FINAL DESIGN REPORT
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AIRPLANE NAME - UDAAN

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Nomenclature:

α	Angle of attack
α CL=0	Angle that lift coefficient is zero
Cl	Airfoil lift coefficient
Clα	Airfoil lift curve slope
Cm	Airfoil pitching moment coefficient
Cm0	Airfoil pitching moment coefficient at zero angle of attack
Cmα	Airfoil pitching moment curve slope
CL max	Wing maximum lift coefficient
CL α=0	Wing Lift coefficient at zero angle of attack
CD,0	Drag coefficient at zero angle of attack for womg
W/S	Wing loading
T/W	Thrust-to-weight ratio
AR	Aspect ratio
V stall	Stall velocity
V cruise	Cruise velocity
V takeoff	Takeoff velocity
V landing	Landing velocity
W	Airplane weight
e	Oswald efficiency factor
CD_0	Zero-lift drag
η	Propeller efficiency
$V_{Stall\ Altitude}$	Stall speed at specific altitude
V_{LO}	Take off speed
V_T	Landing speed
$V_{Min\ PR}$	Minimum power required speed
PA	Power available
D	Propeller diameter
W_P	Propulsion system weight
V_{Max}	Maximum speed
V_{Cruise}	Cruise speed
μ_r	Ground friction coefficient
ϕ	Ground effect
Q	Dynamic pressure
TA	Thrust available
TR	Thrust required
RPM	Revolutions per minute
ESC	Electronic speed controller
KV	RPM per Volt
V	Volt
mAh	Milliampere-hour
lb.	Pound
oz.	Ounces
ft.	Feet
s	Seconds
ft/s	Feet per second
W	Watts
In	Inches
A	Ampere (Amp)
$C_{M,\alpha}$	Pitching coefficient due to AOA
CM,0	Pitching Moment Coefficient at Zero
MAC	Mean Aerodynamic Chord

AC	Aerodynamic Center of the wing
δ_e	Elevator Deflection Angle
δ_a	Aileron Deflection Angle
δr	Rudder Deflection Angle
$C_{M,\delta e}$	Pitching coefficient due to Elevator Deflection
$C_{y,\beta}$Side	Force Coefficient due to sideslip
$C_{n,}\beta$	Yawing Coefficient due to Sideslip
$C_{l,\beta}$	Rolling Coefficient due to Sideslip
$C_{n,\delta r}$	Yawing Coefficient due to Rudder Deflection
$C_{l,\delta r}$	Rolling Coefficient due to Rudder Deflection
$C_{y,\delta r}$	Side Force Coefficient due to rudder deflection
$C_{l,p}$	Rolling Coefficient due to Roll Rate
$C_{n,p}$	Yawing Coefficient due to Roll Rate
$C_{y,\delta a}$	Side Force Coefficient due to aileron deflection
$Cl_{,\delta a}$	Rolling Coefficient due to Aileron Deflection
$C_{n,\delta a}$	Yawing Coefficient due to Aileron Deflection
$C_{y,r}$	Side Force Coefficient due to yaw rate
$C_{l,r}$	Rolling Coefficient due to Yaw rate
$C_{n,r}$	Yawing Coefficient due to Yaw Rate
S&C	Stability and Control
SM	Static Margin
Ksi	Kilo Pounds per Square Inches.
Psi	Pounds per square Inches.
V	Shear Force
M	Internal Moment
σ_max	Maximum Bending (Normal) Stress
w_0	Uniformly Distributed load on the Wing.

I)Mission Description:

The mission is to design and validate an electric remotely controlled aircraft that will fly a specified course while carrying as many simulated bombs (tennis balls) as possible. The bombs will then be dropped on a specified target zone.

The Course will be contained in a flying area with dimensions of 400 feet by 100 feet and an altitude ceiling of approximately 100 ft. The course will be oval in shape and one lap will consist of two 180 degree turns a minimum of 300 ft apart. To complete the mission, a team must fly two laps, a third lap when the payload will be dropped, and then a final fourth lap before landing. The course layout can be found in figure I.1 .

As mentioned previously, the payload of simulated bombs will be dropped on a target zone. Each aircraft will be required to carry a minimum of 4 tennis balls, but there will be no maximum allowable amount. Dropping the payload will occur during the third lap. The target zone will be an area marked off with a size of 10 ft by 10 ft. Each tennis ball that first lands on the target will be scored. Those that roll or bounce on to the target after first contacting the ground will not be counted towards the competition score. Once the combat mission is complete, the plane must complete the final lap before landing. The mission profile can be found in figure I.2 .

One modification was made to the mission profile to be more competitive. That change is to, during the combat lap, instead of a conventional bomber, a dive bomber will be utilized to drop the ordinance. This is mainly dictated by the small time over target and the small target zone.

II) Competitive analysis

The equation for the DVC score is

$$DVCSCR = \left[\frac{NBOT}{WEFR}\right] + \left[2 \times \left(\frac{V_G}{C_A}\right)\right] \quad \text{(Eqn.1)}$$

Using Kline-McClintock method, sensitivities of individual parameters is calculated and reflected in Eqn.2

$$WR = 0.446 = \sqrt{0.1936 + \left(2.42 \times 10^{-3}\right) + \left(1.6 \times 10^{-3}\right) + \left(1.024 \times 10^{-3}\right)} \quad \text{(Eqn.2)}$$

The analysis determines that NBOT is the most important, followed by WEFR
The nominal values, uncertainties and sensitivity parameters are shown in table II.1

Excel program is developed to calculate all possible flight scores with a change in any parameter and the results are shown in Figure II.1

The plot shows that a small change in number of tennis balls can cause a huge impact in the % change of the total score

The second driving factor is the weight; a small % increase in weight can reduce the total score by a huge amount.

25% change was applied to each parameter and the variation of % change in parameters with the % change in total score is shown in Table II.2

III) Concepts and Configurations Considered

The first step of the aircraft selection process is to render various conceptual designs that are thought to be viable options for completing the specified mission[1]. Some concepts take many aspects of the mission into account, while others very few. During the screening process, this becomes evident, but all concepts are screened. They are as follows:

Concept I (Figure III.1.A):	A blended fuselage with many complex curves, mid wing, t-tail, single tractor
Concept II (Figure III.1.B):	Mid wing, standard tail configuration, single tractor, simple square design
Concept III (Figure III.1.C):	Twin tractor, Bi-Plane, standard tail configuration, rounded fuselage
Concept IV (Figure III.1.D):	Pusher-Puller, mid-wing, twin boom, downward facing twin vertical tails
Concept V (Figure III.1.E):	Lifting Fuselage, twin boom, twin tractor, box tail
Concept VI (Figure III.1.F):	Twin tractor, twin boom, box tail, central fuselage, high wing
Concept VII (Figure III.1.G):	Flying wing with empennage, twin boom, box tail, twin tractor
Concept VIII (Figure III.1.H):	Low wing, large central fuselage, twin boom, box tail, single pusher

IV) Concept Weighting and Selection Process:

Once initial concepts are submitted, all are subjected to a screening process. This does not assign a weight to each criteria, or a value to each concept's score. Instead, each is scored with either a (0) , (+), (-) as it compares to a given reference aircraft[2]. Once this is complete, those concepts can be further pursued further, altered and re-screened, or dropped completely. Our screening results were as shown in Figure IV.1 with those concepts being pursued further being shown in yellow. The reference aircraft is a B-27 Mitchell, a conventional bomber from WWII[3].
The next step is to score each concept. In this process, the criteria are narrowed to only include those deemed most important, and each is given a weight relative to importance. Also with this, each aircraft is ranked as compared to the reference aircraft on a scale of 1 to 5, with 5 being much better than the reference, 3 being equal to the reference, and 1 being much worse than the reference[2]. The highest weight value was given to the payload accessibility criteria, with the next highest criteria being durability. This is due to the mission specifying the need for a dive bomber. The scoring results are shown in Figure IV.2 with the final concept selection shown in yellow.

