Placement

Low, middle, and high wing were considered as shown in Table 9. A high wing was selected due to its superior stability characteristics and propeller clearance which reduced the potential for ground strikes. In addition, the fuselage acts as a bounding surface to the high-pressure region of the wing and limits spanwise flow across the fuselage as would occur with a low wing configuration. Although this moves the wing further out of ground effects, which may impede the impact of ground effect, the airfoil is still well within the often cited 20% wingspan limit. Mid-wing was discarded due to perceived challenges with hatch manufacturing along the side of the aircraft and the decrease in cargo bay volumetric efficiency from the spar pass-through. Low wing was discarded due to unfavorable propeller clearance, stability, and control.

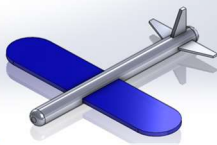
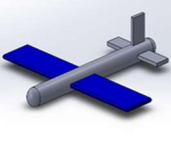
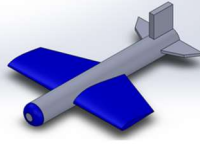
Table 9. Wing Placement

		Score		
				
Criteria	Weight	Low	Middle	High
Stability and control	2			
Manufacturability	3	5	4	5
Takeoff Capability	5	5	4	3
Structures	3	5	5	5
Prop clearance	4	1	3	5
Hatch Placement and Size Consideration	5	5	1	3
Total		84	64	80
Rank		1	3	2

3.4.6 Wing Shape Configuration

Wing shapes such as elliptical, rectangular, tapered, and swept were considered in Table 10. Take off capability, stability, and control were prioritized. Manufacturability and drag were other evaluated criteria. Tapered wing scored the highest due to a balance between manufacturability and an approximate elliptical lift distribution resulting in greater cruise efficiency. While the rectangular wing was easier to manufacture, it was less favorable in terms of cruise efficiency as a function of lift distribution and the Oswald efficiency factor.

Table 10. Wing Shape Configuration

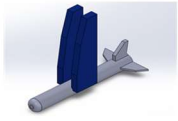
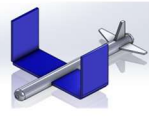
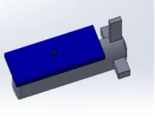
		Score			
					
Criteria	Weight	Elliptical Wing	Rectangular Wing	Tapered Wing	Swept Wing
Stability and control	4	2	4	3	1
Manufacturability	3	1	5	5	3
Drag	2	5	2	4	1
Takeoff Capability	5	5	2	4	2
Total		46	45	55	25
Rank		2	3	1	4

3.4.7 Wing Folding Position

Wing folding was considered in Table 11 to highlight the need to fit in a 2.5 ft parking spot. The team initially opted for a half-fold in the wings; however, Condor was made to have flaps as the design progressed. Given the flap span, the flaps would've had to have been folded as well, complicating the system. So

instead, the wings were folded at the fuselage with the reinforcement of a telescopic spar. Therefore, hinging at the fuselage was preferred in subsequent configuration comparisons and decisions.

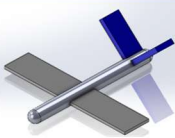
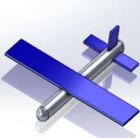
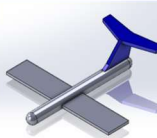
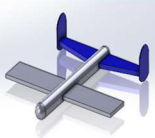
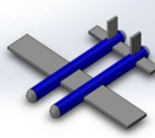
Table 11. Wing Folding Position

		Score			
					
Criteria	Weight	Full Fold	Half Fold	Swept Fold	Swing Wing
Safe Flight Characteristics	5		5	5	5
Manufacturability	2	5	4	2	3
Folding Capability	4	5	5	1	5
Weight	3	4	4	4	3
Take off distance	5	1	5	5	5
Drag	4	4	3	1	3
	Total	73	77	49	72
	Rank	3	1	4	2

3.4.8 Tail Configuration

V-tail, conventional, T-tail, and H-tail were considered; however, the best option was determined to be a T-Tail as shown in Table 12. Although more complex to manufacture, the T-tail is mounted higher, and the wake produced by the wings and propeller thrust cone is avoided, improving the stability and control characteristics at high AOA. The conventional tail experiences less drag but the stall recovery characteristics are less favorable than the T-tail design due to potential tail plane stall. V-Tail almost broke even with the T-tail; though the latter had more favorable drag and stall recovery characteristics.

Table 12. Tail Configuration

		Score				
						
Criteria	Weight	V Tail	Conventional	T Tail	H Tail	Double Boom Tail
Drag	2	4	4	5	3	1
Manufacturability	4	5	4	3	3	1
Stall Recovery	5	4	3	5	3	1
Weight	5	5	4	3	3	1
	Total	73	59	62	48	16
	Rank	1	3	2	4	5

3.5 Final Conceptual Design Configuration

The final conceptual design for Condor is depicted in Fig. 5. Condor is a high wing, twin propeller, tricycle gear aircraft featuring a T-tail and rectangular shaped streamlined fuselage. The wings have endplates and blown flaps to assist with the 20 ft take-off distance. To accommodate the crew, a clear windshield was utilized to satisfy constraint SSR-3. Hatches for M2 and M3 comply with constraints SSR-6 to SSR-8 via loading around the CG of the aircraft and rear side of the fuselage. The fuselage has a cargo volume to carry 64 passengers and adhere to the minimum dimensions of the MSC.



Figure 5. Final Conceptual Design of Condor

4 Preliminary Design

Optimization studies were conducted to determine aircraft parameters while simultaneously assessing trade studies, take-off performance, control, and static and dynamic stability. The research and data collected provided Goose Works with quantitative data to make design decisions and changes for the detailed design stage. Environmental uncertainties were also considered due to the variation in climate conditions between Buffalo, New York and Wichita, Kansas.

4.1 Design Methodology

Goose Works used a three-step iterative design process that started with a sensitivity analysis to inform which performance metrics highly impact the competition score as shown in Fig. 6. Following the sensitivity analysis, the team determined the general configuration via decision matrices (Section 3.4). An optimization code followed the conceptual design which evaluated a plethora of aircraft configurations based on cruise velocity, payload weight, and battery size. UB DBF created the initial aircraft performance parameters using the output of the optimization calculator. In addition, propeller and motor sizing selection were conducted to ensure the propulsion system met or exceeded the expected performance. An in-depth analysis was performed with XFLR5, ANSYS, Athena Vortex Lattice (AVL), and SolidWorks to determine the structural and performance characteristics of the aircraft. V1 was developed using Computer-Aided Design (CAD) software, manufactured, and tested; an FPV (First Person View) system was integrated into the plane to collect in-flight data. The initial test flight concluded that changes were needed to satisfy AR-2 and SR-2. After test flight 1, a design review was conducted to inform design changes to be made for Version 2 (V2) with changes briefly discussed in Section 3.5. V2 was modeled, manufactured, and then tested to inform the team of performance characteristics and minor changes to be made to the Version 3 (V3) aircraft. V2 will serve as a backup aircraft for competition if V3 is damaged and unable to fly again.

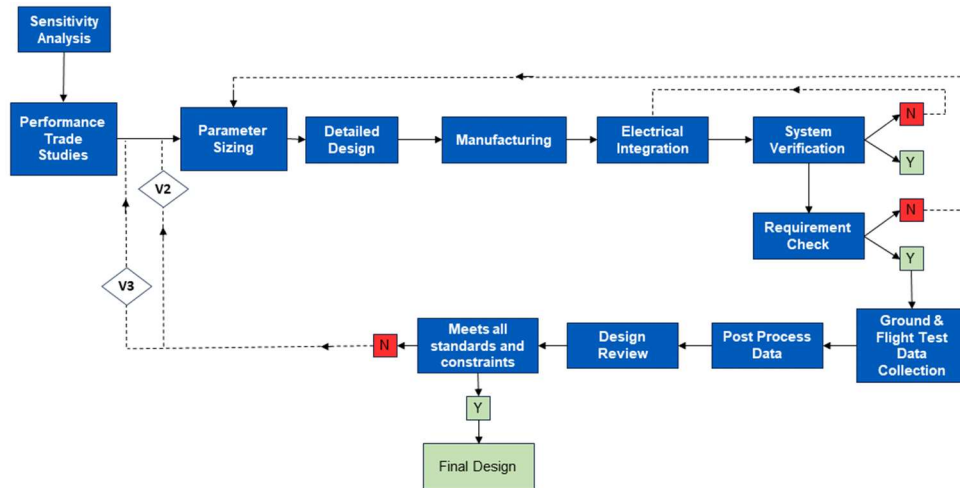


Figure 6. Design Methodology

4.2 Optimization Study

An automated “trial and error” approach was implemented in MATLAB to compare multiple aircraft configurations based on velocity, payload weight and battery size. Written by several team members, the script took vector inputs of possible aircraft design variables and used the MATLAB function “combinations” to create a matrix that was inclusive of all unique combinations of the input vectors. The program tested each case and calculated the respective individual scores. A series of logical filters were employed to delete aircraft configurations that did not meet mission constraints such as: the maximum 20 ft take-off distance, 3-laps minimum flight time and maximum 5 ft wingspan. M1 was excluded.

The optimization calculator had static inputs and assumptions and varying inputs that changed with every loop iteration. The varying inputs were M2 payload weight, M3 payload weight, M2 velocity, M3 velocity, wing area (S) and battery capacity (BC) and are denoted in the following equations with an “i” subscript. Propulsion parameters were not included as inputs as Goose Works determined an optimal propulsion configuration through manual iteration and the use of the propulsion calculator detailed in Section 4.3.4.2. The selected configuration of twin 17x10E APC propellers were represented in the optimization calculator with a custom 3D curve fit for velocity, power, and thrust.

The total weight of the aircraft was assessed using Eq. 6, where Plane Wt. was the assumed empty weight of the aircraft, PWT was the payload weight assumed as a maximum of 40% of Total Wt. and was a variable input as indexed from the plane configuration vector; Δ Foil Wt. and Battery Weight were defined by equations below where Battery Weight is also an indexed input variable.

$$Total\ Wt. = Plane\ Wt. + PWT + Battery\ Wt. \pm \Delta\ Foil\ Wt. \quad (6)$$

The difference in foil weight was expressed by Eq. 7, where $S_{w,a}$ was an assumed baseline wing area taken from V1, *subscript* was the wing area as input from the combinations matrix and b was wingspan. The wingspan was assumed to be the maximum allowed 5 ft from prior iterations, and 0.1 represents the 10%

thickness of the chosen Selig-Donovan (SD) 7032 airfoil. This equation represents a simplified volume calculation of a rectangular prism multiplied by the density of Extruded Polystyrene Insulation (XPS) foam which is used to construct the wings. The difference in foil weight was used to correct the known weight of the currently designed plane to better reflect a smaller or larger plane as evaluated by the calculator.

$$\Delta \text{ Foil Wt.} = 0.1\rho_{xps}(S_{w,a} - S_{w,i})\frac{S_{w,i}}{b} \quad (7)$$

Battery weight is expressed in Eq. 8, where BC was expressed in W-h and the energy density was taken for SMC flight packs.

$$\text{Battery Wt.} = (\text{BC})(E_{\text{density}})(g) \quad (8)$$

If the total weight of the plane approximately equaled the lift force in straight and level flight, Eq. 9 was solved to calculate the lift coefficient required at the aircraft's cruise velocity where *beta* is the bank angle.

$$C_L = \frac{2\text{Total Wt.} \cos(\beta)}{\rho_{\text{air}} S_{w,i} V_{\text{cruise}}^2} \quad (9)$$

$$C_D = C_{D,0} + \frac{C_L^2}{\pi e AR} \quad (10)$$

The total drag coefficient was then calculated with Eq. 10 given an assumed zero-lift drag coefficient, $C_{D,0}$, and Oswald efficiency factor, *e*. AR was calculated with Eq. 12.

$$AR = \frac{b^2}{S_{w,i}} \quad (12)$$

Next, the force of drag was calculated with Eq. 13.

$$F_d = \frac{1}{2} \rho_{\text{air}}(S_{w,i})(V_{\text{cruise}}^2)(C_D) \quad (13)$$

The calculation considered 2 forces of drag; one at an assumed bank angle of 0° for straight and level flight and the other at 60° of bank angle assumed for turning. The two forces of drag were used to analyze the energy draw of the aircraft during cruise and determine how many laps the configuration can fly on its BC. The number of energy-limited laps was limited to a maximum as determined by the cruise velocity. The energy draw of a single lap is presented by Eq. 14 where D_s is the straight distance and D_T is the turning distance.

$$\frac{E_{\text{cruise}}}{\text{Lap}} = \frac{2(f_{\text{power vs thrust}})(F_d)\left(\frac{D_s}{V_{\text{cruise}}}\right) + 2(f_{\text{power vs thrust}})(F_{dtu})\left(\frac{D_T}{V_{\text{cruise}}}\right)}{\text{Propulsive Efficiency}} \quad (14)$$

It was assumed that in straight and level flight, the force of drag was equal to the propulsive thrust. As such, the power requirement of straight and level flight was calculated using the 3D power, thrust, and velocity curve fit for an APC 17x10E propeller. The straight and turning distances were estimated from the provided figure in the 2024 DBF rules. The laps that can be flown by the aircraft configuration are expressed by Eq. 15. The scoring for each iteration was evaluated with the provided equations from the rules excluding the normalizing denominator since a maximum value could not be found on a per iteration basis; the normalized value was found after all iterations were complete.

$$Laps = \frac{Battery\ Size * 3600 - (E_{Takeoff})}{\frac{E_{cruise}}{Lap}} < \frac{300\ s}{\frac{Lap\ Distance}{V_{cruise}}} \quad (15)$$

Once generated, the configuration was checked to ensure that key operational parameters were met. The available thrust always had to be greater than the required thrust, the number of laps had to be greater than 3 for M2 and the configuration had to take off within 17 ft. To calculate takeoff distance, Goose Works first attempted to consult conventional references such as Raymer [2]. It was quickly realized that a conventional mathematical approach included too many assumptions and simplifications to apply to such a critical part of performance for competition. Therefore, a custom takeoff distance calculator was created using numerical methods. A force sum was time-stepped along small-time intervals to integrate from forces to velocity to distance. By doing so, Goose Works was able to incorporate factors like dynamic apparent AOA, dynamic thrust, rotational kinetics, and thrust vectoring into the takeoff distance consideration. The first step was to determine the instantaneous direction of the apparent airspeed vector to find the components of lift and drag in the -x and +y directions. This was further complicated by the two wing-mounted propellers included in the proposed design. As such, the apparent airspeed vector at any given time could be expressed as Eq. 16.

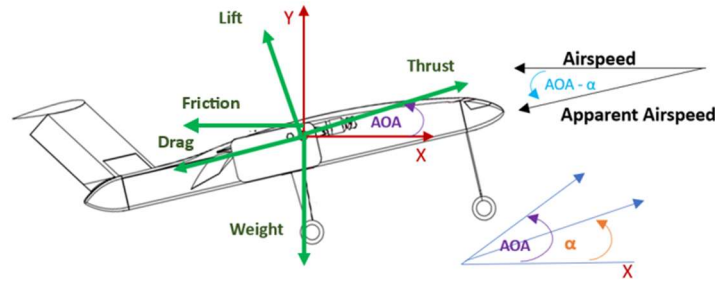


Figure 7: Takeoff Distance Kinetic Sum

$$\vec{V_{apparent}} = \vec{V_{prop\ wash}} + \vec{V_{air}} \quad (16)$$

As the motor nacelles were mounted at zero degrees relative to the chord line of the wing, the instantaneous direction of the prop wash vector relative to the wing will be zero degrees; however, determining the average exit velocity of the propeller wash proved to be more difficult. A simplified momentum theory was employed and is presented in Eq. 17.

$$V_{outlet} = \sqrt{\frac{2\ (Thrust)}{\rho_{air}(Prop\ Area)} + V_{inlet}^2} \quad (17)$$

Therefore, the apparent velocity vector was calculated at each time step and the angle denoted as alpha was found with Eq. 18.

$$\alpha = \tan^{-1} \frac{\sin(AOA) (V_{outlet})}{\cos(AOA) (V_{outlet}) + V_{air}} \quad (18)$$

The above equation utilized the geometric AOA with respect to the airspeed vector while the apparent AOA was calculated by geometric AOA minus alpha. With alpha known, the lift force and drag force were

expressed discretely in the x and y directions. The propulsive thrust was calculated by a custom curve fit equation of thrust vs. inlet airspeed. This was modified by a dynamic “rev factor,” which accounted for the RPM lag of a full-throttle input. The forces in the x and y directions are shown in Eq. 19 and 20, where μ is the equivalent frictional coefficient for a rolling rubber wheel on asphalt [2].

$$F_x = \text{Thrust}(\cos(\text{AOA})) - \mu(F_d(\cos(\alpha) + F_l(\sin(\alpha)) + (W + F_d(\sin(\alpha)) - F_l(\cos(\alpha) - \text{Thrust}(\sin(\text{AOA})))) \quad (19)$$

$$F_y = W + (F_d(\sin(\alpha) - F_l(\cos(\alpha)) - \text{Thrust}(\sin(\text{AOA}))) \quad (20)$$

At each time step, the acceleration, velocity, and distance traveled step were calculated with Eq. 21-23, where the acceleration is a and Δt is the time step.

$$V_{total} = V_{previous} + a\Delta t \quad (21)$$

$$a = \frac{F_x}{mass} \quad (22)$$

$$Distance_{total} = Distance_{previous} + V\Delta t \quad (23)$$

The calculator was set up such that it dynamically changed geometric AOA from a low AOA ground run configuration to a high AOA takeoff configuration at a given distance. The calculator ended when the force sum in the y direction became negative (indicating the point of takeoff). The output distance was then checked against the takeoff limit of 17 ft. If a configuration failed this or any other check, it was deleted from the results matrix. Once each configuration was verified, the scores for each mission were stored in appropriate vectors. After the evaluation of each configuration, the scoring of configurations was determined based on a scaled sum of M2 and M3 scores according to Eq. 24.

$$Scaled\ Sum = \frac{M2(M2_{iterative})}{Max_{M2Score}} + \frac{M3(M2_{iterative})}{Max_{M3Score}} \quad (24)$$

The variables as ordered by the scaled sum are presented in Fig. 8 and output into a table of values that consisted of configurations that score in the 98th percentile or greater. The last entries of this table were considered as “optimal” aircraft configurations. Once output, the inputs were shifted to better reflect parameter ranges centered around the previous optimal outputs to “zoom” in on an increasingly optimal solution. With an optimal set or range of parameters output, Goose Works was able to use more traditional approaches to design components and select configurations that fit within the output parameters as shown in Fig. 8.