V) Justification of Selected Concept

The mission should be the driving factor influencing the selection of a concept[5]. The mission profile has been altered to support a more competitive concept by adding the requirement that the aircraft be able to withstand high g-loads experienced by a dive bomber. This has a large influence on the scoring by adding durability, or the ability to survive sustained high g-loads and possibly rough landings, to the critical design parameters. The other necessity for a dive bomber and in some cases more important is the requirement to hold a large number of bombs that have the ability to be dropped easily and efficiently. The concept chosen was a ground up design. That is, as a group the members started with the payload and took the best attributes from other concepts to combine them into a much more competitive design. The mission was not altered to fit the airplane, but instead the airplane modified to fit the mission. This approach created a mission oriented aircraft whose strengths are the critical aspects of the mission. Being a flying wing with a spar that carries completely through the structure, it results in a very durable aircraft able to withstand the g-loads experienced by a dive bomber. The most important trait of a bomber is the bomb capacity. With a large center-section, this concept has the ability to carry a large number of bombs internally. After furthering the analysis on the concept once it was selected, it became clear that several aspects of this design were underscored, namely: simplicity and ease of manufacture. This further proves that the current concept designed as a mission oriented aircraft is the most viable concept. The final concept that was selected can be shown in figure V.1 as originally drafted.

VI) Mission Derived Requirements and Constraints

- Must carry at least 4 simulated bombs (tennis balls)
- Must be able to complete at least 4 full laps of the course
- Aircraft wingspan must be greater than or equal to 3 feet
- Propulsion battery pack weight can be no more than 1 Lb.
- Only NiMh batteries may be used to power the aircraft
- Current draw to the propulsion system is limited by a 40 Amp slow-blow fuse
- Must be able to withstand high G-Loads due to bombing method being a dive bomber
- Must be able to handle wind gusts[5]
- Internal Payload

Fundamental design and analysis results

Section 1) Aerodynamics

1.1) Conceptual Design

Conceptual aerodynamics design focuses on figuring out the reasonable value of aerodynamic parameters (i.e. Aspect Ratio, Wing Span, Wing Surface Area, and Stall Velocity) and the wing airfoil.

1.1.1) Aerodynamic Parameters

1) Wing Surface Area: The wing area is the key parameter for the stall velocity, wing loading, and total lift. Increasing wing area increases total lift and decreases wing loading (W/S), which reduces stall velocity. This enables a lightweight propulsion system due to low thrust-to-weight (T/W) requirements. (Figure, 1.1 and Figure, 1.2)
2) Wing Span: The wing span is a key parameter for aspect ratio (AR). Higher the wing span increases the aspect ratio. (Figure, 1.3)
3) Aspect Ratio: The aspect ratio affects the $C_{L\alpha}$ and $C_{D\alpha}$. According to Prendtl's lifting-line equation[7] and Helmond's equation[7], higher aspect ratio increases $C_{L\alpha}$ and possibly decreases $C_{D\alpha}$.

1.1.2) Decided Desired Value

During the process of deciding values, aspect ratio and surface area were the key parameters. Having the larger surface area decreases the stall velocity and increases the take off and cruise lift. However, it also makes aspect ratio lower. Having long wing span is not desired for the reason of structure. Moreover, thickness ratio (t/c) is another important parameter due to the airplane model. According to Figure,1.3, there is no possibility to have more than 12ft^2 for surface area, and it is not desirable due to long wing span. So, the final decision is made by pushing down the t/c with tapering the wing.

1.1.3) Airfoil Selection

Important Criteria

Due to the selected design and the DVC mission, our wing airfoil needs to meet following requirement.

- Thickness: According to our design, the tennis balls will be stored inside of the center section of the wing. This means that the chosen airfoil must have sufficient thickness throughout the chord to house the simulated bombs. The airfoil thickness will directly affect wing span, surface area, and aspect ratio.
- Bottom flatness: For the dropping mechanism, the bottom of the airfoil needs to be flat for the construction of the payload doors.
- Coefficients: Lift, drag, and moment coefficients must be reasonable for a high efficiency bomber

Selection Criteria:
Following parameters are used to select airfoil from a broad list of airfoils by NACA, SD, Eppler, MH, and GOE airfoils. These parameters are chosen to satisfy aerodynamics, S&C, and structural requirements and constraints. XFOIL 172, a 2D panel method code, is used for getting the coefficients data.

- Maximum Lift Coefficient, C_{lmax}> 1.2: This parameter is set based on conditions imposed during wing sizing to meet takeoff requirements. Higher $C_{l\,max}$ is desired for lower V_{stall} and weight load (W/S). (Figure, 1.1)
- Lift Coefficient at Zero Angle of Attack, $C_{l,0}$> 0.3: This parameter is set based on conditions imposed during wing sizing to meet cruise requirements. Higher $C_{l,0}$ is desired for lower V_{cruise}. This will be needed to reduce number of battery packs.
- Airfoil Pitching Moment Coefficient Curve Slope, $C_{m\alpha}$< 0: A well-balanced airfoil reduces trim drag through small required elevator deflections.
- Zero-lift Drag Coefficient, $C_{d,0}$< 0.02: An important parameter to reduce profile drag.
- Thickness Ratio, t/c > 15%: Higher ratio required to store tennis balls and minimize chord length and surface area.
- Lower Surface Flatness: Curved and under-cambered lower surface are not desired to store tennis balls stably.
- Manufacturability: Sharp leading edge curvatures and under-camber are difficult for balsa sheeting and for a smooth heat-shrink covering, respectively.

Based on the parameters above, four airfoils are chosen for further analysis: ARA-D 20%, GOE 382, MH 82, and NACA 4418. To compare these airfoils, NACA 2414 is selected as a reference.

1.1.4) Aerodynamic Modules

The wing lift and drags are calculated by using Prandtl's lifting-line theory[7] and using the data from U.S. standard atmospheric model[6] based on the elevation of 1300ft concurrent with conditions experienced in Wichita, KS. (Figure, 1.4 and Figure, 1.5) The aircraft Oswald efficiency factor(e) is assumed to be 0.75.

1.2) Preliminary Design

Preliminary aerodynamics design mainly figure out the value of aerodynamic characteristics (i.e. Wing Span, Wing Surface Area, Aspect Ratio, CL, CD, Cm, and Aerodynamic Center), and takeoff, turning, and landing performance. In addition, there is a change needed from conceptual design.

1.2.1) Changes from Conceptual Design

Due to lack of confidence of the airfoil data, different analysis methods must be used, including no longer using XFOIL[8], which caused a change in the airfoil being used from GOE 382 to NACA 4418. The main reason being there is no published data available for GOE 382, ARA-D 20%, and MH 82, but there were studies conducted by NASA involving the NACA 4418 at low Reynolds Number thus facilitating the change.According to this change, the airfoil thickness has reduced 2% (20% to 18%), C_{lmax} has reduced 0.2 (1.7 to 1.5), and $C_{l,\,\alpha=0}$ has reduced 0.13 (0.58 to 0.45) for the airfoil. Despite these changes, with some having negative effects, altering the proposed wing planform will compensate for this. The wing planform area changed from 611.25in^2 to 745.63in^2, and aspect ratio dropped (4.7 to 4.0). These changes directly affect aerodynamic load estimation, but it will not be a critical.

1.2.2) Assumptions

- Operating at an altitude of 1300 Feet
- The data on Table 1.1 is near enough to our Reynolds Number to be used
- Load factor of 4

1.2.3) Aerodynamic Characteristics

Detailed aerodynamic characteristics are shown on Table1.1. Aspect ratio is lower than what we desired, but this value is based on actual wing span and wing surface area. So, we expect reference aspect ratio to be higher than this; we have complex wing shape.

1.2.4) Load Estimation

To estimate aerodynamic loads, lift, drag, and pitching moment, on the wing, hand calculation from the airfoil dataand advanced aerodynamic tools, LinAir[11] and AVL[9], are used. Since each method is not entirely accurate, using both will provide a better basis for comparison to our hand calculations.

Lift

Kuchemann's equation for a swept wing[7], which is based on Helmbold's equation for low-aspect-ratio straight wing[7], is used for the hand calculations. A panel method code is utilized for AVL[9] and LinAir[11]. As Figure,1.6 shows, the $C_L$curves of these methods are very similar, the estimated $C_{L\ max}$, $C_{L\ cruise}$, and $C_{L\alpha}$is 1.0, 0.24, and 0.058/deg, respectively.

Drag

Finite wing theory[7] was utilized for the hand calculations. As Figure,1.7 shows, the estimated coefficients of the hand calculations and advanced aerodynamic tools are different, but the data still provides a range of $C_{D\ max}$ and $C_{D\ cruise}$. According to Figure,1.7, $C_{D\ max}$ will be a value between 0.01 to 0.02, and $C_{D\ cruise}$ will be approximately .12.

1.2.5) Drag Build

As the airplane is a modified flying wing to a flying wing with tail, the drag buildup is greatly simplified with the components being the wing body, vertical, and horizontal stabilizers, and landing gear. The airplane has two booms to connect main wing and tail, but these are located aft of the propellers and too thin to take into consideration. Initially, hand calculations and LinAirwere utilized, similarto the load estimation. However, the values were deemed too low to be realistic, so ROSKAM's theory is applied additionally (Table 1.2). ROSKAM's theory shows the highest value, so the estimated drag coefficient will be 0.055 instead of 0.028, initial estimated value,and should not exceed this value.

1.2.6) Takeoff and Landing Performance

Takeoff Distance: Key parameters of takeoff distance are $C_{L\ MAX}$, thrust, wing surface area, aircraft weight, and Lift and Drag at 0.7 times of takeoff velocity which is 1.2 times of stall velocity[B]. At this point, we know the $C_{L\ MAX}$, wing surface area, aircraft weight, and Lift and Drag (Table 1.1), so thrust is the parameter that can change the takeoff distance (Figure, 1.8).

Landing Distance: Key parameters of landing distance are $C_{L\ MAX}$, wing surface area, aircraft weight, and Lift and Drag at 0.7 times of landing velocity which is 1.3 times of stall velocity[2]. We know all the parameters at this point, thrust is not a key parameter for landing. So, we can estimate the landing distance (Table 1.1).

1.2.7) Turning Performance

Turn Radius: Key parameters of turn radius are $C_{L\ MAX}$, aircraft weight, wing surface area, and load factor[7]. As the aircraft weight is the parameter, turn radius will be changed before and after the mission (Table 1.1).

1.3) Ground Validation

1.3.1) Total Drag Coefficient at Cruise Condition

As is visible in Table 1.2, the drag values were found to have been higher by 0.017 according to ROSKAM's theory[10] as compared to the wind tunnel data. However, the data is in the range of estimation, so this value is reasonable. Without motors and props, the drag may differ slightly with the full scale model, but these should be negligible and counteracted by the lack of a wind tunnel mount on the full scale article. Moreover, there is no cross wind in the wind tunnel, and these effects are difficult to analyze.

1.3.2) Maximum Lift Coefficient

On the ground validation results, the C_L value is 0.3 lower compared to the design prediction as shown if Figure 1.9. At first, this would appear to be a major obstacle, but further analysis proves otherwise. The possible reason for this problem is based around the Reynolds Number. The major problem is that we were unable to test at the required q's for our flight conditions. One reason for this is due to the tunnel limitations for lift and drag, and also possible q. At high angles of attack, our model would exceed these limits even within the dynamic pressure possibilities of the wind tunnel. The other issue is due to model limitations. Due to an error in the given loads and test conditions, the model was incapable of operating at the high angles of attack at high q's. Structurally, for the q's that were originally provided, the model performed well but when pushed well outside it's designed use, the angles of attack were limited to avoid possible damage to the model and consequently, the tunnel. However, due to decreases in the weight of the full scale model over the original design conditions, the lift is more than adequate.

1.3.3) Cruise and Stall Speed

As the C_L values are lower than predicted, the stall, takeoff, and cruise velocities will increase compared to previous estimates, cruise speed increases7.5 fps (74.8 fps to 82.3 fps) and stall speed increases 7 fps(36.6 fps to 43.6 fps). These are all well under our maximum velocity and while an inconvenience, is not a fatal development.

Section 2) Stability and Controls

2.1) Conceptual design

2.1.1) Requirements and Constraints:

In the conceptual design process, the main focus of S&C is to come up with an aircraft design that would satisfy both longitudinal and lateral stability. Among the requirements of S&C was to maintain the reasonable static margin that is between 5% to 15%[15] .This also involve a minimal change in static margin when the payload is dropped. To achieve this, we had to come up with a design that would allow internal payload storage that is as close to the aircraft CG as possible. Other requirements and constraints included limiting the control surface deflection to 20[15,18] degrees to reduce drag and undesirable hinge moments. This can be achieved by proper sizing of the control surfaces. Another critical S&C issue during conceptual design was to come up with a design that would maintain stability in strong wind

2.1.2) Goals

Maintain Cm_α of less than 0[17]

Minimize wing disturbances on the horizontal tail

2.1.3) Center of Gravity placement:

In order to ease the allocation of the center of gravity to assist in balancing the aircraft, the battery pack has been assumed to be movable along the fuselage length. The weight estimates of different components will be balanced laterally and longitudinally to place the cg in a required location.

2.1.4) Control surfaces:

Analysis and optimization was conducted in selecting the appropriate airfoil for the tail vertical and horizontal. Symmetric airfoils[21] were chosen for horizontal and vertical tails as shown below.
Horizontal tail – NACA 0009
Vertical tail – NACA 0006

2.2) Preliminary design

In the Preliminary design process initial aircraft sizing and predicted aircraft weight was initiated, which is used to determine the lateral and longitudinal control performance of the aircraft. Trade studies are performed and optimization is done in order to come up with the best control surface sizing as possible. To ensure that the aircraft would be able to successfully complete the mission, both static and dynamic stability were analyzed. Control surface sizes, locations, and S&C derivatives are computed and optimized.

2.2.1) Critical S&C parameters

- **Payload Configuration:** Payload must be located such that minimal CG shifts occur after the payload is does not adversely affect stability characteristics[18]

- **Static Margin:** To obtain sufficient maneuverability, the static margin (SM) must be kept at a low value while still maintaining adequate stability[18]

- **Twin Propulsion System:** Motors positioned on booms will create a thrust-induced yawing moment. Large thrust-pitch coupling requires aircraft CG to be near the thrust-line[20].

- **Control Surface Sizing and Placement:** The control surfaces have been sized to maintain lateral-directional control in high cross-wind. The technique of maximizing the distance from the CG to the longitudinal control surface AC to reduce control surface deflections was used. All control surfaces are sized to keep deflections under 20°[15,18].

2.2.2) Components weights and CG placement

All the weights of major components have been estimated using historical data. Table 2.1 Shows the summary of the weights and their respective distances from the leading edge. Aircraft components have been positioned in such a way that they put a resultant aircraft CG at a reasonable position. With the weight estimates, the aircraft CG is place at 6.52 inches from the leading edge of the AC. This gives a static margin of 10%. The propulsion battery pack will be made to be movable to adjust the center of gravity so that the static margin can be flexible within a safety range of 5% to 20%. This will give enough tolerance in case the payload bay will be slightly shifted. The payload bay CG is made to be placed right by the CG of the aircraft.