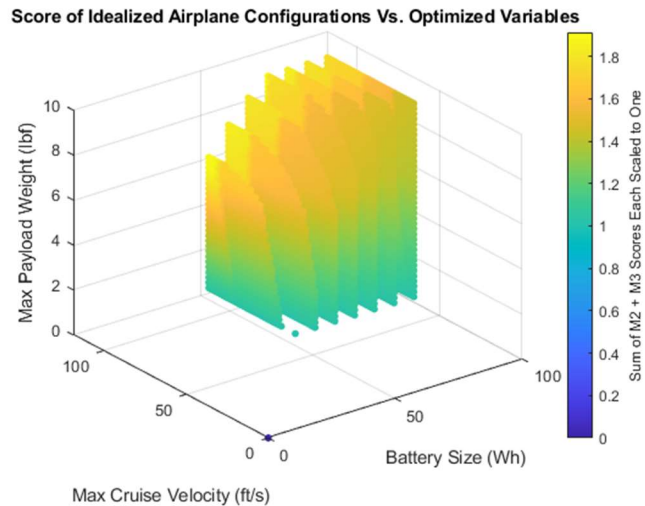


Figure 8. Optimization Study

4.3 Sizing Trade Studies

4.3.1 Airfoil Selection

Airfoil selection is important to determine flight characteristics and mission scores.

An airfoil list was generated from airfoils used by past UB DBF teams, as well as the

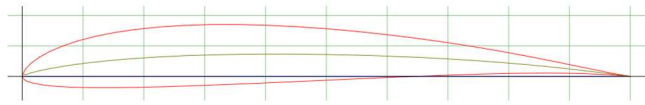


Figure 9. SD 7032 Airfoil [7]

top 3 winning teams of the previous DBF competitions that took place in Wichita, Kansas. The following airfoils were selected as candidates for this year's DBF: Clark Y, NACA 2412, NACA 4408, SD 7032, and SD 7062. To determine the best airfoil for all three flight missions, the team analyzed the maximum lift coefficients, minimum drag coefficients, critical AOA, maximum lift-to-drag ratio, AOA at maximum lift-to-drag, and zero lift angle. The main criteria responsible for airfoil selection were the 20 ft take-off distance and the estimated 22 miles per hour (mph) takeoff velocity (Section 3.1). As a result, the airfoil must have a high Lift-to-Drag ratio over a range of AOA for sufficient lift at take-off and cruise.

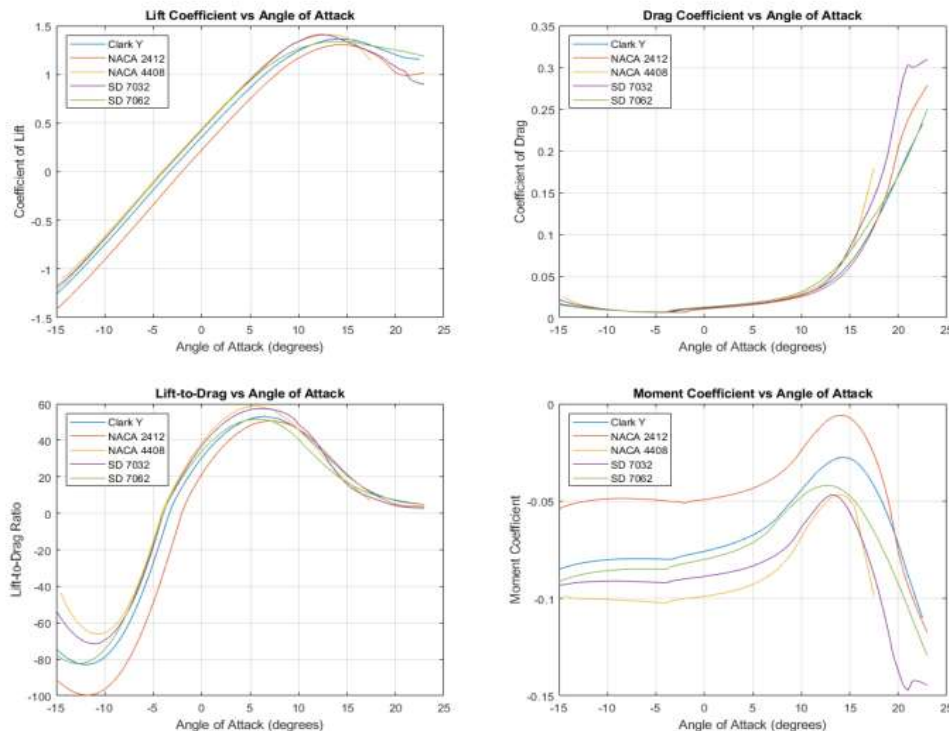


Figure 10. Airfoil Coefficients

Airfoil analysis was performed with open-source airfoil analysis tools XFLR5 and ANSYS. The average velocity, V_{avg} , of Condor is 100 ft/s with an average chord of 16 in. used as the characteristic dimension, L . At an altitude of 13000 ft, the elevation of Wichita, Kansas above sea level, the air density (ρ) is 0.00229 slug/ft³ and air dynamic viscosity (μ) is 3.1384E-7 slugs/ft-s. Therefore, the Reynolds number (Re) was calculated with Eq. 25: The Re was around 730,000 due to the high speeds achieved by Condor.

$$Re = \frac{\rho V_{avg} L}{\mu} \quad (25)$$

Based on airfoil analysis, the Selig-Donovan (SD) 7032 airfoil in Fig. 9 was selected as it met the design criteria better than the other airfoils. From the lift coefficient (C_l) vs AOA plot in Fig. 10, it can be seen that the SD 7032 and the NACA 4408 lead the other airfoils in maximum lift coefficient over relevant AOA, as well as maximum lift-to-drag ratio as seen in the Lift-to-Drag plot. These values are important as a high lift coefficient is needed during take-off to maximize the number of passengers, and the mission scores.

4.3.2.1 Tail Design and Sizing

The tail of the aircraft was sized with the tail volume coefficient (TVC) methods specified by Eq. 26 and 27, where C_{VT} and C_{HT} are the vertical and horizontal TVC, S_{VT} and S_{HT} are the vertical and horizontal tail areas, c_w is the wing chord. b_w is the wingspan, S_w is the wing reference area, L_{VT} is the distance from the aerodynamic center (AC) of the vertical stabilizer to the AC of the wing airfoil while L_{HT} is the same for the horizontal stabilizer.

$$C_{VT} = \frac{S_{VT} L_{VT}}{S_w b_w} \quad (26)$$

$$C_{HT} = \frac{S_{HT} L_{HT}}{S_w c_w} \quad (27)$$

Typical values of the C_{HT} for a homebuilt aircraft are from up to 0.5 to 0.8 for a general aviation twin-engine aircraft [2]. Typical C_{VT} ranges from 0.04 to 0.07 respectively. Other design goals like minimizing empennage weight and ensuring sufficient sizing to allow for rotation during takeoff were considered. A MATLAB script was developed to minimize empennage mass while maximizing tail volume. The script considered distance from the C.G. as a percent of fuselage length where any percentage above 50% was considered to utilize a tail boom at an assumed weight per unit length. With the assumed weight of the carbon tube utilized for the spar, it was found that the optimal fuselage percentage for the tail boom length was 55%. This indicated a small tail boom mounted to the fuselage. To simplify manufacturing and design, a tail boom length of 50% reflected an empennage mounted to the end of a fuselage with wings mounted at the C.G. As takeoff distance increased nonlinearly with additional weight, the total tail size and weight needed to be minimized. To do this, XFLR5 was used to assess the required elevator sizing to rotate the plane to a high AOA takeoff configuration at a takeoff airspeed of 22.4 mph. XFLR5 simulation resulted in a required horizontal C_{HT} of 0.60 which corresponded to a horizontal tail area of 287 in² assuming an elevator sized at 44% of the horizontal area deflected at 40 degrees. As this is within the referenced range from Raymer, the horizontal tail area sizing was based on this constraint and was not made larger to minimize the weight of the tail. UB DBF historical data points to this being more than sufficient during flight with last year's plane sporting a C_{HT} of 0.6. The C_{VT} was sized to 0.055 which is within the referenced acceptable range. Furthermore, yaw stability is aided by the differential thrust of the wing-mounted

propellers. Following testing of the V1 prototype, balance tabs were added to the elevator at a total horizontal area percentage of 8.8% [8]. The balance tabs were added to primarily reduce the required force from the servo and secondarily to assist in rotating the plane during its transition from ground run to takeoff orientations.

4.3.2.2 Surface Design and Sizing

To estimate aileron sizing, the graph in Fig. 11 from Raymer [2] was used. Aileron span can typically range from 90% of the wingspan to approximately 30% depending on chord percentage. Slotted flaps were incorporated into the design to achieve $C_{l,max}$ at take-off that would otherwise not be achievable with only ailerons. According to Raymer, the flap span to achieve $C_{l,max}$ should be as large as possible. Therefore, a minimum aileron sizing should be determined to maximize the effective flap span. Referencing previous Goose Works teams, it was decided that an aileron span of 33% with a corresponding chord percentage of 35% would be sufficient while saving wingspan for flap installation. As aileron span and sizing is dependent upon flap performance ANSYS was used to perform an iterative flap analysis. By changing the flap size from 32-38% chord it was noticeable that lift

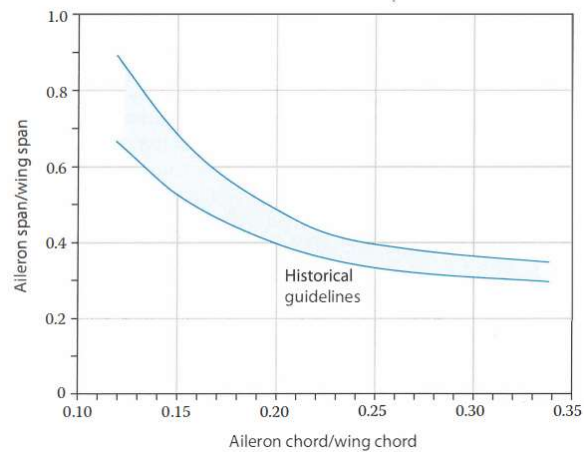


Figure 11. Control Surface Sizing

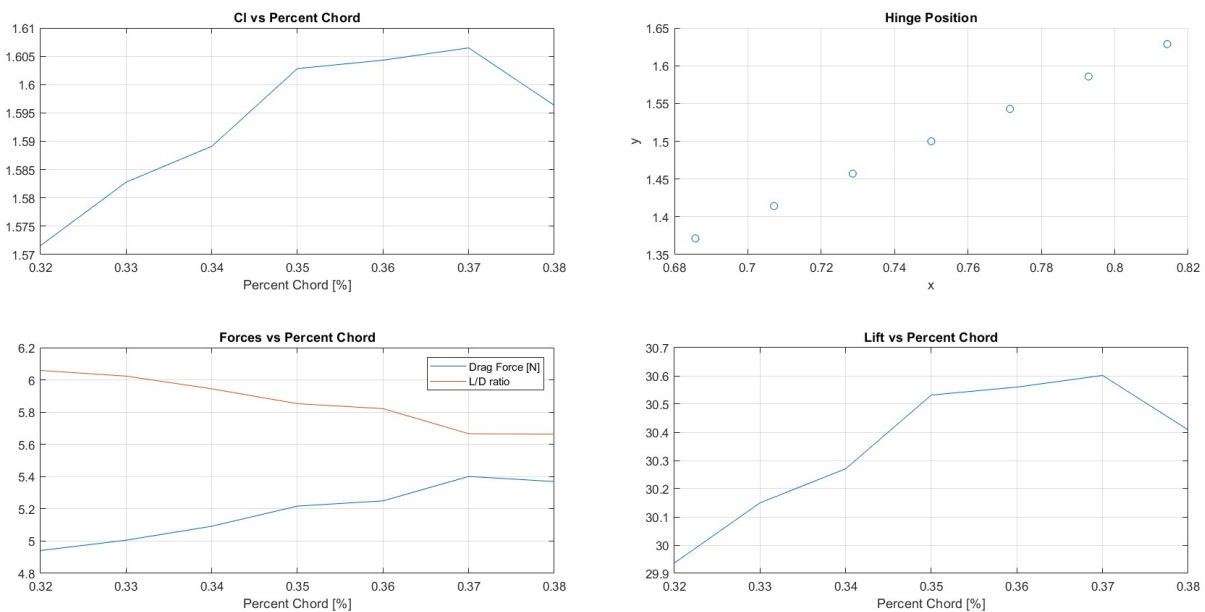


Figure 12. Hinge Position and Aerodynamic Effect of Flap Sizing

drastically increased from 32-35% and the effect tapered off around 38%. Raymer states that 40% flap yields the highest C_l changes, however, due to sizing constraints for hinges, and the experimental data in Fig. 11, a 37% flap at a target 37-degree deflection angle was chosen. The data outlined in Fig.12 was calculated with a fixed hinge location relative to the size of the flap and take-off conditions at an assumed 10 deg AOA and apparent airspeed of 32.8 ft/s. Overall, a 66% span and 37% chord flap were chosen for the design based on the previously mentioned data and Section 4.4.3 stability analysis.

4.3.3 Landing Gear Design and Sizing

The tricycle landing gear was designed according to Raymer's [2] conventions with all parameters within the standard conventions included therein. The nose gear is mounted such that 8% of the weight of the aircraft rests on the nose gear. Fig. 13 is the geometric construction of the landing gear with 6.3 in. of propeller clearance, a tip-back angle of 10 deg, and overturn angle of 28 deg.

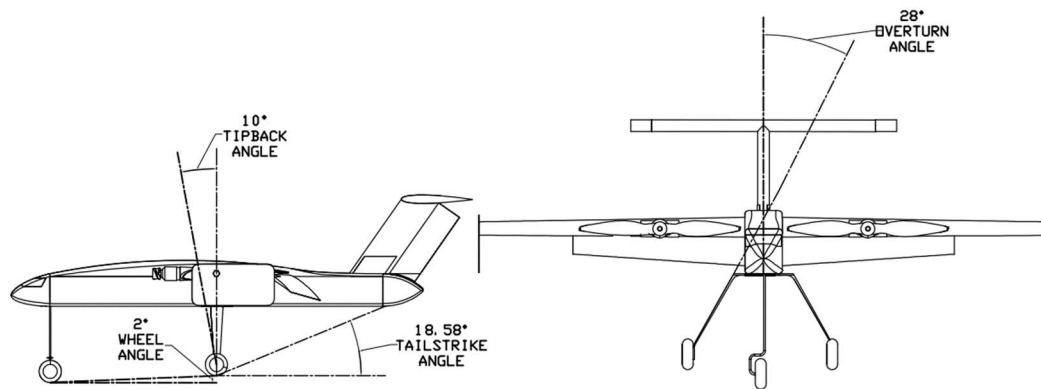


Figure 13. Hinge Position and Aerodynamic Effect of Flap Sizing

4.3.4 Propulsion and Avionics Design and Sizing

The aim of propulsion and avionics design and sizing was to design an electric, propeller-driven propulsion system for a wing-mounted motor configuration and a servo-actuated control system that fulfilled requirements AR-7 through AR-20. Moreover, the propulsion system had to prioritize static thrust for competitive scoring as arbitrated by the optimization study.

4.3.4.1 Propulsion Battery Sizing and Selection

Only LiPo batteries were considered because of their superior energy density and discharge and were evaluated on maximum supply power, capacity, and C-rating.

$$\text{Power} = (\text{Voltage})(\text{Current}) \quad (28)$$

Maximum battery supply power is relevant as it is analogous to the takeoff power limit used to constrain thrust. It can be computed using Eq. 28 and is limited by the maximum current rating of the arming fuse defined by AR-17. Instead of 100 A, 135 A was used to calculate the maximum supply power since MAXI fuses can sustain 135% of their nominal current rating for at least 60 seconds [9] and since the FOS was deemed acceptable.

The SMC HCL-HVP Flight 22.8V-3600mAh-75C battery (Table 13) was chosen to complete M1, M2, and M3. UB DBF opted to push the fuse past 100 A with a 22.8 V pack because the Speedybee F405 Fixed Wing, part of Goose Works' FPV system, isn't compatible with higher voltages. The flight pack can safely discharge up to 270 A and has a capacity of 82.08 W-h.

Table 13. Preliminary Propulsion Battery Selection Specifications

Battery	Nominal Voltage (V)	Maximum Supply Power (W)	Capacity (Wh)
SMC HCL-HVP Flight 22.8V-3600mAh-75C	22.8 V	3078	82.08

4.3.4.2 Propeller and Motor Sizing and Selection

Attaining the static thrust target of 20.5 lbf to satisfy AR-2, the takeoff distance requirement at MTOW was emphasized when sizing propellers. To satisfy AR-9, suitable propellers were found by fitting data for commercially procurable counter-rotating APC propellers and matching thrust with mechanical power estimated from half of the propulsive battery power listed in Table 14. 16x10Es and 17x10Es were favored for high efficiencies and thrust potential.

From there, MATLAB code following Brushless Direct Current (BLDC) motor design theory [10], APC performance data [11], and the principle of energy conservation was devised to iteratively solve for static thrust, RPM, and power draw given motor, propeller, ESC, and battery specifications to identify prospective motors. The solver was cross-validated with historic UB DBF propulsion test data and eCalc [12]. Table 14 summarizes the convergent results for the chosen configurations.

Table 14. Preliminary Static Thrust Prediction

Motors	ESCs	Propellers	Thrust (lbf)	RPM	Power (W)	Current (A)
HobbyStar 4130 (5065) 435 Kv	Onyx 80 A	APC 16x10E	21.65	7301	2855.1	128.9
HobbyStar 4130 (5065) 435 Kv	Onyx 80 A	APC 17x10E	22.95	6790	3147.8	145.3

There should be enough headroom to accommodate 17x10Es, since HobbyStar 4130 (5065) 435 Kv motors are rated for 73 A, and the 100 A fuse time-delay characteristics are projected to allow for a current draw past 135 A.

Observing Fig. 14, both propellers can exceed the target M2 cruise velocity of 120.4 ft/s, and 16x10Es are marginally more efficient than 17x10Es at M3 cruise. Thus, after evaluating takeoff and in-flight performance, the 17x10E configuration—subject to physical testing—was tentatively favored for M1, M2, and M3.

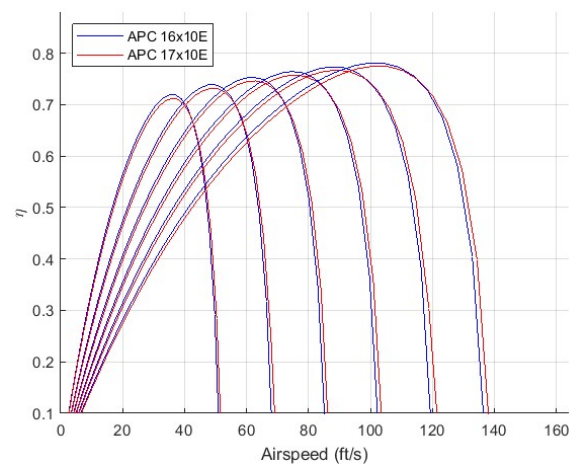


Figure 14. Propeller Efficiency at Select Ranges of RPM in Operation

4.3.4.3 Avionics Sizing and Selection

Sourcing redundancy receivers to cope with the advent of Carbon Fiber-Reinforced Polymer (CFRP) radio interference was the biggest limiting factor in avionics design, so the FrSky RX8R Pro and FrSky XM+ receivers were finalized first. Afterward, the KST X10 V2 servos were selected to operate the 6 control surfaces because they are lightweight and provide sufficient response time and 8 lbf-in of torque at 7.2 V, enough to overcome the associated hinge moments. The VICMILE 7.4 V 1200mAh LiPo was then accordingly chosen to power the servos and receivers and satisfy the AR-15.

4.4 Aircraft Performance Predictions

4.4.1 Uncertainties

Assumptions pertaining to the analysis and conditions of flight were taken to influence the design of Condor. According to the average weather forecast for Wichita Kansas, there is an average wind speed of 21 during the month of April. The average direction of the wind is 41% South facing, and 30% north-facing as shown in Fig. 15. Cloud cover is on average 65%, and the probability of precipitation is between 25% and 28%. In April, Wichita has a probability of snowfall, tornadoes, and rainstorms. Daylight hours are from 7 am to 8 pm over the timeframe of the competition. The average temperature is a high of 70°F and a low of 49°F. The potential for a crosswind at take-off implied that improved pitch and yaw authority was also necessary. This information also influenced the control surface sizing in Section 4.3.2.2.

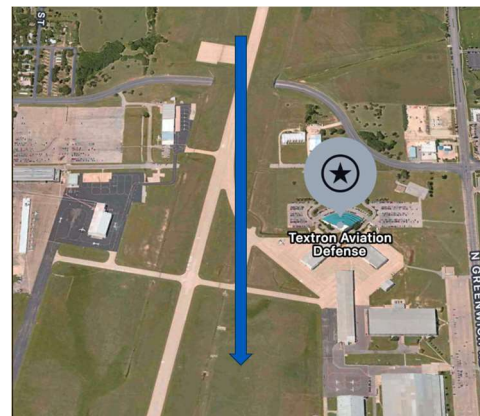


Figure 15. Wind Direction on Runway

4.4.2 Static Stability Analysis

The static stability of the aircraft was analyzed for Condor with AVL for each mission. The fuselage body was not modeled because slender bodies lead to errors in AVL. Only the wings and tail configuration, their relative positioning and hinge locations, and the necessary propulsion and payload components were incorporated for each mission configuration. The aircraft is statically stable when there is a tendency for the aircraft to develop forces and moments that oppose the perturbation state returning the aircraft to its initial trimmed condition [13]. To have longitudinal static stability, the aircraft must have a positive zero lift moment coefficient and the pitching moment coefficient must become more negative as α increases [8]. The dihedral effect ($C_{l\beta}$) is negative for lateral static stability, and the $C_{n\beta}$ (yaw moment coefficient) must be positive for directional static stability. The neutral point was calculated and the static margin for M1, M2, and M3, was identified with respect to the neutral point as the difference between the CG and the NP for each mission. The MSC payload was mounted such that the CG aligned with the aircraft CG to minimize

the change in CG between missions. Identified in Table 15 are the stability derivatives calculated for all missions using AVL. The range of the stability derivatives is referenced from Sadraey [8].

Table 15. Stability Derivative for Each Mission

Mission	Stability Derivatives					Static Margin	Neutral Point
	$C_{m\alpha}$	C_{mq}	$C_{m\beta}$	C_{nr}	$C_{l\beta}$		
Range	-0.3 to -1.5	-5 to 40	0.05 to 0.4	-0.1 to 1.0	< 0	0.1 to 0.3	-
M1	-0.7851	-26.9374	0.3598	-0.80972	-0.4599	0.1041	0.83016
M2	-0.8137	-28.5231	0.3503	-0.7721	-0.3747	0.1027	0.82906
M3	-0.8498	-27.2132	0.3499	-0.7659	-0.4013	0.1035	0.82968

Table 15 showed that all the stability derivatives fall within the suggested ranges and that Condor is both longitudinally and laterally stable. At cruise, each mission was trimmed individually to assess which aircraft configuration required additional trim than others to meet the CI requirement with the fuselage not rotated. With trim, the AOA needed to provide enough lift for M1, M2, and M3 was -3.6, -1.2, and -1.8 degrees respectively. Therefore, it was decided to mount the wings at -2 degrees to best fit the average requirement.

4.4.3 Dynamic Stability

AVL was also utilized to calculate the eigenmodes of the aircraft for each mission. Dynamic stability is the tendency of the aircraft to return to the steady state trim condition without pilot assistance after the aircraft is disturbed [8]. More specifically, it includes the rate at which the perturbation dampens out. The aircraft has longitudinal dynamic stability, if the short and long period modes dampen. If all modes are damped, then the aircraft also has lateral dynamic stability. In Fig. 16, the root locus for each mission is plotted and analyzed. The Dutch roll, phugoid, short period, spiral, and roll modes were all verified for positive dynamic stability to ensure the return to equilibrium position. The roots that reside on the left half of the graph on the real axis will stabilize exponentially. However, the roots that have imaginary components will oscillate and decay to the equilibrium state based on the short and long period modes. From Fig. 15 the various modes for the aircraft during the missions were identified along with the corresponding damping ratio (ζ), natural frequencies (ω), and settling time (t_s). The longitudinal and lateral dynamic stability modes are outlined in Table 16.

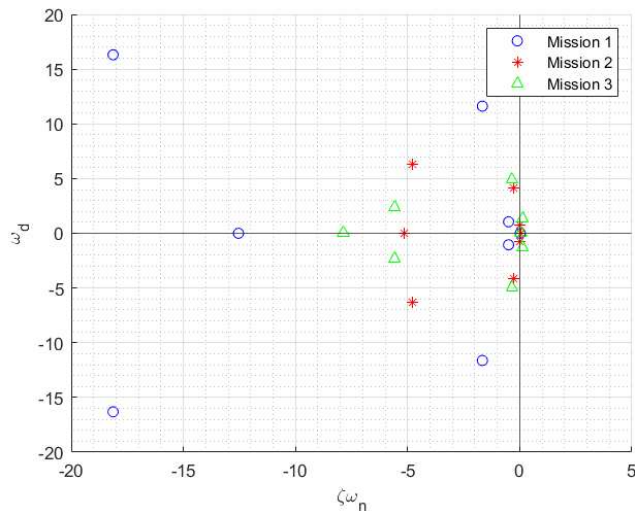


Figure 16. Root Locus Plots

Table 16. Dynamic Stability Modes for Each Mission

Mission	Short Period			Long Period		
	ζ	ω	t_s	ζ	ω	t_s
M1	0.7435	24.291	2.4017	0.1419	11.7363	0.319
M2	0.6072	7.9001	0.8339	0.0672	4.1795	14.243
M3	0.9211	0.0569	76.335	0.0699	4.9775	11.493

Mission	Dutch Roll			Spiral			Roll		
	ζ	ω	t_s	ζ	ω	t_s	ζ	ω	t_s
M1	0.4284	1.1635	8.0251	1	0.0205	194.94	1	12.5405	0.319
M2	0.0079	0.7911	636.43	1	0.0646	61.932	1	5.1597	0.7752
M3	0.0907	1.3376	32.978	1	6.062	6.598	1	7.8569	0.5091

The short period for M3, Dutch roll for M2, and spiral for M1 and M2 were determined to have no significant issues for flight with the corresponding values due to pilot input that can correct the perturbations.

4.4.4 Takeoff Performance

The takeoff performance analysis results from the optimization study are presented in Fig. 17 with the conservative lift coefficient of 1.6 observed in Fig. 12. As discussed, the implemented takeoff strategy involved blown slotted flaps and thrust vectoring in the rotated state. Slats and Fowler flaps were other high-lift generation devices considered, but not executed because of projected manufacturing and structural complications. Flaperons were set to be evaluated after flight testing, as they could be programmed in software easily at the slight cost of total upward deflection.

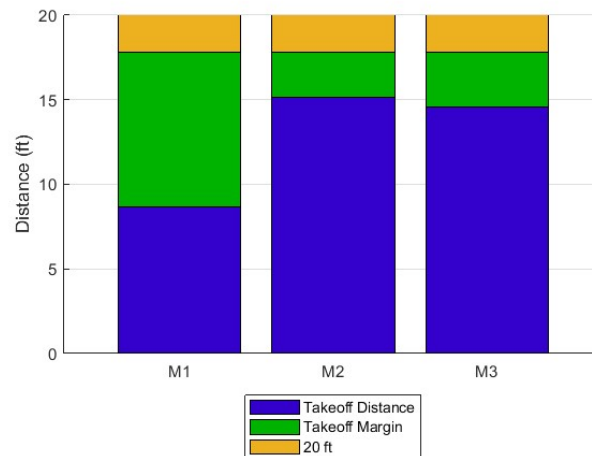


Figure 17. Takeoff Distance Prediction

4.4.5 Turn Radius

The turn radius was calculated for each mission based on the intended speed at cruise for each mission and is represented in Fig. 18. The turn radius for each mission was utilized to determine the total flight time for M1, M2, and M3. The analysis served as an addendum to the optimization.

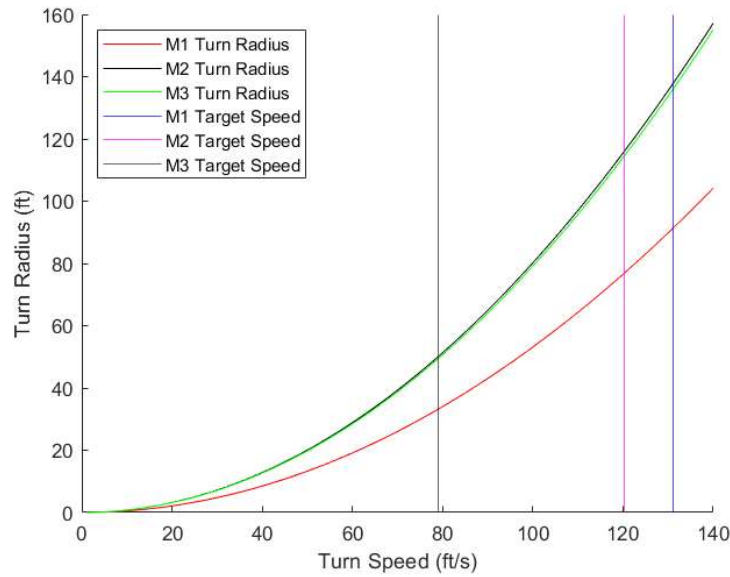


Figure 18. Mission Turn Radius

4.5 Aerodynamic Characteristics

4.5.1 Lift Prediction

Using ANSYS Fluent as seen in Fig. 19, Goose Works simulated the amount of lift generated at cruise for each mission with ANSYS Fluent shown in Table 17. The estimated lift predictions for each mission at cruise were shown for each component and summed together to determine the lift prediction for Condor based on the individual mission parameters.

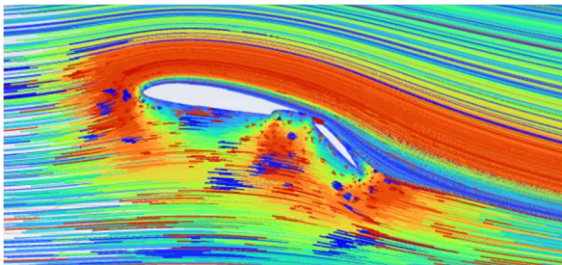


Figure 19. ANSYS Wing Simulation

Table 17. Lift Predictions

Lift [lbf]			
Component	M1	M2	M3
Fuselage	5.155	4.34	1.871
Tail	-0.732	-0.273	-0.618
Landing Gear	0	0	0
Wing	38.089	32.064	15.547
Sum			
	42.512	36.13	16.8

4.5.2 Drag Prediction

The total drag coefficient (Eq. 29) was determined by simulating the fuselage, tail, landing gear, and wings in ANSYS Fluent to determine the parasitic and induced drag components for each mission based on the aircraft velocity at cruise.

$$C_{D,T} = C_{D,0} + C_{D,i} \quad (29)$$

In ANSYS, each component could have been analyzed individually but was positioned such that it was attached to the aircraft in the control volume preventing flow in specific directions. For example, the wings could be aligned with the left or right side of the control volume to simulate the effect of being attached to

the fuselage without the entirety of the fuselage being simulated. This methodology was both computationally efficient and helped the team to run into less problematic geometries when meshing. Therefore, the total parasitic drag of the aircraft was found in Eq. 30 and the individual component values are represented in Table 17.

$$D_{D,0} = D_{D,0,F} + D_{D,0,T} + D_{D,0,W} + D_{D,0,LG} \quad (30)$$

The parasitic drag coefficient was then calculated for each component for each mission with Eq. 31.

$$D = C_{D,0} \frac{\rho V^2}{2} A \quad (31)$$

Following this, the lift coefficient for each component that was characterized in ANSYS was applied to Eq. 32 to determine the induced drag coefficient.

$$C_{D,i} = \frac{C_l^2}{\pi e AR} \quad (32)$$

Finally, Eq. 29 was used to determine the total drag coefficient for each mission and is depicted in Table 18 with the information from Section 4.5.1.

Table 18. Drag Predications

Component	Drag [lbf]		
	M1	M2	M3
Fuselage	1.084	0.927	0.432
Tail	0.773	0.663	0.316
Wing	2.010	0.859	0.801
Landing Gear	0.859	0.726	0.317
Total Drag	3.985	3.355	1.447

4.5.3 Power Curve

After total drag was ascertained, the static thrust solver from Section 4.3.4.2 was reconfigured to iteratively compute the pulse width modulation (PWM) duty cycles at which the thrust available equals the drag in straight, level flight for M1, M2, and M3 for both the 16x10E and 17x10E configurations. From there, the minimum battery power drawn at airspeed was calculated as depicted in Fig. 20. The power analysis was used as additional validation for the optimization BC estimations (endurance).

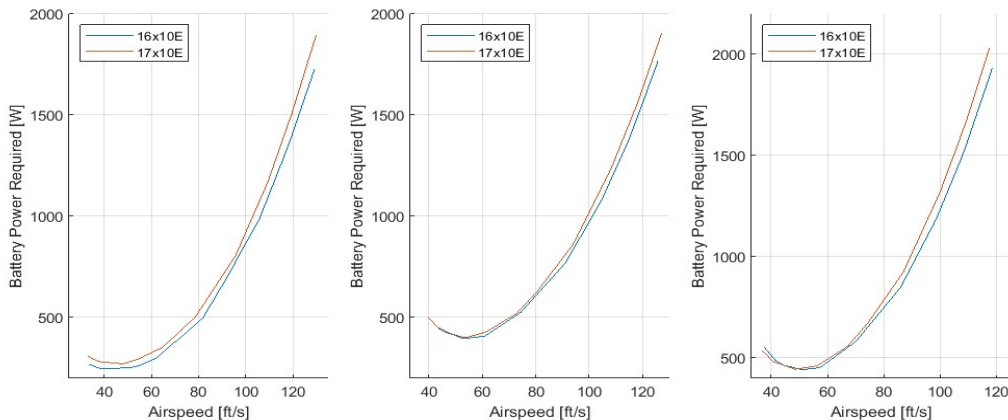


Figure 20. Battery Power at Airspeed for M1 (Left), M2 (Center), and M3 (Right)

4.6 Predicted Mission Performance

4.6.1 Flight Performance

The mission performance was calculated with the optimization calculator described in Section 4.2, the power analysis described in Section 4.5.3, and weather uncertainties. By obtaining drag prediction data from ANSYS and inputting aerodynamic characteristics into the previously mentioned codes, the team was able to predict the various flight performance parameters in Table 19. However, due to the uncertainty of other team scores, M2 and M3 scores were calculated by the constant (1 and 2) plus the numerator.

Table 19. Aircraft Flight Performance

Parameter	Mission 1	Mission 2	Mission 3
Payload Weight/Size	Crew	7 lbs	64 Passengers
Battery Capacity	82.08	82.08	82.08
Takeoff Gross Weight [lbs]	11.12	16.48	16.23
Takeoff Distance [ft]	8	14	13.8
Average Cruise Speed [ft/s]	131	120.4	79
Mission Duration [s]	103	136	374
Number of Laps	3	3	5
Score for Mission	1	1.051	6.266
Flight Score	8.317		

4.6.2 Ground Mission Performance

As described in Section 3.1 the GM is a configuration demonstration. The GM was broken up into three sections: Stage 1, Stage 2, and Stage 3. Stage 1 incorporates the first instance where the stopwatch is started and stopped, and stage 2 incorporates the second instance when the stopwatch is started then stopped, and so on. To determine the time each stage would take, the times from the ground test 1 were adjusted by 100% for a conservative estimate of improved mission performance from V1 to V2 due to structural modifications that made the mission profile easier to perform as shown in Table 20.

Table 20. Ground Mission Performance

Ground Mission			
	Stage 1	Stage2	Stage3
Time [s]	70	180	50
Total Time [s]	300		

5 Detail Design

5.1 Aircraft Specifications

Table 21 shows the overall aircraft dimensions and main propulsion component specifications.