2.2.3) Wing Sizing:

For a blended wing configuration, the wing occupies more space than any other component. The control surfaces of the wing have been sized using the Etkin[16] and Raymer[21] books. The aileron span has been assumed to extend to 50% of the wing span and 30% in chord.
.Aileron span is 13.8 inches
.Aileron chord is 1.8 inches

2.2.4) Tail and boom sizing sizing:

An H-Tail[21] configuration with a high horizontal is the one that has been chosen for our design to keep it out of the wake at high angles of attack. The tail booms have been assumed to extend from the wing section. They are the same booms that are being used to support the props. The length of the boom is calculated using the method in Raymer[22]. The moment arm (distance from the tail quarter chord to the wing quarter chord) which is part of the boom is calculated to be 39 inches which gave enough length to give a reasonable moment

2.2.5) Control surface sizing:

In order for the plane to have the required control in take-off and turn conditions, reasonable size of control surfaces is required. The control surfaces have been sized using Etkin and Raymer books .In order for a rudder to trim the aircraft for a sideslip at a particular velocity whether during cruise or stall, it has need sized to be 90% of the vertical span and 50% of the vertical chord. Also in order to trim the aircraft, the elevators[23] have been sized to be 90% of the horizontal span and 50% of the horizontal chord

2.2.6) Horizontal tail

A high horizontal has been the preferable one so as to keep it away from the wing disturbances as possible .The horizontal tail span is the same as the distance between the two booms.

- Airfoil is NACA 0009
- Horizontal tail chord is 5.7 inches
- Horizontal tail span is 21.6
- Elevator chord is 2.75 inches
- Elevator span is 19.44 inches

2.2.7) Vertical tail

The distance between the verticals is the same distance between the two props because they are also sized by the booms.

- Airfoil is NACA 0006
- Vertical tail chord is 5.7 inches
- Vertical tail span is 7.1 inches
- Rudder chord is 2.25 inches
- Rudder span is 6.39 inches

2.2.8) S&C Preliminary predictions:

The methods outlined in Youtch and Morriss[20] together with detailed techniques in Roskam[15] were initially used to estimate the S&C derivatives and they are summarized in table 2.1. Later on the stability characteristic was evaluated using the vortex lattice method as implemented in AVL. The resulting derivatives obtained proved the plane to be stable in both static and dynamic conditions.
Table.2 shows the summary of the control derivatives
The trim plots are reflected in fig.2.1

2.3) Detail Design

2.3.1) Aircraft Component Weight & CG Buildup

The weights associated with components used for the aircraft are maintained and updated regularly.Slight changes in the component weight made a slight difference in the aircraft CG but different components were moved around in order to maintain the reasonable SM . The new aircraft component CG buildup is presented in Table 2.3.
The updated CG of the airplane is at 7.7 inches from the leading edge with a SM of 8.3%

2.3.2) Wind Tunnel test and result analysis

Wind tunnel tests were run at different aerodynamic pressures and the data was analyzed and compared to the predicted values. There were slight offsets in the parameters due to different possible

reasons. The change in lift coefficient slope increased the $C_{M\text{-}alpha}$ to -0.004706.This is quite a huge swing but it still holds the stability of the airplane even at high angles of attack.
Pitching moment at zero angle of attack was found to be larger than the initially predicted value. One of the possible reasons here might be the inaccuracy in setting up the elevator deflection during model installation into the tunnel. Further different tests were run at the predicted elevator deflections during cruise and 1.2Vstall and the pitching moment trend proved the stability of the airplane since it was negative all the way even though the pitching moment coefficients were not as exact as the predicted numbers.

The plots for the moment coefficients at different aerodynamic pressures for cruise and take off conditions are shown in fig.2.3

2.3.3) Cruise and take off Condition Analysis.

The model was tested at different dynamic pressures of 6,16 and 24 to simulate cruise conditions. Elevator deflections were introduced to check trim conditions. The stall speed for the model was calculated to be 40.23 ft/s compared to 35.35 ft/s as predicted. The change in the value is because the model was tested at wide range of speeds with big intervals in between. Lesser speed intervals would have produced more accurate results. At higher speeds the model was able to generate enough lift with less elevator deflections. The pitching moment plot showed that the aircraft had a negative pitching moment (with a negative slope) the entire time indicating a stable flight.

Section 3) Propulsion

3.1) Conceptual Design

The first step during conceptual design was to study all the possible propulsion configurations that would fit the designs selected and analyze them with a mission-oriented focus. In order to do these studies assumptions were needed, but as the project and the analysis continue, the assumptions are going to be changed for real values.

3.1.1) Assumptions:

The following assumptions let us had a better and closer idea of the parameters that were critical during this step of the design process.

- Batteries have an average voltage of 1V per cell.
- Batteries being considered: Elite 2,000 (2,000 mAh).
- Airplane weight, W = 8lb. (128 oz.).
- Oswald efficiency factor, e = 0.75 (most aircraft are between 0.7 and 0.85).
- Zero-lift drag, CD_0 = 0.02045.
- Propeller efficiency, η = 0.4.
- Lowest Wichita altitude = 1,296 ft.
- Highest altitude allowed = 1,670 ft.

3.1.2) Battery Analysis:

The initial analysis made to the batteries was to obtain the maximum number of cells allowed since we have a weight restriction of 1lb. (16 oz.) for the entire propulsion system battery packs.

- At 20 cells weight is 18.2 oz., which is more than the weight allowed.
- At 17 cells weight is 15.47 oz., which is below the weight allowed but since we are running 2 motors we need an even number of cells.
- Maximum number of cells would be 16, which is a total weight of 14.56 oz. or 0.91 lb.

3.1.3) Single-Engine vs. Dual-Engine:

Analyzing the mission and the assumed weight of the fully loaded aircraft(including payload), a single-engine would be produce enough thrust; but after reviewing and doing analysis on the weight, efficiency, and performance of the single motor, a dual-engine propulsion system had more configuration options available.

- Weight difference between 1 and 2 engines was minimal in order to be considered an important factor. A single-engine configuration would need to be bigger with a heavier propeller; a dual-engine configuration would need to be smaller with lighter propellers.
- Cost was considered since it is one of the scoring factors. The difference in cost between a single and a dual-engine configuration was about 25% to 50% giving an advantage to the single-engine configuration.
- Considering performance, there is a big advantage using 2 engines. Using the best single engine at a max of 40amps gives us the performance of a basic/below average dual-engine configuration.
- Also using a dual-engine configuration providesmore market options when trying to optimize our airplane on the next design steps.

3.1.4) Calculations:

The following values were obtained after doing the proper analysis using the assumptions mentioned at the beginning of this section:

Stall speed:

- 1,670 ft.: $V_{Stall\ Altitude}$ = 64.6ft/s
- 1,483 ft.:$V_{Stall\ Altitude}$ = 64.4ft/s
- 1,296 ft.: $V_{Stall\ Altitude}$ = 64.2ft/s

Take off speed:

- V_{LO} = 39.6 ft/s

Landing speed:

- V_T = 42.9ft/s

Minimum power required at cruise condition:

- 1,670 ft.:$V_{Min\ PR}$ = 44.8 ft/s
- 1,483 ft.:$V_{Min\ PR}$ = 44.7 ft/s
- 1,296 ft.:$V_{Min\ PR}$ = 44.5 ft/s

3.1.5) Predictions:

Also the following parameters were predicted:

Power available:

- PA = 240W.

Propeller Diameter:

- D = 8.5 in.

Endurance:

- 240 seconds.

Propulsion system weight:

- $W_{P\,=}$ 1.81 lb.

3.2) Preliminary Design

Continuing with the following step of the design process, preliminary design,validation and verification of the data previously obtained needs to be done. Some coefficients were updated, changing assumption for real parameters. After being done with these initial steps, the results were improved with better equations and more deep analysis, as needed. Takeoff and landing analysis was rechecked since assumptions previously made were vague due to minimal information on the landing gear configuration.