Table 21. Aircraft Specifications

Sizing				Component Specification	
Fuselage		Wing		Avionics	
Total Length	60.00 in	Airfoil	SD7032	Transmitter	Radiomaster TX16s
Max. Width	4.00 in	Span	5.00 ft	Receivers	FrSky RX8R Pro and FrSky XM+
Max. Height	6.60 in	MAC	16.62 in	Avionics Battery	VICMILE 7.4V 1200mAh LiPo
Nose Length	5.60 in	AR	3.84	Servos	KST X10 V2
Rear Taper Length	6.00 in	Wing Area	6.50 ft ²	Propulsion	
Landing Gear		Root Chord	20.50 in	Motors	Hobbystar C4130-435 KV
Height	10.00 in	Tip Chord	12.00 in	Propellers	APC 18x10E and APC 18x10EP
Thickness	0.25 in	Angle of Twist	3.00 deg	Propulsion Battery	HCL-HVP 22.8V-3600mAh-75C
Base Width	15.25 in	Mount Angle	-2.00 deg	ESCs	Onyx 80 A 3-6S Brushless Air ESCs
V. Stabilizer		H. Stabilizer		Arming Fuse	MAXI 100 A
MAC	9.40 in	Span	28.00 in		
V. Area	1.13 ft ²	MAC	10.30 in		
Max Deflection	30.00 deg	H. Area	1.92 ft ²		
Sweep Angle	35.10 deg	Max Deflection	30.00 deg		

5.2 Structural Characteristics and Capabilities

Section 4.4.1 previously mentioned the concern for high wing loading due to the wind conditions in Kansas. Along with this, the aerodynamic, inertial, and landing loads were taken into consideration when designing the structural components of Condor. The wings were chosen to be folded at the fuselage and the concern for shear failure of the hinging mechanism was dismissed by the load-bearing spar that runs through the entirety of the fuselage and wings. Therefore, the maximum aerodynamic loads that the wings can withstand were calculated based on the strength of the wing spar along with the tail affected by the aerodynamic loads. The necessary structural integrity of the fuselage was evaluated by the loads produced by the propulsion system and the inertial loads. An impact loading case was studied using FEA to ensure the landing gear meets required loading.

5.3 System and Sub-System Design/Integration/Architecture

5.3.1 Fuselage

The aircraft's fuselage was sized based on internal volume dependent upon the output optimal number of passengers from the optimization calculator as well as factors such as rotational inertia and presented cross-section as it applied to drag. Therefore, the fuselage with the smallest presented area would have the lowest drag, and lowest rotational inertia in the roll axis while allowing for more span to be used for wing area, which increases the AR of the wing and therefore cruise efficiency as compared to a wider fuselage configuration. To facilitate a strong and light fuselage design, a Carbon Fiber Reinforced Polymer (CFRP) square tube was used for the main longitudinal structural member of the aircraft as well as serving as the

crew compartment. Preliminary testing of fuselage tube samples revealed that the primary weakness of the design was the buckling of the fuselage walls well in advance of predicted strength. To combat this, ribs were implemented in the span between the main and nose gear to reinforce the hatch opening, and the span was exposed to the largest dynamic loading upon landing. To better facilitate maintenance and swapping of components, all avionics were included on top of the fuselage tube underneath a hinged aerodynamic fairing which is a thinly molded CFRP part to save weight. All hinged components leveraged an integral Kevlar hinge as the method was proven historically and provided the lightest hinge solution evaluated by Goose Works. Coincidentally, regions with Kevlar hinges were additionally reinforced by another composite layer which aids in local reinforcement around hatch openings.

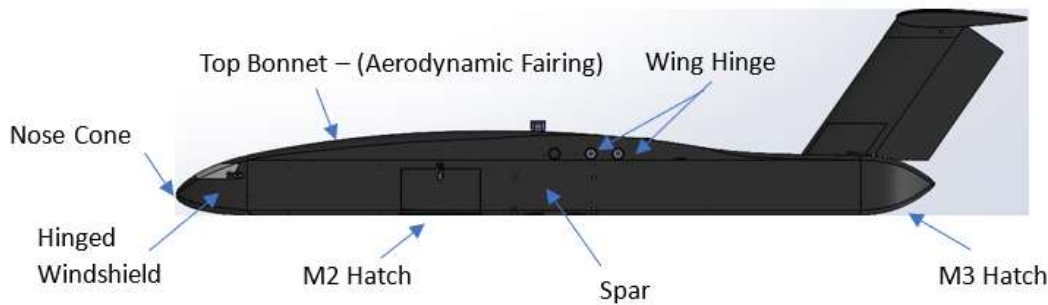


Figure 21. Fuselage Detail View

5.3.2 Nosecone/Bulkhead

The bulkhead had three separate sections which were conceived to add strength to the landing gear. The gap shown in Fig. 22 between the top of the bulkhead and the top of the nosecone was covered by the bonnet as shown in Fig. 21 by the streamlined fuselage. The added bulkhead sections increased the rigidity of the nose gear, allowing for a higher FOS during landing, during which the greatest number of forces are being felt by the landing gear. Additionally, the nosecone contained the hatch used to insert crew members.

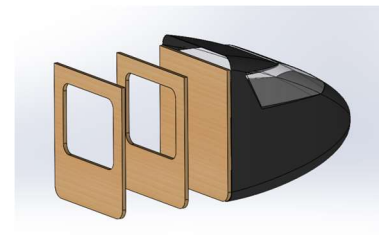


Figure 22. Nosecone/Bulkhead

5.3.3 Mission Component Subsystems

5.3.3.1 Hatch Placement and Opening Mechanism

The fuselage contains three hatches in total: one 6 in hatch that accesses the passenger bay from the back of the plane to load the M3 cargo cockpit for crew insertion, one at the cockpit for crew insertion, and a 6 in long side hatch intended to load M2 cargo. The crew hatch on the nosecone has a hinge located on the same side as the hatches for M2 and M3.

The M2 hatch in Fig. 23 exists on the port side of Condor, slightly forward of the center of mass. Placing the hatch at that location allows easy positioning of the MSC (3 in by 3 in by 3.5 in) at the center of mass. Additionally, the positioning helps minimize the cargo loading time increasing the score of GM. While having the hatch exactly at the CG would further minimize the loading time of the MSC, the team decided to move the hatch forward of the CG due to structural concerns. By



Figure 23. M2 Hatch Open/Closed

doing so, the extra space near the center of mass had added structural support to increase fuselage strength for better support of both the imparted wing loading from the wing spar and the shock loading of the main landing gear. Furthermore, having the hatch ahead of the MSC nullifies concerns regarding the MSC breaking the hatch open during a high G turn. This allowed the implementation of a lighter hinging mechanism, and a smaller external hatch.

The M3 hatch (Fig. 24) was placed on the back, left side of the fuselage below the tail to prevent crossing the fuselage centerline. The back of the aircraft is an aerodynamic fairing to reduce drag and the left side of this fairing is turned into a hatch. A wooden structure was placed at the fuselage centerline to guarantee the hatch does not cross it. The placement of this hatch was selected solely for minimizing M3 loading

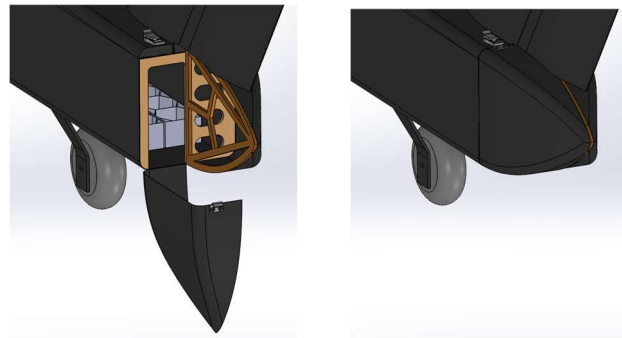


Figure 24. M3 Hatch Open/Closed

time. Two continuous inserts that span the fuselage length are used. They are meant to be pushed into the plane from the back, all the way to the front to load M3 crew passengers.

All the hatches made use of the same mechanism to open and close. Each hatch is attached to the fuselage by a sheet of Kevlar which acts as a hinge, the same method used to attach the control surfaces. The difference is the hatches will have a small draw latch attached to them that is used to hold that hatch closed.

5.3.3.2 Primary Cargo Restraint System

The primary cargo restraint system (Fig. 25) is four ¼" wooden dowels that run through the width of the fuselage, located at the CG. This system is utilized to both constrain the MSC in M2 and the passenger

inserts in M3. The dowels themselves are constrained by pins to hold them in place so they do not translate during flight. In M2, it accomplishes this task by tightly fitting the MSC between the dowels preventing it from translating longitudinally during flight. Additionally, the fuselage ribs contacting the MSC are designed to prevent it from translating laterally or vertically. In M3, the passenger inserts are constrained by having holes that the dowels go through, constraining their longitudinal and vertical movement. Like the MSC, the passenger inserts are laterally constrained by the fuselage ribs.

The implemented cargo constraint system may not be ideal for cargo loading times due to the removal and insertion of the dowels; however, it does a great job at constraining passenger inserts and MSC during flight. The team was comfortable with that compromise because of the high risk it would represent to have any type of cargo loose during the flight due to the potential shift of the CG and loss of stability.

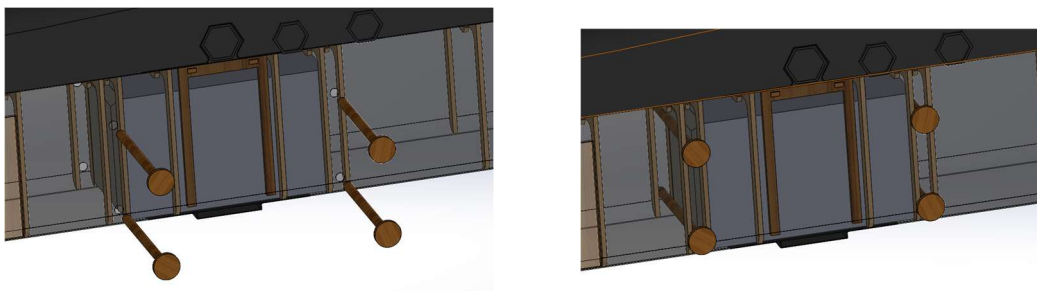


Figure 25. Primary Cargo Restraint System Loading Phase (Left) and Flight Configuration (Right)

5.3.3.3 Secondary Cargo Restraint System

The secondary cargo restraint system (Fig. 26) consists of two pieces of Velcro attached to the inner wall of the fuselage placed forward of the M2 hatch. The Velcro pieces are used to constrain the Gurney and a small insert for the EMTs in M2. The insert for the EMTs has another sheet of Velcro to be connected to the one on the fuselage wall. It was proven that the Velcro joint could constrain the patient on the gurney and EMTs successfully due to their low weight. This solution is also good for loading time because it is very quick to attach due to the ease of use of the Velcro sheets. During M3, the Velcro within the fuselage will not do anything, and will not interfere with the loading of the M3 inserts.

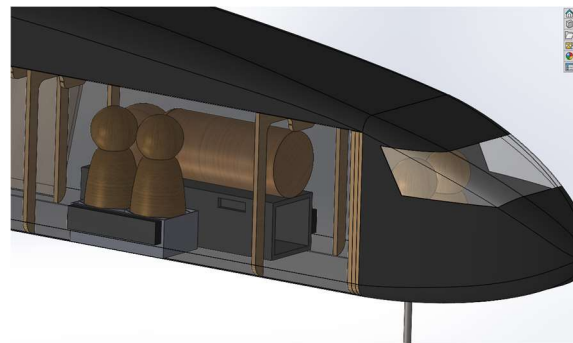


Figure 26. Secondary Cargo Restraint System

5.3.3.4 Crew/EMT/Passenger Inserts

All the inserts used to hold the wooden peg people will have the same fundamental design, which is a repeating hexagon pattern that can hold the passengers during flight using a friction fit. The insert is made of 3D printed PLA because of the required precision needed to get the desired friction fit. Additionally, 3D

printed PLA allows to optimize the specific wall thickness and infill density to minimize weight while ensuring the inserts will not break when used, which was often the case with lighter materials, i.e. foams.

The crew (Fig. 27) will be held into the airplane by the same mechanism used as the EMTs inserts in M2. The insert consists of two hexagons precisely made to friction-fit two passengers. The primary differences are that the crew holding mechanism will be permanently mounted to the bulkhead, and the EMT insert will have a sheet of Velcro on its side to mount inside the fuselage.

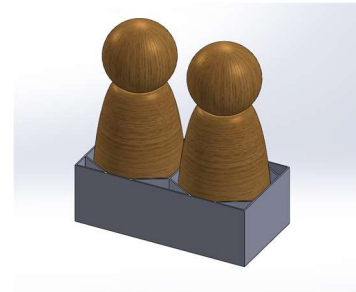


Figure 27. Crew Insert

The M3 inserts for the passengers utilize the same repeating hexagon pattern but on a far larger scale. Each insert for M3 (Fig. 28) is 45" long and carries 30 passengers, for a total of 60. They are made to be

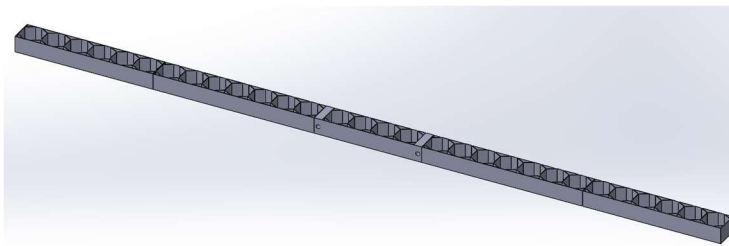


Figure 28. M3 Insert

continuous to easily slide from the rear hatch to the front of the cargo bay. The first insert loaded will be shifted right once inside to make room for the second to be inserted. This solution optimizes loading time. A single insert holding 30 people can be slid into the aircraft quicker than sliding in multiple

small inserts. The center section of the insert has two holes for the dowels from the primary cargo restraint system to slide through, restraining the insert, and by extension the passenger. The main issue with this design is the fact that the continuous inserts are too long to fit on any single 3D printer bed. Therefore, each insert will be printed in 5 separate pieces that can then be bonded together with glue.

5.3.3.5 Patient on Gurney and Medical Supply Cabinet

The gurney is a 3D-printed squared tube (2 in by 6 in by 1.5 in) with two slots cut into it. It was decided to 3D print the gurney as a continuous piece to reduce the chance of failure, compared to a tube made of separate sides fastened together. The slots in the tube are for Velcro straps to thread through, which will be utilized to hold the patient down on the gurney. The side of the tube will also have a sheet of Velcro on it, to attach to the Velcro in the inside fuselage wall. These two aspects of the gurney make it ideal for reducing cargo loading time. It is very fast to put straps over the patient to secure him to the gurney and attach two pieces of Velcro to secure the gurney to the fuselage.

The MSC is made of two parts, an open box that will hold weights and a cap that can be used to close the box to secure the weights during flight. 3D-printed PLA is used to accomplish the desired manufacturing precision of the MSC's dimensions. The MSC will be sized perfectly to fit between the dowels of the restraint system and the ribs of the fuselage so that is constrained in all axes.

5.3.4 Wings

5.3.4.1 Wing Folding Mechanism

The wing folding mechanism was created to allow Condor to have a wingspan of 5 ft. while also being able to fit in a parking spot of 2.5 ft. As shown in Fig. 29, the design of Condor requires a hinge that can both facilitate translation and rotation of the wing. The hinge shown in Fig. 29 was designed to solve that issue. The hinge design leverages telescopic hexagonal carbon fiber tubes to facilitate the translation of the hinge and occlude the hinge location when the wing is down. The wings get into a locked position when the hinge point is retracted within the outer telescopic piece. The hinges themselves are 3D-printed inserts that are glued into cut sections of the smaller of the telescopic tubes. There are 2 hinges per wing which are necessarily offset for clearance and glued to the top of the fuselage tube as shown in (Fig 29). A methyl methacrylate-based glue will be used to bond the composite surfaces. This is advertised to be designed for composite bonding. It has also been tested by Goose Works to be substantially stronger than other glues such as polyester and epoxy resins, especially in peel and shear resistance. The hinges have a hole drilled to receive a spring pin clip which will be used to retain the wings and prevent translation in a flight configuration. As such, the hinge of the wings goes as follows: depression of the spring clips, translation of the wings, and hinging of the wings. As shown in Fig. 29, the translation allows for clearance of the thickness of the airfoil and the wall of the fuselage. This distance also allows for a greater than 90-degree hinge which allows the wings to rest against each other and be fastened simply with a hook that can be deployed from the stationary wing endplate.

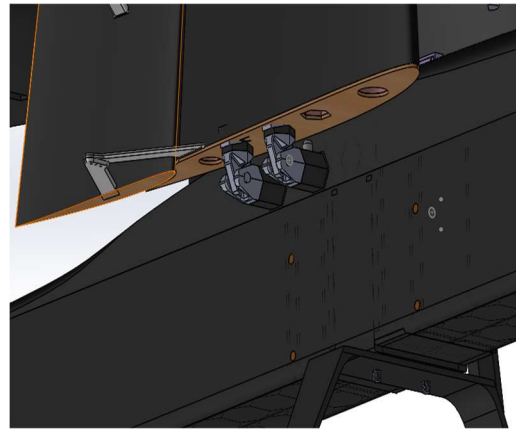


Figure 29. Wing Folding Mechanism

5.3.4.2 Flap Hinge

The design of the flap hinges began with the iterative flap sizing process as detailed in Section 4.3.2.2 which determined the geometry of the flap cove and the required gap between the cove and the L.E of the airfoil to ensure optimal takeoff performance. Starting with this data, the flap hinges were designed in CAD to maintain a constant gap between the wing and flap airfoils in terms of percent instantaneous chord. As the wing and flap are tapered, the hinges must be scaled along the span to maintain this gap. The design of the hinges was determined from the initial design of the root hinge. The hinge point was determined iteratively in CAD as the point that would facilitate the desired gap at the desired deflection of 37 degrees. Once the basic geometry was determined, the hinge was designed to

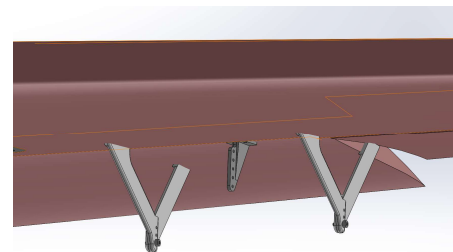


Figure 30. Flap hinge

have adjustable hinge points such that the gap could be adjusted for real-world conditions that may occur. Once the first hinge was designed, subsequent hinges were scaled from the first at a scale proportional to the ratio of the wing chord to the root chord. As shown in Fig. 30, 4 hinges were used to ensure sufficient hinge rigidity with space remaining in the center for the mounting of a control horn to deploy the flap. The flap hinges were designed to be 3D printed from Acrylonitrile Styrene Acrylate (ASA) as it was determined that this would require the least man hours for the relatively complex geometry as shown in Fig. 30.

5.3.5 Nacelles

The nacelles to mount the motor are molded carbon fiber parts designed so that they can fit directly over the geometry of the tapered and twisted SD 7032 airfoil. They will be mounted using a methyl-methacrylate adhesive to securely bond them to the wing skin as shown in Fig. 31. The motors will be bolted to the nacelles to prevent the motors from shifting during flight.

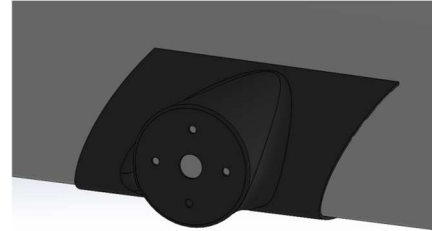


Figure 31. Motor Nacelle

5.3.6 Empennage and Tail

The T-tail (Fig. 32) is broken up into a vertical and horizontal stabilizer, each with an integrated control surface on a Kevlar hinge. The vertical and horizontal stabilizers are made with an XPS foam core skinned with CFRP. The vertical stabilizer is permanently fixed to the fuselage. It is designed to slot into a molded CFRP mount which will be affixed to the fuselage with methyl methacrylate glue. The horizontal stabilizer is fastened to the vertical stabilizer with three bolts, which pass through a 3D printed PLA hardpoint in the horizontal stabilizer and interface with trapped nuts in another plastic hardpoint on top of the vertical stabilizer.



Figure 32. Empennage and Tail

5.3.7 Landing Gear

Finite Element Analysis (FEA) was run on the landing gear (Fig. 33) to analyze how the sub-system would react under a large load. The inertial load of the aircraft and its components on the landing gear was conservatively estimated as 80 lbs, which is 4.5G under the heaviest mission configuration. Fig. 33 shows that the highest stress concentrations occur in the landing gear bolt holes and the horizontal section of the landing gear that is annexed to the fuselage. The FOS at the bolt holes is 0.8, and the FOS of the landing gear section next to the fuselage is about ~1.3. Since the operating conditions are going to be much lower than in the FEA study case, the landing gear will be structurally safe.

The mount for the landing gear is a plywood plate sandwiched between the fuselage and a series of carbon fiber flanges. Some bolts permanently sit in the plywood as can be seen in Figure 34. The tall, vertical

flanges mount to the inner walls of the fuselage through slits cut into the bottom of the fuselage. The short, horizontal flanges mount to the outer surface of the bottom of the fuselage. Flanges are glued to the fuselage using methyl methacrylate.

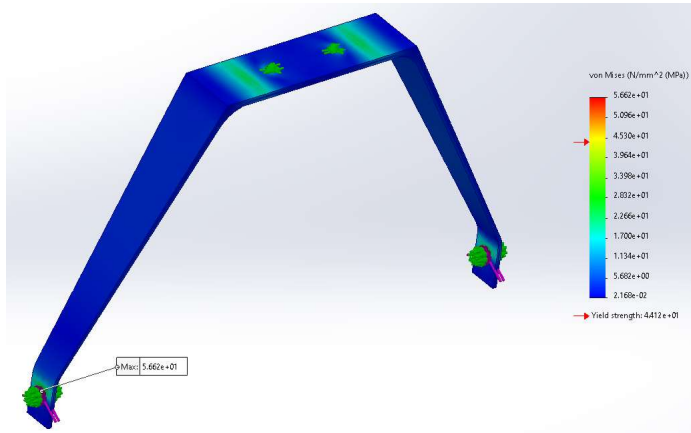


Figure 33. Landing Gear FEA

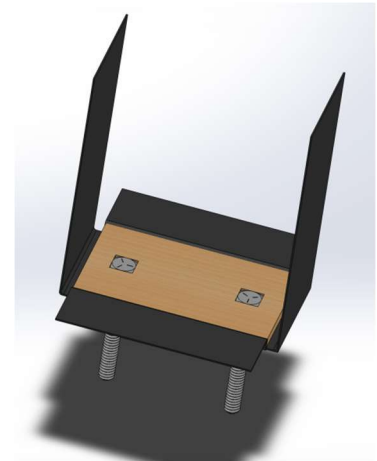


Figure 34. Landing Gear Attachment

5.4 Propulsion and Avionics Subsystems

The preliminary propulsion design was not changed. Additional thrust testing validated the initial preference of 17x10Es propellers (18x10Es clipped to size).

Fig. 35 presents the wiring for both propulsion and avionics subsystems. Incidentally, the radio switch delineated by AR-11 was the only addition to the preliminary avionics selection. The batteries, ESCs, and primary receiver are free to be moved around within the bonnet for stability. The only fixed component is the 100 A fuse, which must be mounted aft of the propellers.

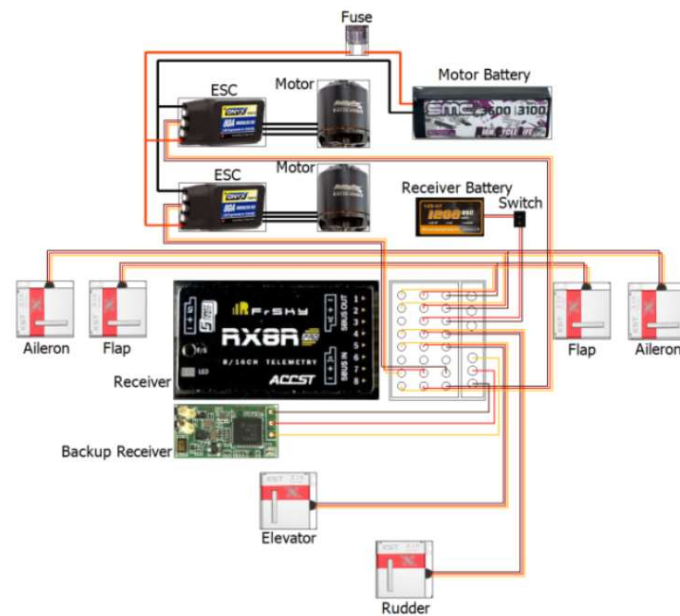


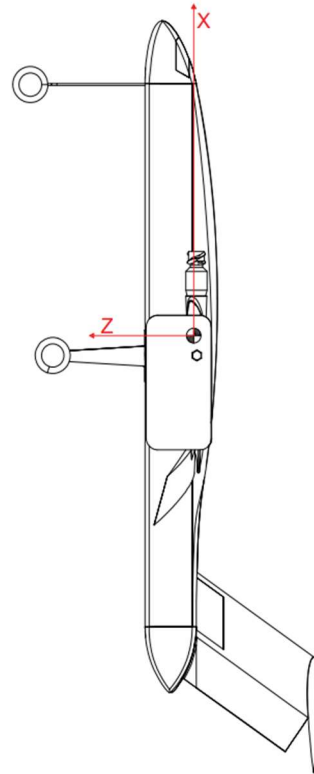
Figure 35. Electronics Schematic

5.5 Weight and Balance

Outlined in Table 22 is the weight and balance of Condor. Electronic components such as the propulsion battery, avionics battery, and ESCs were excluded and placed according to the necessary locations to achieve the desired CG location upon manufacturing the aircraft.

Table 22. Weight and Balance

Component	Weight (lbs.)	CGx (in.)	CGy (in.)	CGz (in.)
Flap Servo (x2)	0.06	-2.4	10.69	0
Aileron Servo (x2)	0.06	-2.4	25.25	0
Rudder Servo	0.06	-26.8	-0.35	-3.56
Elevator Servo	0.06	-30.13	0.91	3
Main Landing Gear	0.25	0.76	-8.59	16.74
Nose Landing Gear	0.15	22.8	1	16.74
Fuselage (Empty)	6	-1	2	-3.09
Wing /Wing Structure	1.25	-0.09	-2.13	1.42
Horizontal Stabilizer	0.3	-33.5	0	-8.06
Motor (x2)	0.9	5.09	-11	0
Vertical Stabilizer	0.2	-19.89	0	-3.09
Battery	1.07	5.8	0	1.66
Mission 1				
Crew	0.18	24.74	0.04	6.37
Mission 2				
MSC	6	0	-1.54	-3.92
Patient+Gurney	0.5	-15.72	0.44	5.05
EMTs	0.18	13.5	0.44	4.5
Crew	0.18	24.74	0.04	6.37
Mission 3				
Passengers	5.5	-11.88	0	6.42
Crew	0.18	24.74	0.04	6.37
CG Location				
M1	11	0	0	0
M2	17.5	0.25	0	0
M3	16.5	-0.1	0	0



5.6 Flight Performance Final Design

The performance was recalculated with the optimization calculator described in Section 4.2 with consideration of the aircraft produced for V1 and detailed CAD design and preliminary manufacturing. By obtaining drag prediction data from ANSYS and inputting aerodynamic characteristics into the preliminary design codes, the team was able to predict the various flight performance parameters for V2 shown in Table 23.

Table 23. Predicted Flight and Mission Performance

Parameter	Mission 1	Mission 2	Mission 3
Payload Weight/Size	Crew	7 lbs	60 Passengers
Battery Capacity	82.08	82.08	82.08
Takeoff Gross Weight [lbs]	11.93	19.23	17.43
Takeoff Distance [ft]	12.04	19.78	15.03
Average Cruise Speed [ft/s]	120.00	118.11	65.60
Mission Duration [s]	99.80	98.70	296.00
Number of Laps	3.00	3.00	5.00
Score for Mission	1.00	1.19	6.00
Flight Score	8.19		
Ground Mission			
	Stage 1	Stage2	Stage3
Time [s]	63	162	45
Total Time [s]	270		

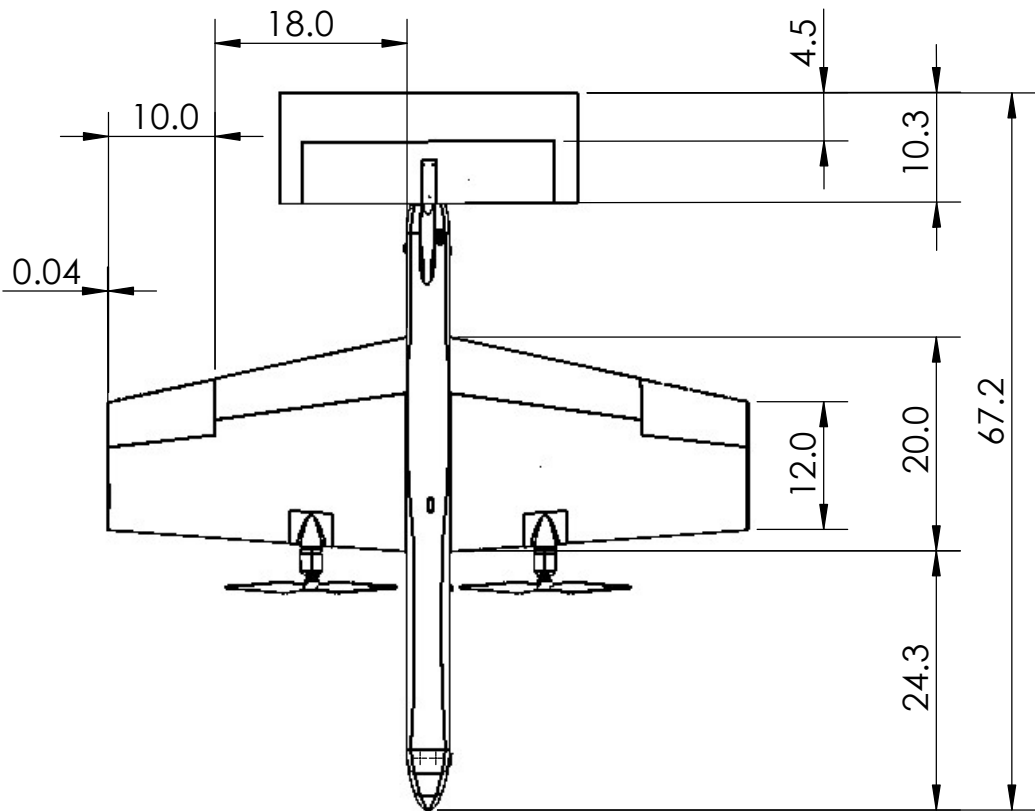
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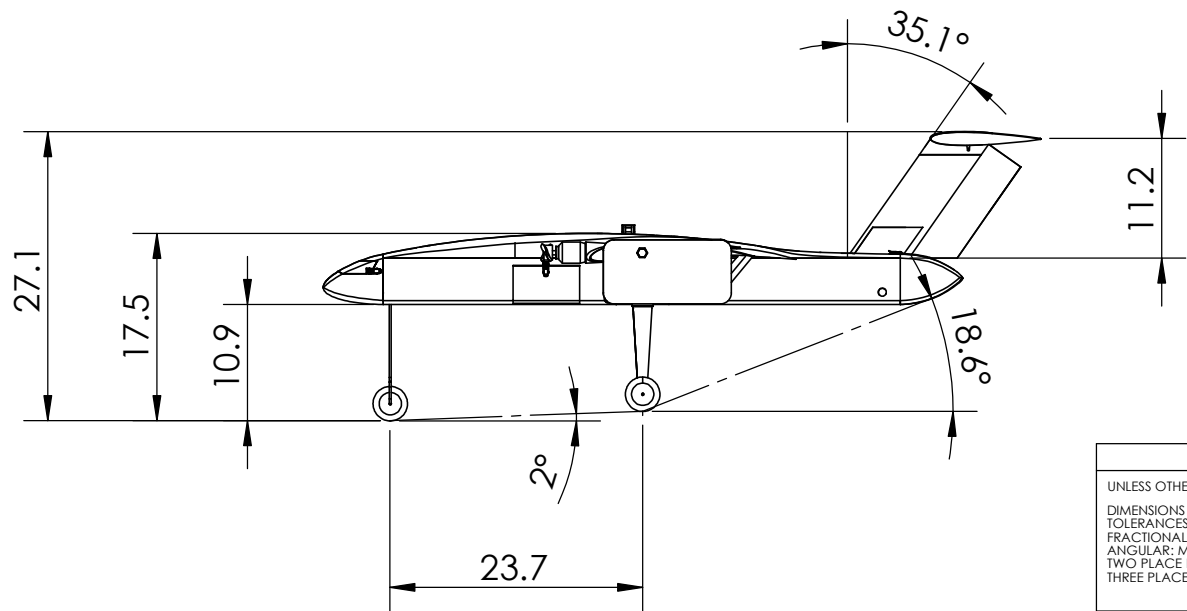
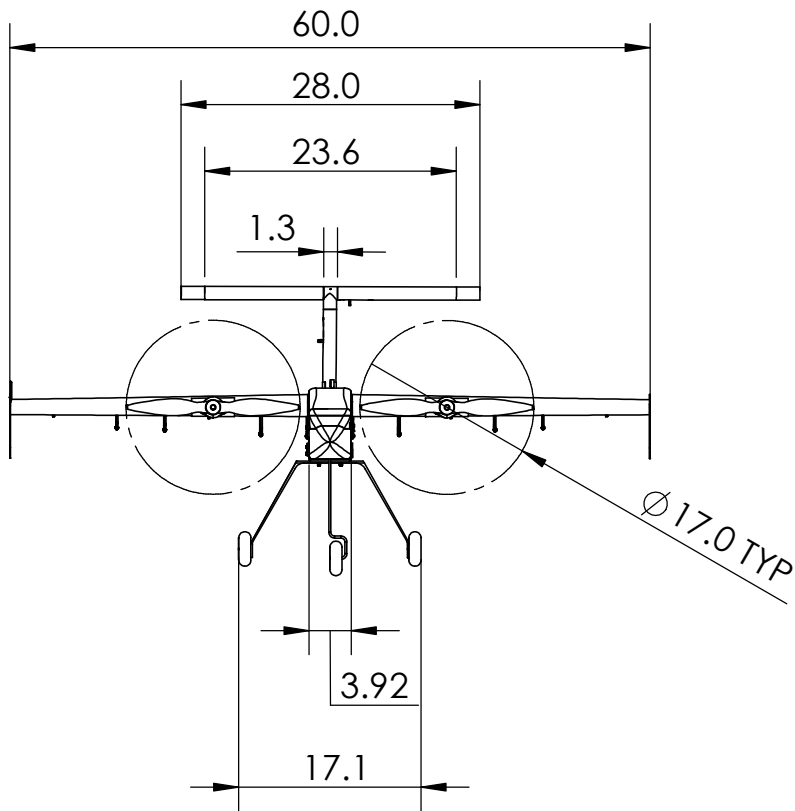
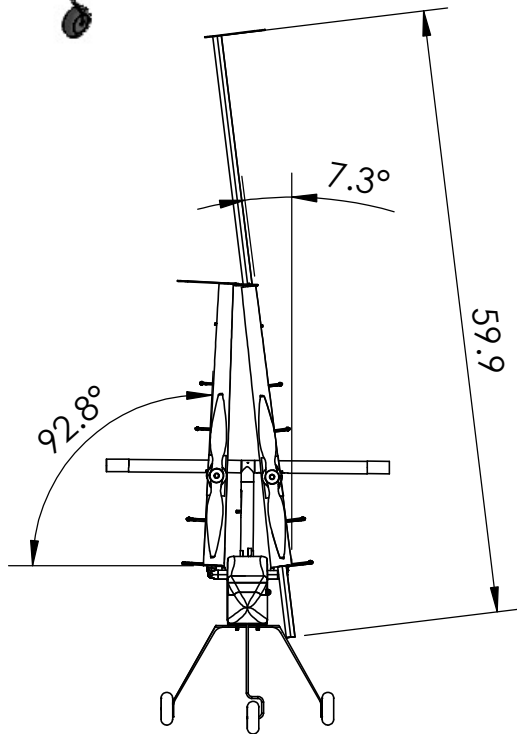
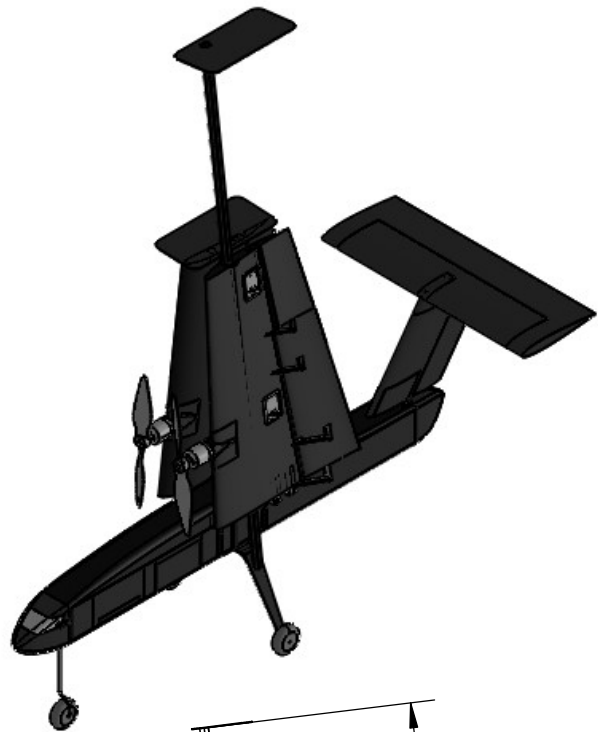
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Takeoff Configuration