With all previous updates, the diameter of the propeller was obtained and, in order to have better range for analysis and market options, propellers that were 1 in. smaller and 2 in. larger were taken into consideration. Our range of search was between 7 in. and 10 in. propellers, and the brands taken into consideration were APC and AeroNaut.After doing the proper analysis using JavaProp the best three propellers arranged from high to low efficiency were: APC 10x5E, APC 9x4.5E, and APC 8x4E.

The idea of having a counter-rotating propeller configuration was taken into consideration. If a propeller with a counter-rotating configuration was selected within the best three and the efficiency difference was not that high,the selection would change giving preference to the counter-rotating propeller. Torque effects produced by the engines would be balanced and no engine would be considered a critical engine, in case of a failure.

Being said this and after analyzing, the new propellers arrangement giving preference to the counter-rotating configuration and arranging from high to low efficiency was: APC 9x4.5E, APC 10x5E, and APC 8x4E. The counter-rotating propeller or pusher propeller is APC 9x4.5EP and the efficiency sacrificed is about 4%.

Once the propeller selection was done and since all power requirements were already analyzed, the motor selection started. Motors that have a KV value between 700 and 1,200 were considered in order to run a motor at low revolutions per minute (RPM) and high voltage. Having a low KV motor will help to produce more torque.

Different market options were considered, but in the end the motors that fitted the requirements and needs were the Rimfire .10 and the SuperTigre .10. Rimfire .10 motor has the following characteristics: 325W, 30A, weight of 0.156lb., and a KV rating of 1,250. SuperTigre .10 has the following characteristics: 320W, 29A, weight of 0.150lb., and a KV rating of 1,250. With these values, Rimfire was the primary option and SuperTigre the backup option. It is important to remember that Udaan has a twin-engine configuration.

Initially, the battery configuration was two packs of 7-cells Elite 2,000, but after concerns about its poor and variant efficiency the batteries were changed. The final configuration is two packs of 8-cells Elite 1,500.

3.2.1) Assumptions:
These are the assumptions made during this process.

- Takeoff altitude = 1,300 ft.
- Cruise altitude = 1,439 ft. (considering the highest altitude in Wichita plus 100 ft.).
- Friction coefficient, μ_r = 0.4 (assuming paved surface)
- Ground effect, ϕ = 0.81

3.2.2) Results and Data Analysis:
These are the final results and data obtained after the proper analysis. A thrust available versus thrust required plot for cruise condition can be found on Propulsion Plot 1.

- Power from the motor, Pa = 315W / motor.
- Endurance = 270 seconds.
- Average propeller efficiency, η = 0.5.
- Minimum power required speed, $V_{Min\ PR}$ = 31.2ft/s.
- Cruise speed, V_{Cruise} = 73.5 ft/s.
- Maximum speed, V_{Max} = 87.3 ft/s
- Total propulsion system weigh, W_P = 1.65 lb.

3.2.3) Propeller Analysis:
Counter-rotating configuration with one APC 9x4.5E and another APC 9x4.5EP propellers. Plots obtained from the analysis made at different RPM's on JavaProp can be found on Propulsion Plot 2, Propulsion Plot 3, Propulsion Plot 4, and Propulsion Plot 5.

3.2.4) Motor Analysis:
Two Rimfire .10 motors with the SuperTigre .10 motors as backup.

3.2.5) Battery Pack Analysis:
Two packs of 8-cells Elite 1,500 with an average endurance of 270 seconds.

3.3) Detail Design

As the design process continued, ground validation took place. During this step, the propulsion system was tested and reviewed in order to proof calculations and estimates previously obtained. The final propulsion system consists of the following items: propellers, motors, motor mounts, ESC's, wires, fuse and batteries.

3.3.1) Propulsion System Weight:

The initial step during this part of the design process was to have the propellers balanced and break in the battery packs. After finishing these, a more accurate and realistic weight was obtained. Previously our weight was estimated to be at 1.6 lb. (without motor mounts); now our actual weight is 1.76lb. (Including motor mounts). This tells us that our weight estimate was correctly made and that the time spent obtaining information was adequate. A detailed table that includes individual and total weight can be found on Propulsion Table 1.

3.3.2) Battery Packs:

After testing our battery packs, we were pleased with the data and performance obtained. Our batteries are capable of holding more than 1Vgiving us the performance desired. As an example drowning 20A, which will be the maximum amperes drowned by each pack, we got an endurance of 195 seconds. This allows us to be more confident about our configuration and the analysis done on the batteries. A plot comparing battery tests at different amperes and endurance can be found on Propulsion Plot 6.

3.3.3) Thrust:

The next essential step, and maybe one of the most critical, was to validate the thrust obtained from the motors and propellers. Previous calculations indicated static thrust of 3.28 lb. (total, for two motors) and a cruise thrust of 3.22 lb. (total, for two motors).

The settings for our propulsion test were to have the motors running at 17V and drawing 20A at different Q's (dynamic pressure) for cruise, maximum and landing conditions. A detailed table with the results obtained from the wind tunnel test can be found on Propulsion Table 2.

After reviewing the data obtained from the wind tunnel test, some errors were found. During the propulsion test one of the propellers was not properly adjusted and came loose while testing cruise thrust, so data for that run wan not recorded. This means that data for static thrust was not obtained properly and the value presented on the table above is not the real one since it only shows a fraction of it. Also some of the wiring was not label correctly or compatible with the equipment on the wind tunnel at the beginning of the test, but they were solved before the test runs while mounting the propulsion system.

Data for cruise, maximum, and landing are reliable and show that even though the thrust values are lower than predicted (i.e. 0.6 lb. less on cruise thrust), they provide enough thrust to perform with great success the mission required. The difference between the calculated and the real thrust obtained show that all the previous analysis was done correctly, since many factors such as real propeller efficiency and motor performance is not going to be known until testing is complete.

3.3.4) ESC (Electronic Speed Controller):

The choices for motors, propellers and batteries were carefully selected considering losses on each category. The configuration tested on the wind tunnel proved that it was ready to be mounted, but once it was installed and ready for be programmed the ESC's did not work due to an over-voltage.

The ESC's provided were not compatible with the battery configuration in series since they are providing around 22V. New ESC's that are capable of holding more Volts were used for test flights and competition.

Section 4) Structures

4.1) Conceptual Design Report

4.1.1) Requirements and Constraints:

a) To provide a structural framework that would withhold the stresses and forces for all the conditions during the mission.
b) The wingspan has to be a minimum of 3 feet.
c) The payload has to be placed inside the flying wing hence the structure should have enough room (thickness).
d) The aircraft should be made as light as possible without neglecting the structural abilities and integrity.

4.1.2) Assumptions:

a) The beams used in the structure are assumed to be prismatic, homogeneous and uniform.
b) Other properties like thermal, impact etc. are ignored at this phase of design.
c) The materials used are assumed to be Hookean in nature.
d) Half the wing accounts for 50% of the total lift.
e) The lift generated is uniform across the wingspan and the lift profile is rectangular.

4.1.3) Material Selection and Properties: A careful analysis of different materials was done for their physical and structural properties. Balsa and spruce were chosen because of their high Modulus of Elasticity and high strength to weight ratio. These two materials are easily available and cheap too.

4.1.4) Factors of Safety and Load Factors[4.1]:

a) Limit Load: The maximum load the aircraft is going to experience during the flight. The load is typically the highest amount of lift generated.

b) Proof Load: The product of Limit load and proof factor gives us the Proof load which in reality is the amount of load the structure should hold without any damage. The proof factor lies somewhere between 1.0-1.5.

c) Ultimate Load: The product of Limit Load and the ultimate factor gives us the ultimate load, which is the load beyond which the structure fails. The ultimate factor is usually 1.5..