Parking Configuration



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B

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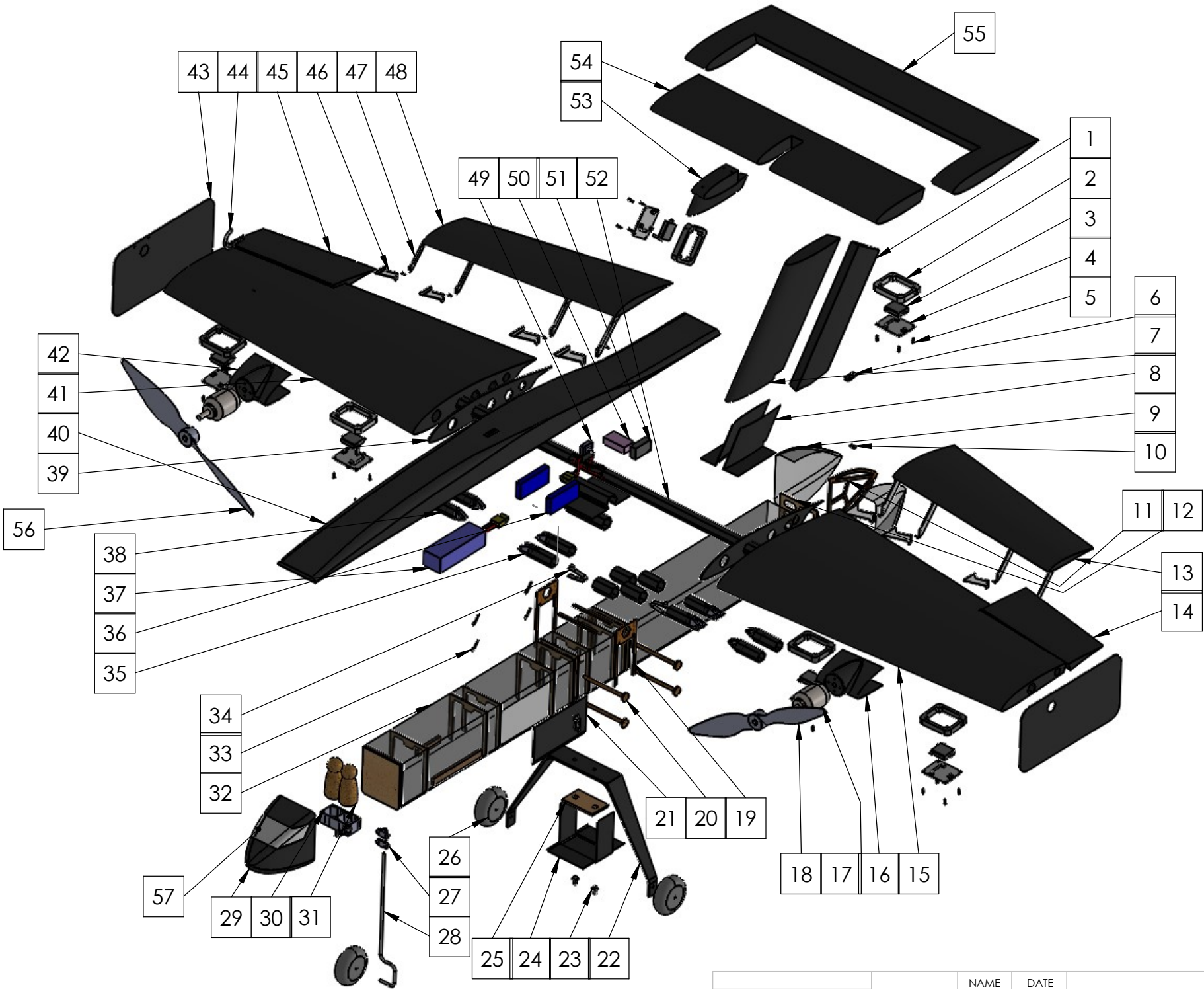
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Item No.	Item	Material	Quantity
1	Rudder	CFRP & XPS	1
2	Servo Mount	Mixed	6
3	Servo	Mixed	6
4	Servo Plate	PLA	6
5	M2X0.6 5mm	Steel	32
6	Latch Bottom	Steel	3
7	Vertical Stabilizer	CFRP & XPS	1
8	Tail Mount	CFRP	1
9	Tail Fairing	CFRP	2
10	Latch Top	Steel	3
11	Tail Rib	Wood	1
12	Back Bulkhead	Wood	1
13	Flap (Port)	CFRP & XPS	1
14	Aileron (Port)	CFRP & XPS	1
15	Wing (Port)	CFRP & XPS	1
16	Nacelle (Port)	CFRP	1
17	Motor	Mixed	2
18	Propeller APC 18x10EP	Fiberglass Reinforced Polymer	1
19	Wingspar Mount	Wood	1
20	Restraint Pin	Wood	4
21	M2 Hatch	CFRP	1
22	Landing Gear	CFRP	1
23	1/4"-20	Steel	2
24	L.G. Mount	CFRP	1
25	L.G. Plywood Mount	Wood	1
26	Wheel	Rubber	3
27	Front L.G. Mount	Plastic	1
28	Front L.G.	Steel	1
29	Nosecone	CFRP	1
30	Crew Seats	PLA	1
31	Crew Members	Wood	2
32	Fuselage	CFRP	1
33	Cotter Pin	Steel	4
34	Spring Clip	Steel	4
35	Wing Hinge (Inside)	CFRP & PLA	4
36	Electronic Speed Controller	Mixed	2
37	Propulsion Battery	Mixed	1
38	Wing Hinge (Outside)	CFRP & PLA	4
39	Wingroot Rib	Wood	2
40	Top Bonnet	CFRP	1
41	Wing (Starboard)	CFRP & XPS	1
42	Nacelle (Starboard)	CFRP & XPS	1
43	Wingtip Endplate	CFRP	2
44	Wingspar Hook	PLA	1
45	Aileron (Starboard)	CFRP & XPS	1
46	Flap Hinge Front	PLA	8
47	Flap Hinge Back	PLA	8
48	Flap (Starboard)	CFRP & XPS	1
49	Fuse	Mixed	1
50	Avionics Battery	Mixed	1
51	Receiver	Mixed	1
52	Wing Spar	CFRP	1
53	Elevator to Rudder Adapter Mount	PLA	1
54	Horizontal Stabilizer	CFRP & XPS	1
55	Elevator	CFRP & XPS	1
56	Propeller APC 18x10E	Fiberglass Reinforced Polymer	1
57	Windshield	Plastic	1

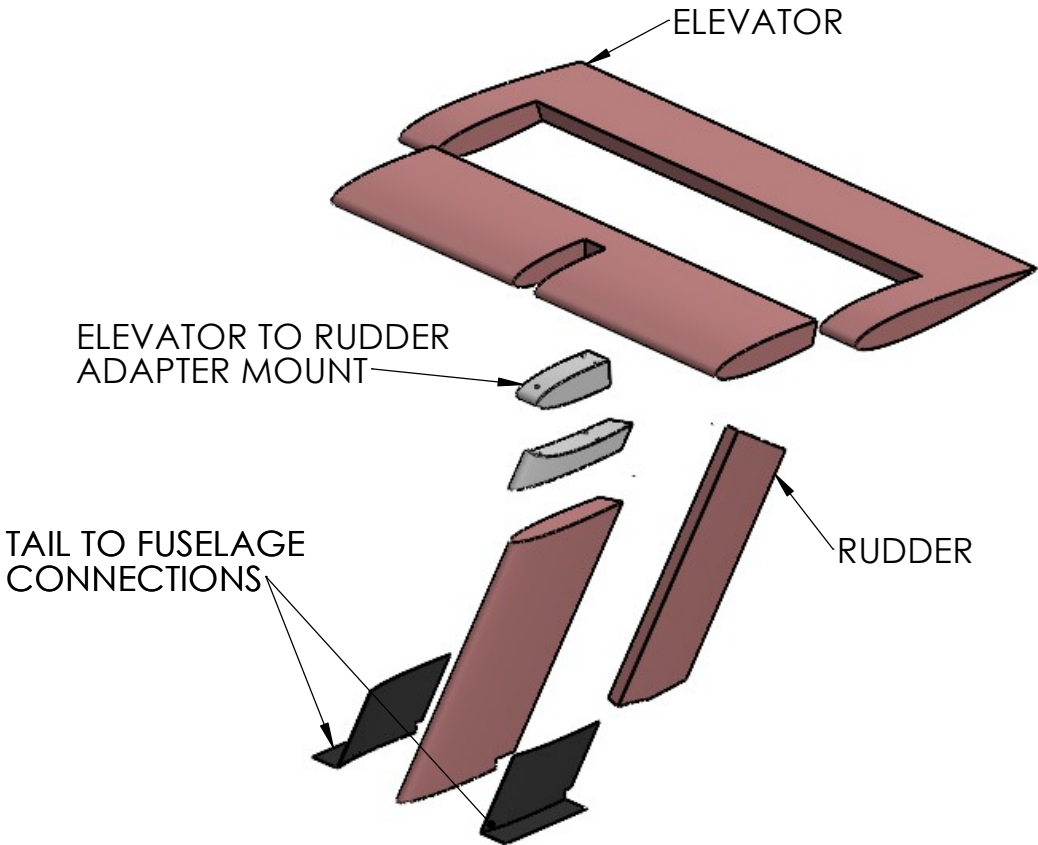
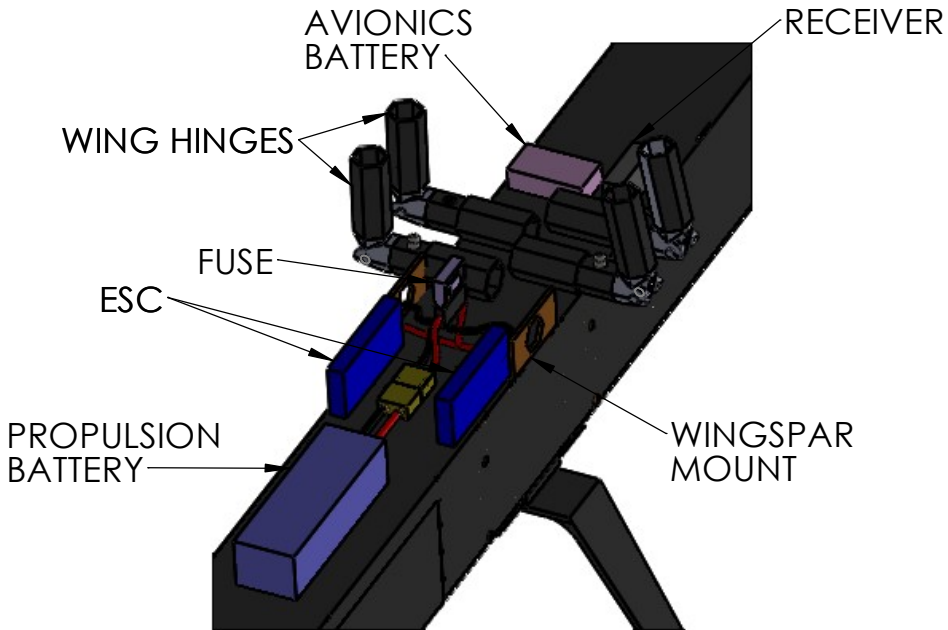
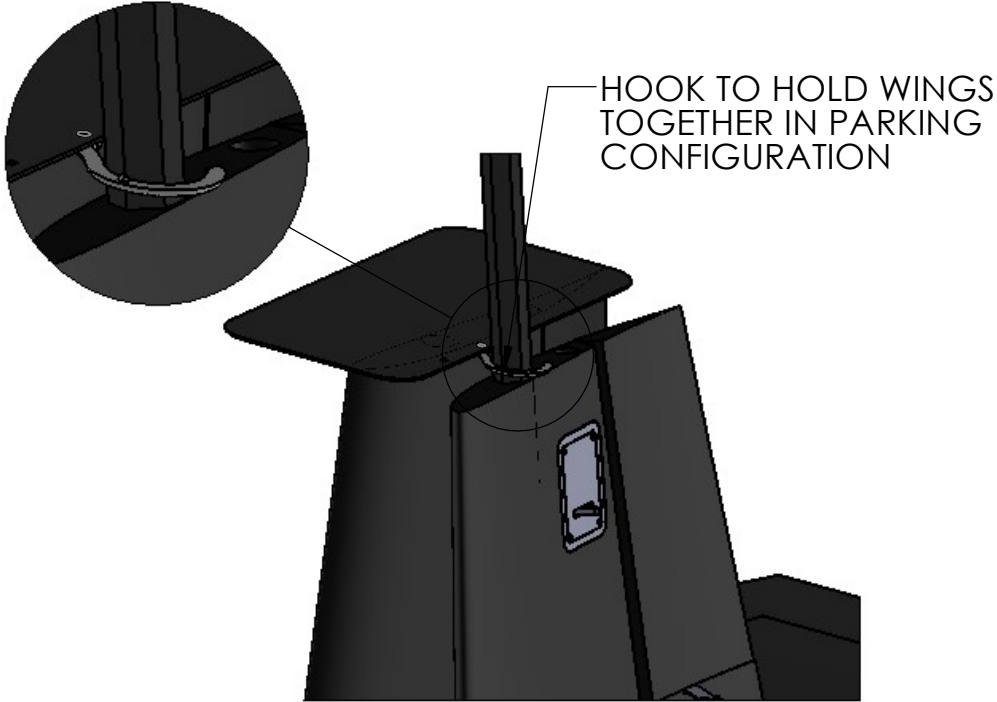


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WINGS IN PARKING CONFIGURATION

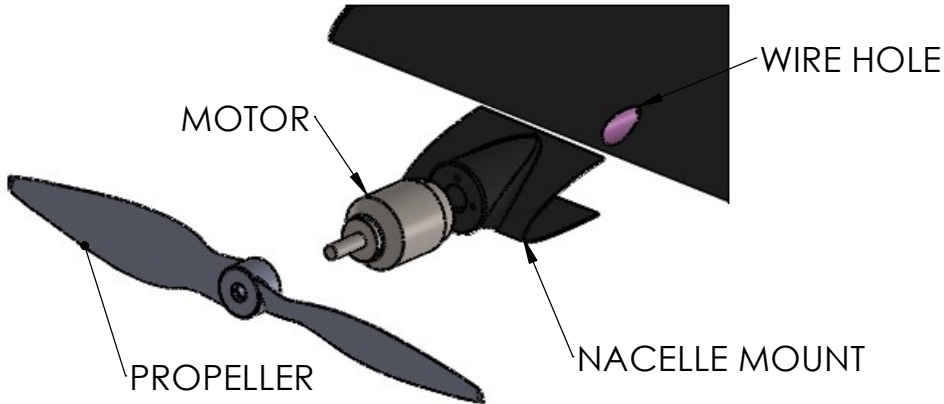
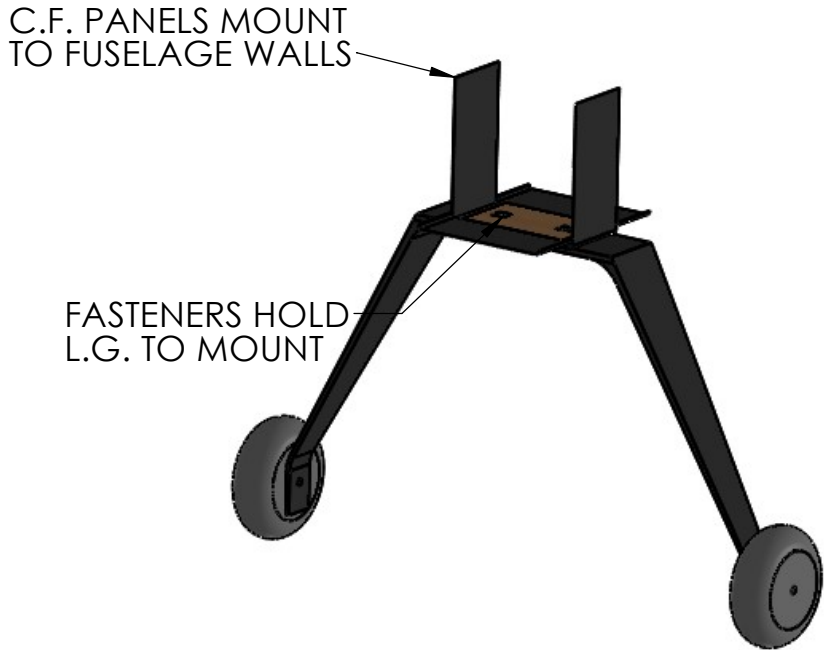
UNDER BONNET VIEW - HINGES UP & ELECTRONICS

TAIL INSIDE VIEW



LANDING GEAR MOUNT

PROPULSION SYSTEM



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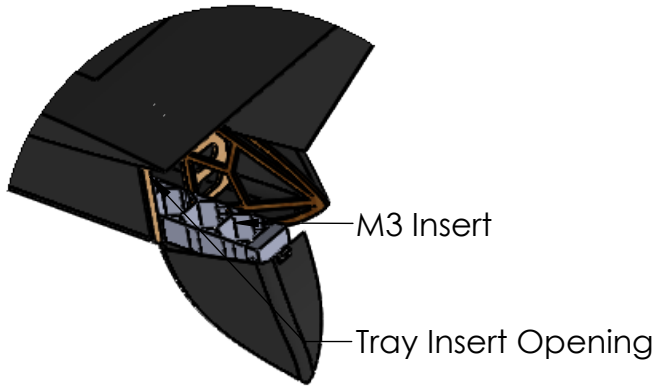
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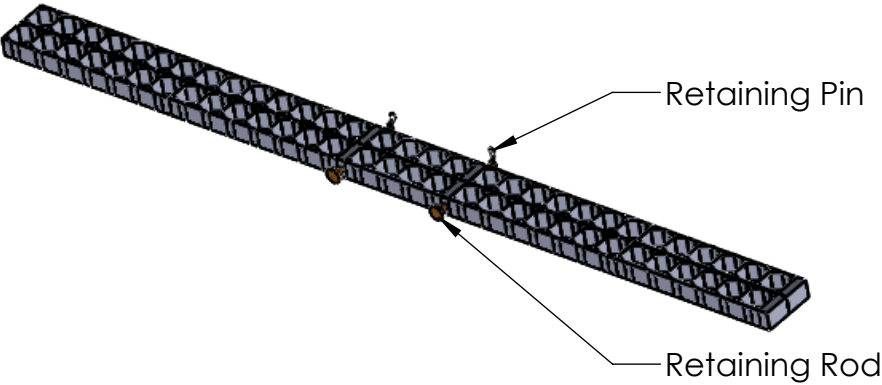
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Mission 3

M3 Opening

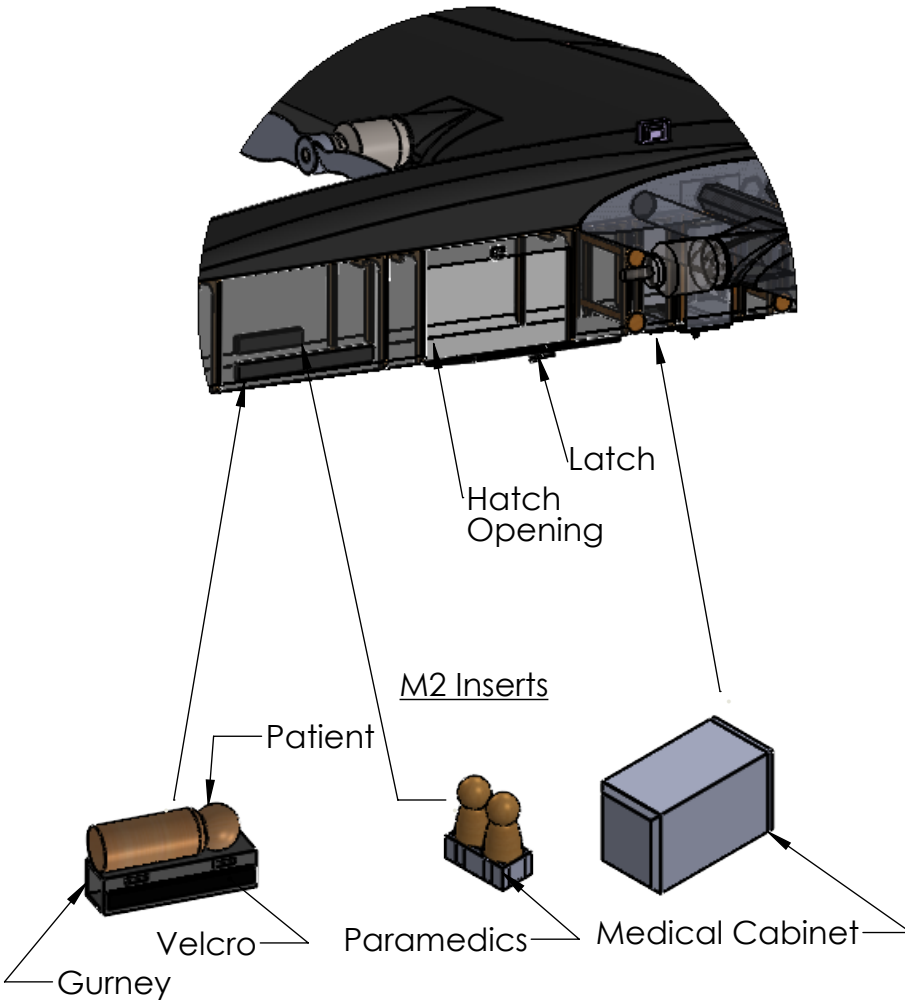


M3 Insert



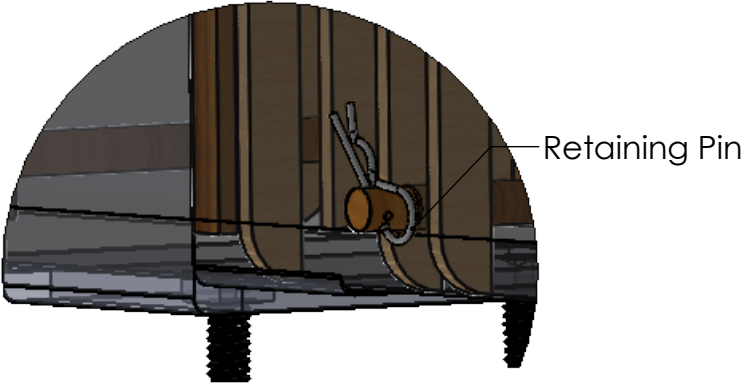
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M2 Opening

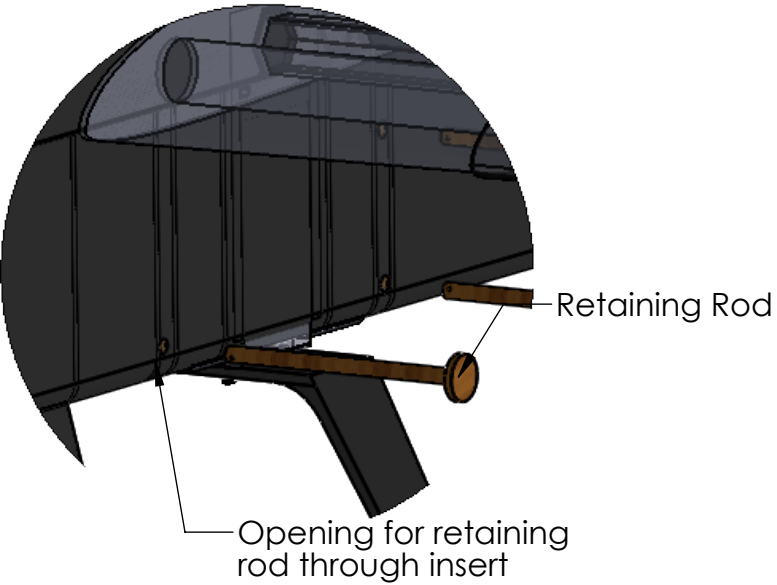


Mission 2 & 3

M2/M3 Restraint System

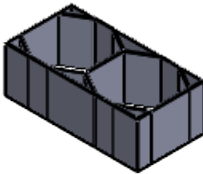


M2/M3 Restraint System

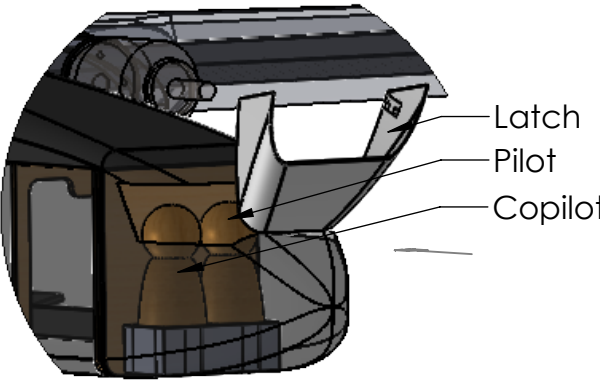


General

Pilot & EMT Holder



Crew Hatch



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6 Manufacturing Plan

Fig. 36 outlines the design manufacturing plan for developing the aircraft within the timeline shown in Fig. 2. The manufacturing and assembly of the V1 prototype began while working on the detailed design of the aircraft. V1 was manufactured using Computerized Numerical Control (CNC) cut foam core wings made of XPS foam, laser-cut plywood ribs, and a wooden and carbon fiber fuselage structure. Incorporation of the mission requirements will be introduced during the detailed design process for components like the hinge door design, wing folding mechanism, and tray insert. The integration of all mission components will be competition ready in the V2 prototype. The second prototype will be made of a carbon fiber fuselage and carbon fiber folding wings. V2 will be used to improve the manufacturing processes for the polymer composites that will be manufactured for the final aircraft, V3. The last version will closely model the second prototype, with minor improvements made to the UAV design to improve mission performance and reliability.

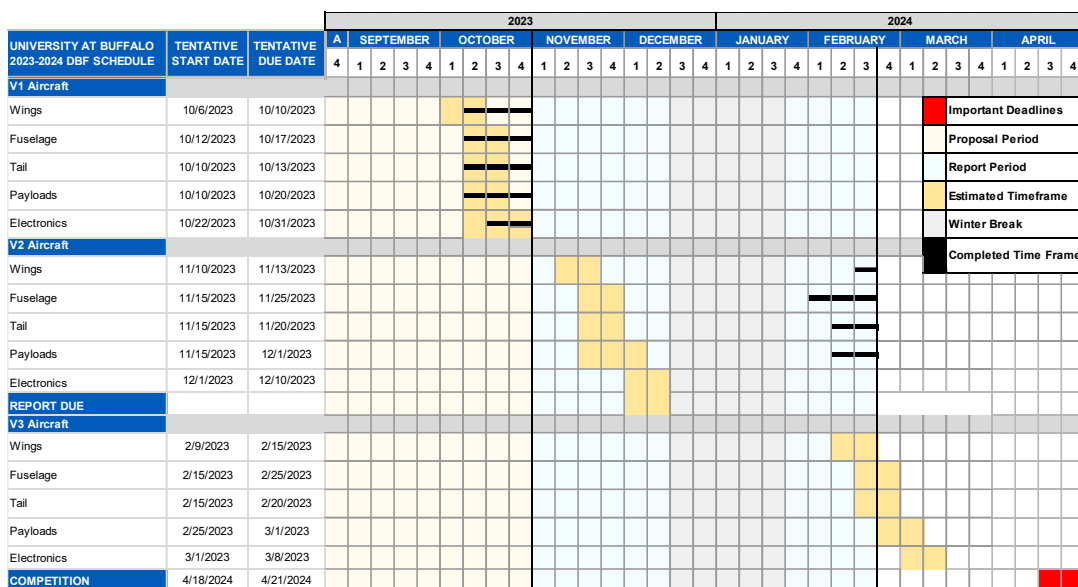


Figure 36. Manufacturing Plan

6.1 Manufacturing Processes Investigated

6.1.1 3D Printing

3D printing was the method of choice to generate dimensionally accurate parts for the implementation of solution systems and manufacturing processes. This method of manufacturing was used to create multiple prototype solutions, structural elements for the tail, and functional parts such as servo hardpoint covers, inserts, MSC, and others. Additionally, 3D printing allowed to generate molds for composite manufacturing easier than using a CNC router or hand tools while also being more accessible. In these cases, 3D printing is used due to the ability to rapidly produce prototype molds resulting in a lower man-hour alternative to more conventional manufacturing processes.

6.1.2 CNC Hot Wire

A homemade 4-axis CNC hot-wire foam cutter is used to cut airfoil profiles out of XPS foam using open-source programs Wing Wire and grbl to generate cut paths and g code respectively. Hot-wire cut foam parts are used as wing core material, which are then laminated with a composite layup. This method is dimensionally accurate, easy to repeat, and very reliable. Similar results can be achieved with a hand-held hot-wire bow, which is used to core out small parts of the wings with laser-cut stencils that provide a path for the wire to follow.

6.1.3 Composite Vacuum Bagging

Vacuum-bagged composites (Fig. 37) offer the capability to create high-strength-to-weight and durable fiber-reinforced polymer (FRP) composites. Fiber reinforcement materials used include fiberglass, Kevlar, and carbon fiber. The polymer matrix used in composite layups is PRO-SET LAM-145 Thixotropic Laminating Resin. The process requires expendables such as layers of peel/ply, breather/bleeder, and bagging. While this process is laborious and relatively expensive when compared to manufacturing comparable parts out of materials such as PLA or plywood. This process is justified as the strength-to-weight ratio of the resulting parts is far superior in comparison to conventional manufacturing methods.



Figure 37. Vacuum Bagging Setup

6.1.4 Foam Core Composites

Foam core composite manufacturing can produce high-load structures at a relatively low weight, with high durability FRP composite forms. Foam cores are produced via CNC hotwire cutting operations. A fiber reinforcement layer can be bonded to a dimensionally accurate foam core using epoxy resin and a vacuum bag to apply an even, high pressure. In addition, unique integrated structures are available such as active Kevlar hinges. Therefore, foam core composites are primarily used by Goose Works to manufacture wings and control surfaces.

6.1.5 Composite with Negative Mold

Alternatively, negative molds may also be used to lay composites. The mold can be tailored to the desired geometry which is then 3D printed and sanded for a smooth surface finish. Then, composite vacuum bagging can be done as described in Section 6.1.3. However, precautions must be taken to ensure that the corner of the mold does not puncture the vacuum bag or the molded part mistakenly adhere to the mold.

6.1.6 Laser Cutting

Balsa and plywood parts are cut to shape using a CNC laser cutter. A 3mm Baltic Birch Plywood is used for structural parts that must be lightweight, where composites would not be practical. The primary uses are to manufacture ribs, bulkheads, flap hinges, and hardpoints for screws.

6.2 Manufacturing Process

6.2.1 Wings

The wings were manufactured with foam core composites and plywood ribs, interfaced with a hexagonal carbon fiber spar and tip-mounted end plates. Foam wing sections were cut by a hotwire CNC and then glued together with laser-cut wooden ribs. Kevlar hinge sections were laid up underneath the carbon skin for the flaps and ailerons. The foam cores were wrapped in a single layer of 67 GSM 3K carbon fiber laid up at 60° with respect to the edge of the airfoil. 3D-printed flap hinges were attached to the main wing section from hand-routed sections with insert balsa attachment points. The servos for both the aileron and flaps (Fig. 38) are similarly mounted via hand-routed sections and then constrained with 3D-printed covers.



Figure 38. Hand Routing

6.2.2 Fuselage

The fuselage is a carbon fiber body manufactured using custom 3D-printed negative molds (Fig. 39) along with a carbon and plywood sandwich subframe (Fig. 40). Two separate molds were made for the left and right side, and they were joined using methyl methacrylate. The wings were attached via the wing hinge mechanism and a carbon fiber spar to elevate the excess loading of the hinges. The subframe is attached to the skin of the fuselage via 3D-printed ASA interface pieces and then attached with epoxy to the skin. The completed sub-frame and skin assembly are then prepared for a vacuum bagging operation. A base layer of epoxy resin is applied to the surface and cooled until it reached the B-stage, where it begins to get tacky. The hatch reinforcement and Kevlar hinge are pressed onto the base layer before a single layer of fiberglass type is applied and wetted, followed by vacuum bagging. After curing, trimming, and cutting the hatch, the surface is sanded before applying a spray-on UV protective coating.



Figure 39. Negative Mold



Figure 40. Fuselage with Ribs

6.2.3 Empennage

The production of the empennage was completed using the same process that was used for the wings where the empennage surfaces are primarily CFRP-wrapped foam cores utilizing Kevlar hinges to attach their respective control surfaces. The elevator is connected to the rudder with a custom 3D-printed insert as shown in Fig. 41. The insert is laid up along with the foam cores and is adhered to both the carbon fiber skin and foam core. The included threaded holes of the insert allow for the elevator to be rigidly attached to the rudder using M3 x 0.5 mm machine screws. Consequently, this allows the elevator to be removed for ease of transportation and easily installed on the site of the competition. The empennage is securely mounted to the fuselage tube using a carbon fiber flange which is adhered to the fuselage using methyl-methacrylate glue.



Figure 41. Elevator to Rudder Adapter Mount

6.2.4 Folding Wing Connection

The connection point for the folding wings will be a hinge made of hexagonal telescopic carbon. The corresponding inserts will be 3D printed for the hinge mechanism as depicted in Fig. 42. Manufacturing the hinges in this way leverages the accurate reference surfaces of the telescopic tubes to facilitate accurate wing alignment with the fuselage. It also ensures the smooth operation of the translational component of the hinge assembly. A hole was drilled in the fuselage side of the telescopic tube, and a spring clip was mounted to retain the wings in their flight configuration.

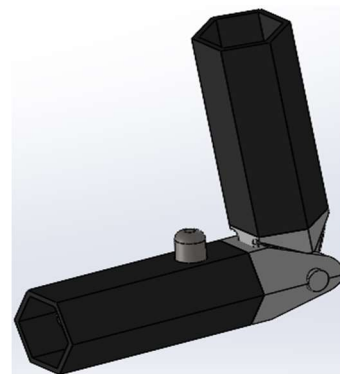


Figure 42. Hinge Mechanism

6.2.5 Nacelles

The nacelles were manufactured using a vacuum-bagged carbon fiber layup process in a negative 3D-printed mold. The layup was performed with 4 layers of 200 GSM 3k carbon fiber each layer rotated by 45°. A balsa wood plate was sandwiched between the 2nd and 3rd layers of carbon fiber to add rigidity for the motor to mount. After the layup was finished, trimming and sanding was done to provide a smooth surface finish (Fig. 43). They are connected to the surface of the wing primarily by a glue joint and optionally by bolts interfacing the wing via plywood hardpoints.



Figure 43. Nacelle Before and After Finishing

6.2.6 Mission Inserts

The mission inserts will be manufactured primarily using an FDM 3D printing technique as it can create complex geometries out of semi-flexible plastic which allows for a degree of compliance in the inserts. The slight deformation of the M3 insert facilitates the frictional fit of the loaded passengers into the insert; this is made possible by our ability to process flexible plastic materials through 3D printing. For M2, the 3D printed gurney and EMT inserts allow for rapid production based on changes in the manufactured fuselage.

6.2.7 Hatch Installation

The hatches were cut out of the fuselage using a dremel equipped with a diamond abrasive cut-off wheel. Due to safety reasons, the carbon fiber was cut with water to mitigate the possible inhalation of carbon dust. The hatches were attached to the fuselage utilizing Kevlar hinge joints, bonded with epoxy. During the bonding process, uniform pressure was then applied on the joint until curing using a vacuum bag, improving operation and the strength of the hinge.



Figure 44. Fuselage

7 Test Plan

7.1 Test Objectives

Table 24 outlined the tests conducted, their purposes, and methods, to guarantee all system requirements were fulfilled allowing for the continuation of the development of a durable and reliable aircraft. Systems tests (ST), GM tests (GMT), and flight tests (FT) are all provided.