4.1.5) Beam analysis[4.2]:

Euler-Bernoulli's beam theory was used to analyze the different cross sections of beams. Two major spars will be utilized at approximately 25% and 70% chord. The center of pressure is more likely to be along the 25% chord line and hence the main spar (at 25%) would account for approximately 75% of the wing loading. The aft-spar at 70% chord would carry the remaining 25% load and also support control surfaces like ailerons. The lift distribution was assumed to be rectangular and uniform across the wing-span, which is actually the worst case scenario.

a) I –Beam: I-beams for different materials were analyzed to carry the proof load without significant bending; I-beams weigh lesser and provide significant moment of Inertia. History tells us that I-beams are one of the best choices when it comes to spars for aircraft wings. The maximum deflection at the wing tip while experiencing the proof load was found to be about **1.3** inches.

b) Solid Rectangular Beam: Solid rectangular beams for different materials were analyzed for the spars. These beams provide good structure and rigidity but weigh heavier than the I-Beams.

c) Circular beams for the booms: Circular beams will be used to build the twin-booms. The booms will have to restrain bending as well as torsional loads.

4.1.6) Stresses on the Beams:

a) **Shear Stress**[4.3]: Shear stresses were calculated along the beam using the Mechanics of Material's approach. The maximum shear stress was observed at the midsection of the wing. Figure 4.2 shows the distribution of shear stress along the wing span. The maximum shear stress was calculated to be **95 Psi**. (Fig 4.3)
b) **Internal Moments**: Internal moments were calculated along the beam using the Mechanics of Material's approach.
c) **Normal Stress (Bending)**[4.4]: Normal stress or bending stress was calculated along the wing-span, again the maximum normal stress was found to be at the center of the wing-body.

The maximum Bending stress calculated was about **3.6 Ksi** (Fig. 4.3)

4.1.7) Twin Booms: There are 2 booms that connect to the power plant and all the way across to the tail section. The vertical stabilizers are attached to the booms. The booms will provide structural stability and support for the airframe. They will also act as the moment carriers for stability and control. The twin booms will be made off thin walled tubular beams for structural rigidity and strength.

4.1.8) Landing Gears: For the purpose of ground control the best type of landing gear would be a tricycle for the flying wing. The gears will be picked off the shelf, readily available in the market and the landing gear assembly will be designed such that it would be capable of handling 5g loads (Proof Load) which would ensure that no damage results from a rough landing.

4.1.9) Tails: Tails (Vertical and Horizontal) will use spars made up of spruce and have a I cross-section. Further analysis on the tails will be done at a later phase of design.

4.2) Preliminary Design Report

4.2.1) Material Finalized:

a) Spars – Spruce sticks
b) Ribs – 1/16th and 1/8th Balsa
c) Aluminum 6061T for the Tail-Booms.
d) Bulkheads: Balsa and spruce sticks.

4.2.2) Spars: - Two spars (main and aft) will be used to make the wing structurally sound. Because of the uneven geometry, placing the main spar at 25% of the MAC would involve using joints, which would mean weaker points and would also complicate the structural analysis. The main spar would be at 25% at the wing tips and the center section. The center of the aircraft would have stronger ribs to transfer and account for the moment generated by the forces at the AC. The spars are designed to withhold bending stresses up to 3.8Ksi[4.1] and shear stress of 164 Psi, 107 Psi at the main and aft spars respectively. The detailed dimensions and stresses on the spars can be found on figure 4.3 and 4.4 respectively[4.3].

4.2.3) Ribs: Ribs are responsible to give the preferred aerodynamic shape to the wing-body. After careful analysis and shear force calculations balsa wood sheets was found to be the best choice for the ribs. Two different thicknesses of ribs will be used along the span. The payload section (15'x24') will use 1/8th balsa whereas the rest of the wing will use 1/16th Balsa. Shear flows and shear stresses were calculated along the surface of the ribs to make sure that they can handle the different forces generated during the flight.

4.2.4) Twin Booms: Twin booms on the aircraft prove to be very vital as they connect most of the important parts of the airplane. The booms run from the very tip of the aircraft till the very end. The motors will be mounted on the booms and the booms would carry through the wings and into the tails as shown in the figure 5.1. Finding a good material for the boom was crucial as it is almost 4 feet long and we have 2 of them. After careful analysis thin-walled cylindrical aluminum tubes were found to be strong and light enough to hold all the bending and twisting. The booms selected were of 3/8th inch diameter with 0.035' thickness. The 2 booms combined weigh about 0.3 Lbs...

4.2.5) Payload Section: The tennis balls will be placed between the ribs somewhere along the center-line of the aircraft. Span-loading is incorporated to reduce weight and reduce structural complexity. The bombing mechanism is discussed at a later part of the report. Thin sheets of ply would be used as the bomb-bay door. They will be hinged to the ribs and spars. A small scaled prototype of the bomb bay and the doors was analyzed to predict the thickness of the ply and the amount of stresses it encounters.

4.2.6) Tails (Horizontal and vertical): The actual sizing and location of the tails can be found in table 6.1. Two verticals will have 1 spar located at the aft to support the rudder and provide a better structure. 1/16th Balsa will be used for ribs and ½ x ½ (inches) spruce sticks will be used for spars. Similarly the horizontal will have a spar to support the elevator as well as provide the strength to account for the lift and moments on the tails.

4.2.7) Landing Gear: There are three angles/distances that a taildragger configuration must adhere to. These include: the C.G. location, static margin, tail down angle, maximum roll angle till wingtip to ground contact. These are not determined by the landing gear, but are a requirement for its correct placement.

The tail dragger configuration has the positive consequence, of allowing for steerable tail gear to aid in ground handling. By connecting the tail gear directly to the rudder, steerable gear can be achieved with very little increase in weight and does not require additional servos. This becomes a favorable tradeoff, making takeoff possible in more adverse conditions.

The first of these is the rotation necessary for takeoff. For the landing gear, this must be in a range between 10 and 15 degree [37]. Since our necessary rotation is only 8 degrees, a maximum rotation of 12.6 degrees allows for extra margin of error to avoid damage to the tail from over rotation. This value falls into the required range angle as well. See figure 4.5 for the side view

The next angle is from the landing gear location to the C.G. Then entire static margin, furthest forward C.G., and furthest aft C.G. to the landing gear wheels should lie between 16 and 25 degrees. This gives the final landing gear location to be 5.3" aft of the leading edge, and a height of 7" which will provide a total measurement to the C.G. in the vertical direction of 9".

The lateral location is also important to make sure the aircraft will not overturn the main gear [38]. The optimum landing gear location for structural considerations is with them being mounted to each of the twin booms. The angle from the gear to the wingtip in the tail down configuration should be great than 25 degrees. In this current location, this angle is 35 degrees. See Figure 4.6 for the front view. This easily meets the requirements and should also aid in landing stability by creating a larger platform to land on [27].

While a set of landing gear must be strong and fit all the requirements, it must also be as light as possible. For the main gear, a set of off the shelf aluminum main gears were chosen being that they were strong, the exact dimensions needed, and still fairly lightweight. Composite main gear was not chosen due to higher cost, the geometry not being as close, and finally strength. While composites are strong, finding a set with the required dimensions capable of handling our aircraft fully loaded were essentially nonexistent. If a set was close, it was made for a much heavier airplane and was consequently too heavy. The next components were wheels. With the low weight of the aircraft, foam wheels were chosen due to their very light weight. They weigh 0.095 Ounces [28] each for the main tires, and 0.016 Ounces [29] for the rear tires for a total tire weight of 0.101 Ounces. The front have a diameter of 1.6875" and the rear have a diameter of .75".