Table 24. Test Objectives

Test Number	Test	Purpose
ST-1	Wing Load	Validated wing strength compared to expected wing loads at competition
ST-2	Material	Identified material to use for composite structures based on material properties
ST-3	Static Thrust	Optimized propulsion system with thrust, power consumption, battery, motor, and propeller sizing
ST-4	Wind Tunnel	Validated CFD analysis from ANSYS
ST-5	Payload	Conducted test to analyze tray-constraint system durability and ensure compliance with SSR-1:SSR-11
GMT-1	Ground Mission	Conducted test to optimize GM score
FT-1	V1 Flight	Conducted test for proof of flight and mission performance for M1 and M3
FT-2	V1 Flight (FPV)	Collected in-flight data for all missions to influence V2 design considerations
FT-3	V2 Flight (FPV)	Collect in-flight data for all missions to influence V3 design changes and check rule compliance
FT-4	V3 Flight (FPV)	Perform all flight tests and GM and ensure compliance with all rules and regulations for competition

7.2 Systems and Ground Test

Systems tests were conducted to validate and establish a greater understanding of the aircraft's structural integrity, ground handling, and flight characteristics prior to taking flight to ensure the safety of the crew involved. Ground tests were performed to optimize the ground mission along with handling on the ground.

7.2.1 Wing Load Test

Due to the probability of high winds occurring in Wichita, Kansas, in addition to high G turns, a wing load test was conducted based upon Section 4.4.1 to determine whether the wings are strong enough to withstand these loads acting on the aircraft during flight. The XPS foam wings with carbon fiber composite skin were manufactured and tested by applying a load incrementally on the body while fixed at the wingtips. As weight was added the deflection from the neutral position was recorded until failure.

7.2.2 Five-Minute Staging Test

Two tests were developed to demonstrate the reliability and simplicity of the folding wings. The first test was for the hinge connection securing the wings in place to prevent unwanted movement or unhinging during flight. The second is to test the time allocated for taking the wings from the parked configuration to the flight configuration. To test the ability to fold the wings up and down a stopwatch was used to time how long it takes to change from the parking configuration to the flight configuration and then back to the parked configuration first. Then the final test was to perform the entire operation of changing from the parked configuration to the flight configuration under the 5-minute staging window.

7.2.3 Material Test

Strength-to-weight ratios of various composite samples were analyzed through material testing to determine what structural components should be utilized for the fuselage ribs. The test samples included balsa and plywood sandwiched with 1.89oz/67gsm and 6.5oz/200 GSM carbon fiber. Samples were about 150x14x3 mm and were weighed prior to stress testing. The composites were assessed through a 3-point bending test, ASTM D6416 (Fig. 45), and data was collected to identify the mechanical properties of each sample. and which composite should be utilized for the fuselage ribs.

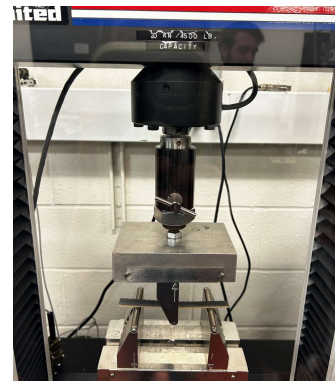


Figure 45. 3-Point Bend Test

7.2.4 Static Thrust Test

The static propulsion test was conducted to verify our predicted propulsive performance. To emulate real conditions, power and thrust data was recorded for a single motor and propeller in partial loading condition at 100% throttle and full battery charge. The motor was fastened to a thrust test stand with a calibrated load cell connected to a 24-bit precision ADC (HX711) and Arduino Uno. The voltage, current, power, and thrust readings were taken from a wattmeter and the Arduino serial monitor in real-time.

7.2.5 Wind Tunnel Test

The University at Buffalo has a low-speed, Eifel-type tunnel constructed by Engineering Laboratory Design, Inc. with a test cross section of 30.48 cm x 30.48 cm that is 60.96 cm long. A model of the aircraft was developed and tested in the wind tunnel at various airspeeds and AOA to simulate speeds ranging from takeoff to cruise. The purpose was to validate the CFD analysis discussed in Section 4.3. Fig. 46 shows the scaled model inside the tunnel connected to the rod in the center of the wind tunnel.



Figure 46. Wind Tunnel Test

7.2.6 Flap Hinge Location

Flap hinge geometry was determined for a deflected position of 37 degrees. The flap cove shape and the hinged travel range was iteratively designed via cardboard modeling (Fig. 47), SolidWorks, and Ansys Fluent.



Figure 47. Flap Hinge Geometry

7.2.7 Payload Durability and Constraint Test

All payload components must remain in their initial orientation upon landing the aircraft. Each constraint systems is functionally tested to ensure all payloads are safely secured. Each cargo is loaded onto Condor, and then the aircraft will be turned upside down, shaken, and rotated in an orientation such that a “rough flight” is simulated. All parts must remain in their respective part of the tray insert to pass the test. The test is conducted prior to FT to ensure the safety of those on the ground.

7.2.8 Ground Mission

The GM test was conducted to optimize the time necessary to start in the parked configuration and to cycle through the loading and unloading processes. To optimize timing for GM, several team members performed the operation 3 times. The candidate capable of performing the mission in the timeliest manner was selected to carry out the mission. A stopwatch was used, and the scenario outlined in Section 3.1.2 was followed to simulate the real-life competition scenario. The data collected was used to discuss and refine issues for the redesign process from V1 to V2 and the data from V2 will influence V3, the competition prototype.

7.3 Flight Test

FT was conducted to identify variability between theoretical and physical performance to further the development of Condor. To document the aerodynamic and structural envelope, FT-2 through FT-4 were designed to employ the FPV system specified in Table 25. The angular rates, inertial forces, airspeed, and battery power draw were the primary data used for improving analysis methods, identifying faults, and tuning mission performance.

Table 25. FPV Data Collection System Specifications

Component	Location	Purpose
Speedybee F405 Fixed Wing with 16 GB MicroSD Card	Near CG	Sense and log voltage, current, angular rates, inertial loads, wind, barometric altitude, attitude, and control input data
Matek ASPD-4525 Airspeed Sensor	Nose	Measure airspeed
WS-M181 GPS Module	Plane top	Collect Global Positioning System satellite data
Caddx Ratel 2 Analog Camera, RDQ Mach 3 Video Transmitter	Horizontal tail	Observe airflow during takeoff and landing
Fatshark Analog Goggles	Ground	Video recording and backup data logging

7.3.1 Test Schedule

Table 26 outlines the specific dates that each test was and will be conducted before the 2023-2024 DBF competition.

Table 26. Test Dates and Objectives

Test	Date	Objective
Ground	11/4/2023	Analyzed landing gear, ability to maneuver control surfaces, and performed GM
Ground	11/11/2023	Verified motors and servos were capable of full throttle and maximum deflections
Flight	11/11/2023	V1 maiden flight test for controllability, flight stability and landing
Flight	11/11/2023	V1 test flight for performance capability, M1, and M3
Flight	2/10/2024	V1 flight collected FPV data for all flight missions
Ground	3/1/2024	Analyzed V2 landing gear, ability to maneuver control surfaces, and performed GM
All	3/1/2024	V2 maiden flight test for controllability, completed of all flight and ground missions, collected FPV data
All	3/8/2024	V2 flight to test performance envelope and identify improvements for V3
Ground	3/8/2024	V3 ground test for ground handling, propulsion system check, and control surface deflections
Flight	3/29/2024	V3 maiden flight test to verify controllability
All	3/29/2024	V3 performance of all missions
All	4/6/2024	Mock competition performed and results compiled
All	4/8/2024	Conservative practice for competition
All	4/12/2024	Practice for competition

7.3.2 Flight Test Plan and Checklist

Preflight checks and inspections were conducted to ensure the safety of each flight before takeoff as summarized in Table 27. An 'X' was used to represent the successful completion of each pre-flight task.

Table 27. Preflight Checklist

Pre-flight Inspection			Pre-flight Controls		
Item	Check	X	Item	Check	X
Wings	Locked and secured		RX Switch	Powered on	
Tail	Secured		RX/TX	Range check confirmed	
Landing gear	Secure & damage free		Failsafe	Correct	
Fuselage	Secured		Control Surfaces	Correct deflections & trim	
Nacelles	Mounted & secured		Control Inputs	Correct	
CG	Balanced at CG mark		Arming fuse	Armed	
Wires	Connected & secured		Motors arming	Armed	
Propulsion battery	Strapped, plugged in & fully charged		Takeoff area	Clear	
Avionics battery	Strapped, plugged in & fully charged		Propulsion run-up	Completed	
Servos and linkages	Secured & functioning		FPV system	On & logging	
Motors	Rotating inboard		Post-flight Inspection		
Propellers	Damage free & nut tightened		Item	Check	X
Crew insert	Secured		FPV system	Off	
Insert	Fastened		Motor arming	Disarmed	
Hatches	Closed & latched		Arming fuse	Off	
Wing tip load test	Passed		Batteries	Unplugged	
Wind direction	Verified for takeoff		Visual inspection	No or minimal damage	

8 Performance Results

8.1 System Performance

8.1.1 Material Test

Upon analysis of the strength-to-weight ratios, shown in Table 28, and stress vs. strain diagrams (Fig. 46) of various composite samples, the team opted to manufacture the rib structure 67 GSM carbon fiber sandwich composite with plywood. Furthermore, it was determined to align the fibers at 90 degrees because at 45 degrees the carbon fibers pulled out of the matrix causing delamination with the epoxy. Meanwhile, epoxied at 90 deg, the load is evenly distributed along the length of the test sample at higher loads. The 200 GSM carbon was not chosen due to having a lower strength-to-mass ratio compared to the 67 GSM.

Table 28. Strength-to-Weight Ratios

Core Material	GSM	Fiber Orientation (degrees)	Yield Stress (ksi)
Plywood	200	90	29.37
	67	90	22.94
		45	16.76
Balsa	67	90	13.82
		45	6.41

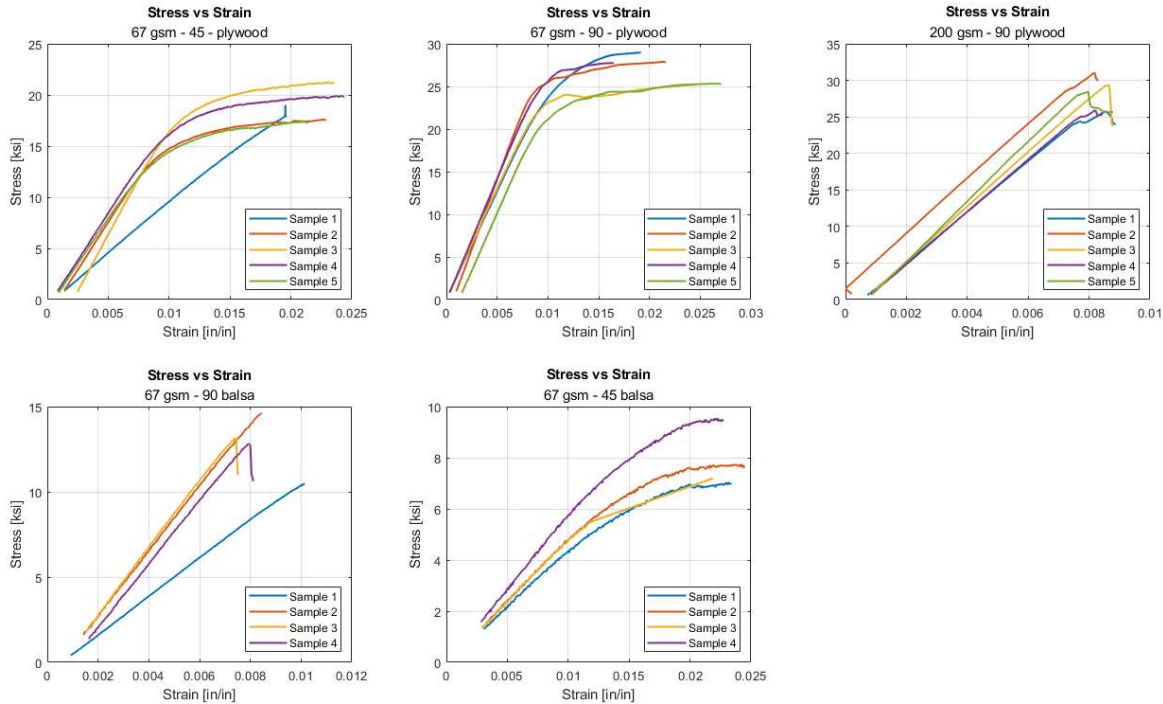


Figure 46. Stress vs Strain for Composite Samples

8.1.2 Static Propulsion Test

The static propulsion testing results, presented in Table 29 for full loading, closely matched the preliminary performance predictions. None of the components overheated. Additionally, the 100 A fuse was supplied with a constant load of 120 A to determine its operating time limit above max rated current. The battery ran out of energy before the fuse melted. Therefore, the “17x10E” and “17x10EP” propellers were favored.

Table 29. Static Propulsion Test Results

Propellers	Thrust (lbf)	Power (W)	Voltage (V)	Current (A)
16x10E and 16x10EP	21.45	2770.8	22.60	122.6
17x10E and 17x10EP	23.08	3146.4	21.85	144.0

8.1.3 Wind Tunnel Test

The wind tunnel testing results, presented in Table 30 for the scaled tail, body sizing, and configuration, gave similar results to ANSYS simulations. Two sets of tests were conducted. One for a 10-degree AOA simulating takeoff conditions, and one for a 0-degree AOA simulating cruise conditions. The take-off scenario was tested for 5 different low airspeeds, meanwhile, the cruise configuration was tested over a wider range.

When the wind velocity reached 40 mph in takeoff condition, the lifting force generated by Condor’s tail overcame the frictional force of the holding mechanism. That caused Condor’s body to pitch downwards and stabilize at about 2 degrees geometric AOA.

The same behavior occurred when repeating the test multiple times, showing that a negative pitching moment was produced by the tail, effectively demonstrating the longitudinal stability that was recorded from the AVL software. The second set of tests conducted for a 0-degree AOA demonstrated Lift and Drag forces for cruise conditions with up to a maximum airspeed of 118 mph. Fig. 47 shows that at both takeoff and cruise configurations, the Lift-to-Drag ratios peaked at low velocities, and then stabilized at higher velocities for the cruise condition.

The data resulting from the wind tunnel testing served to corroborate the aerodynamic performance prediction analysis presented in Section 4.5.

Table 30. Wind Tunnel Data

AOA	Velocity [ft/s]	Lift [lbf]	Drag [lbf]	L/D
10	80.6946	0.0009	0.0016	0.5624
10	113.7060	0.0022	0.0034	0.6562
10	146.7174	0.0040	0.0055	0.7382
10	173.8601	0.0065	0.0082	0.7929
10	212.7402	0.0076	0.0114	0.6693
0	80.6946	0.0000	0.0014	0.0000
0	113.7060	0.0013	0.0032	0.4218
0	146.7174	0.0022	0.0048	0.4687
0	179.0069	0.0034	0.0071	0.4762
0	212.0587	0.0036	0.0098	0.3662
0	245.1104	0.0038	0.0126	0.3042
0	278.1622	0.0049	0.0162	0.3050
0	311.2140	0.0067	0.0199	0.3394
0	344.2658	0.0081	0.0242	0.3343
0	377.3176	0.0101	0.0286	0.3543

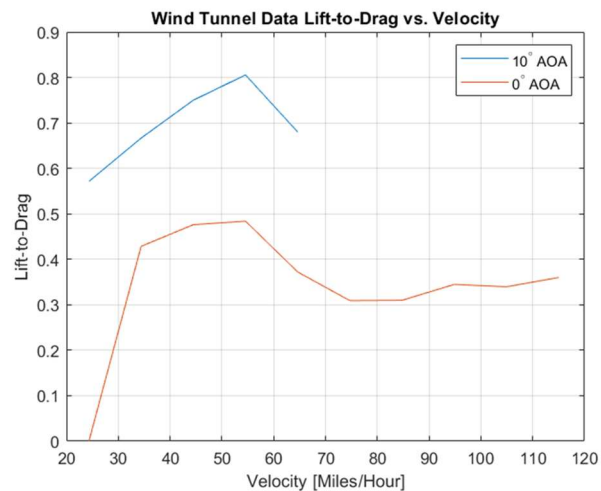


Figure 47. Wind tunnel Data

8.1.4 Ground Mission Performance

When performing the GM for V1 (Table 31), there were rigidity and volumetric efficiency issues with the XPS foam inserts that forced the change to 3D-printed inserts. Additionally, when lowering the wings from the parked configuration, the utilized plastic hinges mounted to the fuselage wall with bolts were not sufficient for securing and holding the wings in the flight position, forcing the change to the internal folding.

Table 31. Ground Mission Performance Test 1

Ground Mission			
	Stage 1	Stage2	Stage3
Time [s]	132	360	110
Total Time [s]	602		

8.2 Flight Test

8.2.1 Results

FT-1 flight performance is documented in Table 32, where M1 was completed successfully before failing to land M2. The observed takeoff distance, mission time, and battery endurance (99.9 W-h) were substandard.

Table 32: FT-1 Flight Performance

Mission	Weight	MSC Weight	Takeoff Distance	Lap Time	Lap count	Mission Time
M1	11.95 lb	-	19.5 ft	33 s	3	101 s
M2	16.54 lb	4.4 lb	34 ft	38 s	3	120 s

For FT-2, landing concerns due to high wind-restricted testing to M1. Many factors contributed to the high average power draw and low airspeeds on V1: high parasitic drag, high roll, and pitch rates, elevator input required to sustain level flight, and stalled airflow at the tips because of the lift distribution associated with inboard-spinning propellers [14].

Table 33: FT-2 Flight Performance

General		Propulsion		Structural	
Average Velocity	45.3 ft/s	Max. Power	2355.9 W	Weight	12.42 lb
Stall Speed (Flaps)	34.5 ft/s	Avg. Power	1093.2 W	Max. Bank Load	4.6g
Theoretical Top Airspeed	84.6 ft/s	Endurance	297 s	Max. Landing Load	3.6g
Max. Number of Laps	6	Climb Angle	30.4°	Avg. Wind Speed	13.09 ft/s



Figure 48. V1 Test Flight

8.2.2 Improvements

To enhance takeoff performance, V2 was designed to facilitate rotation, the flaps were redesigned with ANSYS for better flap cove geometry, and the propulsion system was sized for 20% greater static thrust. Inboard spinning propellers were also kept centering the lift distribution at the flaps for takeoff with a 3° geometric twist and taper added to counter the stalled airflow at the tips. To further improve in-flight performance, the fuselage was narrowed to reduce drag, and the required control surface trim was verified.

V1 didn't land M2 during FT-2 because the nose gear support was ripped off the plywood bulkhead that it was mounted to; the load was likely greater than the effective 44.8 lbf load recorded during FT-2. As the nose gear structure on V2 will also be mounted to the bulkhead, the bulkhead was designed to have more layers. Wing loading was never a concern.

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