4.3) Detail Design:

4.3.1) Ribs & Spars: The ribs were made up of $1/8^{th}$ and $1/16^{th}$ balsa as analyzed during the preliminary design phase. The spars had to be modified slightly in order for them to be readily available in the market. Precautions were taken while choosing the sizes keeping in mind the load factors and stresses associated with them. The final sizing of the spars are provided on Figure 4.4. The overall weight of the spars increased approximately by 7% but that was counterbalanced by making lightening holes on the ribs and the landing gears.

4.3.2) Booms: The aluminum booms were sized exactly as analyzed during the preliminary design phase. No changes were made as they were readily available in the market. The booms were attached to the motors on one end and the tails at the other end. The bending on the boom was approximately 1.2E-3 inches[4.2]at the load factor of 3.8.

4.3.3) Landing Gears: Landing gear was purchased off the shelf in accordance with the analysis done during the preliminary design. Aluminum plates and tubes were used to make the landing gear mounting plates which were then welded to the booms.

4.4) Ground testing and validation:

4.4.1) Experimental Results: Structural load test was performed by loading the wing along the spar with sand-bags. 19 sand bags each weighing 1.02 lbs. were loaded uniformly along the main-spar and 9 sand-bags were loaded uniformly along the aft spar each weighing 0.52 lbs. The total load on the wing was 24 lbs. The wing deflected by 1.13 inches, approximately 0.07 inches less than what was predicted.

4.4.2) Discussions: As far as the slight deviation on the spars' deflection is concerned, it's mainly due to not so accurate loading. Accurate loading was not possible because of ribs in between the spars. In addition to that the spars used on the aircraft were slightly bigger than the one initially analyzed, that was to avoid construction issues (Joints) as these were off the shelf readily available spar sticks. The spars are still within the estimated bending region (75% of fracture state). As a conclusion the structure will be able to hold the desired load factors and different flight conditions during the mission.

Section 5) Payload Analysis

The payload that our aircraft will be carrying consists of 20 tennis balls held internally of the main wing. They will be arranged in a pattern that is 5 columns, each oriented to run the length of the chord, of 4 tennis balls. (See Figure 5.1). A large portion of our design hinged on the idea of using a payload bay. The issue was finding a mechanism that was both strong and still lightweight. The mechanism that we designed can be run by a total of 2 servos for all the tennis balls. It will allow the bomb bay doors to both open and close on command to minimize the drag. After building a small mockup for a proof of concept test (See Figures 5.2 and 5.3) several things became apparent. One is that for the design to be most effective, the servo and rod mechanism must be mounted centrally forward to aft along the doors. This minimizes the torque placed on the linkage mounts, and will minimize the necessary structure as much as possible. The bay doors will be oriented so that they hinge parallel to the slipstream. What this does is minimize the force necessary to open and close the doors while the aircraft is in flight.

There were two other configurations considered, but both were deemed unusable for our airplane design. One idea was to just open the doors, and leave them open. The next was to open them and then have a sort of spring close them again. Neither of these options were chosen because while lightweight, they did not securely close the doors which was an unnecessary risk to take with the payload being carried internal to the wing that is relied upon for lift.

When matched with the mission profile, that of a dive bomber, the ability to drop the entire payload quickly is crucial for accuracy with such a small target. With the payload bay designed to drop the payload all at once, this will complement the mission profile and provide the best accuracy.

Section 6) Lessons Learned

There were many lessons learned during this project that were both part of the project, and others dealing with the team. The ones learned dealing solely with the project mostly dealt with construction. The main lesson deals with the adhesive. The build time greatly increased due to wasted time waiting on the glue to cure. This problem was made worse due to the design of the airplane. Instead of having a wing, fuselage, and tail assemblies, the airplane has just the wing and tail assembly. This means much less to do during the down time which negatively affects build time and makes planning even more crucial. The other lesson deals with both team members and building experience. While there is no solution besides gaining experience, or already having it, after building the aircraft, having more build experience would not only speed things up, but would also make the process go much smoother. Also with experience, the issue of having experience with remote control systems becomes very apparent. Again there is no solution other than experience. A lot can be learned from other sources, but it is still no replacement for experience dealing with these systems. There were also lessons learned dealing solely with the team. The main one deals with finding a meeting time that will work for everyone. It's harder than it would seem for a group of seniors. It is actually not jobs that really make this difficult, but instead is other classes that are the main source of interference. This leads directly into the next lesson that was learned. When few meeting times were found, the possible options were the weekend. After the meetings were established during those times, production suffered greatly. The lesson is that the team members do need downtime to stay focused and efficient. While there were others, these proved to be the most prevalent.

Section 7) Final Project Costs

At the beginning of the project, true to industry examples, the team was required to propose a budget for the project. This was challenging with no one having any prior experience, but the proposal actually was fairly close to the actual costs. Initially, our estimate was a total cost of $440,812.5 overall. For a breakdown, see Figure 7.1, and Table 7.1. The reason for the similarities is simple, overestimation on some areas, not using certain facilities, and going over budget in other areas. All of this helped to even out the costs and led to our final project costs being $441,150 with operating costs. A breakdown of the actual values can be found in Figure 7.2, and Table 7.2.

Section 8) Vehicle Cost

The initial estimate on the vehicle cost was also fairly accurate. Until the ESC issues, the airplane cost was actually under budget at a cost of $478.39 but with the addition of the new ESC's, the price rose to $516.84 as it currently stands. The reason it was initially so accurate was that when the initial estimate was constructed, the actual prices of generic components were used, and then an additional buffer of 10% was added to cover shipping and any price differences. With some parts being less expensive, or on sale, that dropped the price further and helped absorb some of the price increase of the new ESC's.

Section 9) Conclusion

The aircraft has undergone many changes from conceptual design, to the completed model. The completed model ended up weighing less than our original design, and even less than our most recent weight estimate. The steps taken from conceptual design to detailed design and then construction pointed out the flaws in our original design, and gave us insight into what changes needed to be made. The final airplane is light weight, and very low drag which fits the mission profile of designing a high efficiency bomber perfectly. Especially when combined with the alterations required for accuracy that necessitated the need to change the profile to that of a dive bomber. By staying mission oriented throughout the project the aircraft design benefited greatly, which in turn led to the team receiving additional benefits from this experience. It was not the hardest design to analyze, but was a challenge nonetheless. This will greatly help the team as they go on to their careers, by enhancing what was already an important aspect of their education. All of this hard work culminated in an airplane that flies very well, and is very stable in flight.

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Charts and Figures

Mission Description

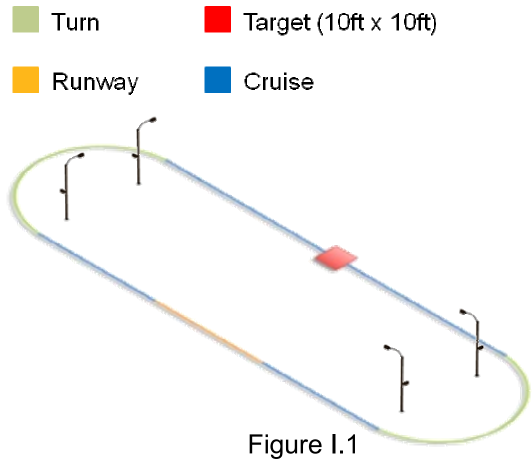


Figure I.1

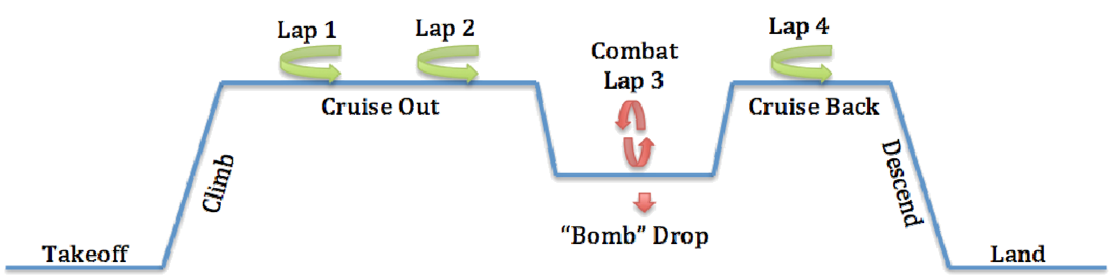


Figure I.2

Competitive analysis

Parameter	Nominal Values	Uncertainties	Effect On Total Score
NBOT	4	2	0.97
WEFR	4.5	0.25	0.012
VG	80	10	0.0083
CA	500	50	0.0052

Table II.1: Nominal values, uncertainties and sensitivities for scoring parameters

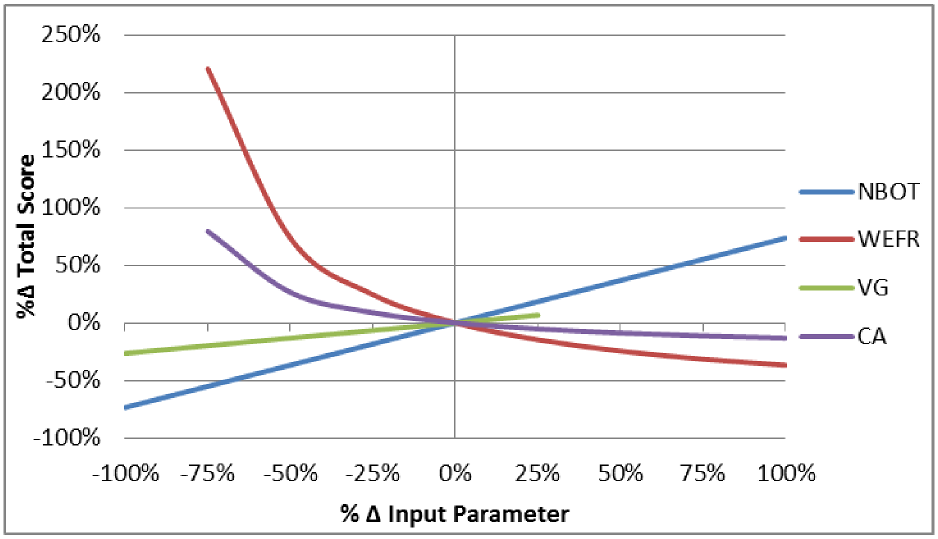


Figure II.1 :Competitive score sensitivity

NBOT			
%change NBOT	NBOT	SCORE	% change in SCORE
-100%	0	32%	-74%
-75%	1	54%	-55%
-50%	2	76%	-37%
-25%	3	99%	-18%
0%	4	121%	0%
25%	5	143%	18%
50%	6	165%	37%
75%	7	188%	55%
100%	8	210%	74%

Table II.2

WEFR			
%change WEFR	WEFR	SCORE	% change in SCORE
-75%	1.125	388%	221%
-50%	2.25	210%	74%
-25%	3.375	151%	25%
0%	4.5	121%	0%
25%	5.625	103%	-15%
50%	6.75	91%	-25%
75%	7.875	83%	-32%
100%	9	76%	-37%

Table II.3

VG			
%change VG	VG	SCORE	% change in SCORE
-100%	0	89%	-26%
-75%	20	97%	-20%
-50%	40	105%	-13%
-25%	60	113%	-7%
0%	80	121%	0%
25%	100	129%	7%

Table II.4

CA			
%change CA	CA	SCORE	% change in SCORE
-75%	125	217%	79%
-50%	250	153%	26%
-25%	375	132%	9%
0%	500	121%	0%
25%	625	114%	-5%
50%	750	110%	-9%
75%	875	107%	-11%
100%	1000	105%	-13%

Table II.5

Concepts and Configurations Considered

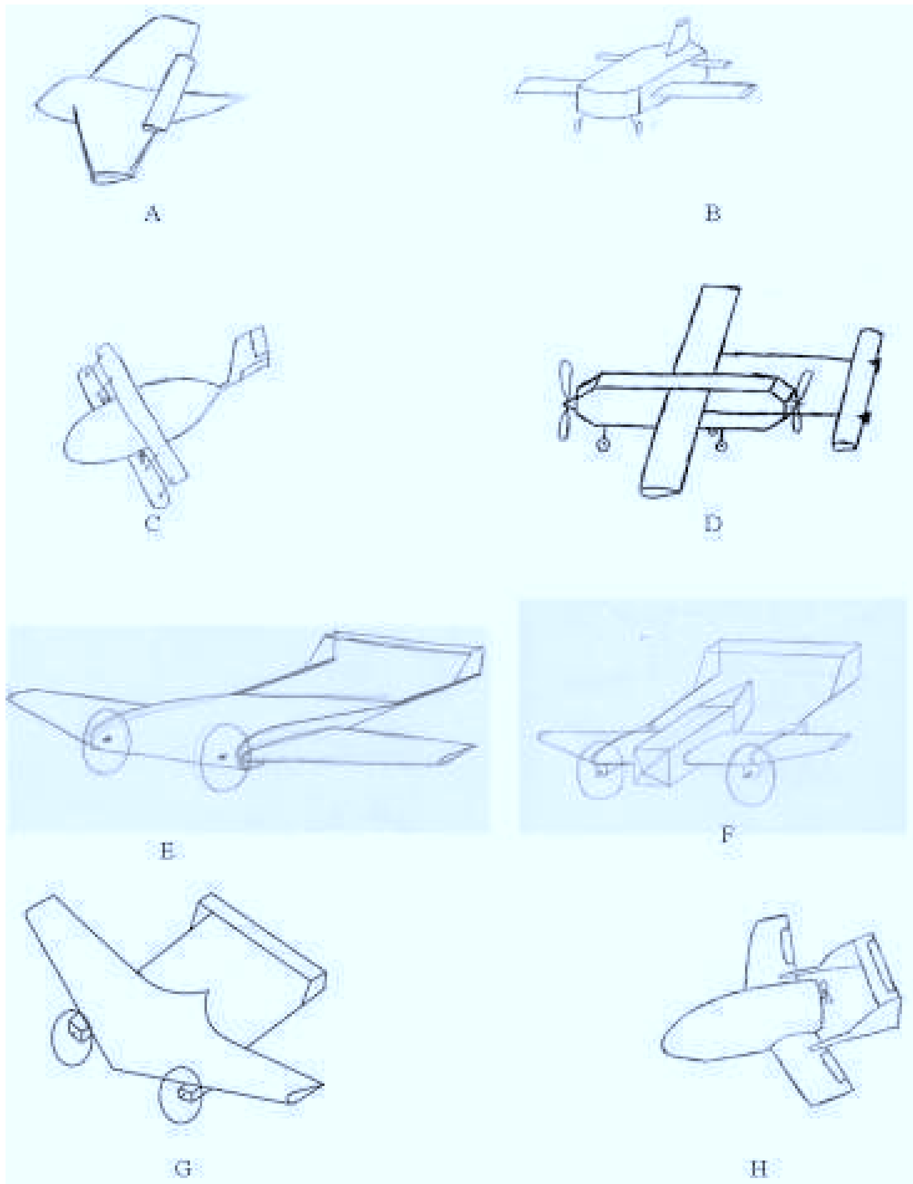


Figure III.1

Concepts and Configurations Considered

Screening Criteria	Concept I	Concept II	Concept III	Concept IV	Concept V	Concept VI	Concept VII	Concept VIII	Control
Ease Of Loading	-	-	0	0	-	0	+	+	0
Ease of Bombing	-	-	-	0	-	0	0	-	0
Repairability	0	0	0	0	0	0	0	0	0
Ease of Manufacturing	+	+	-	0	0	0	0	+	0
Accessibility of Systems	0	0	0	-	-	0	0	0	0
Landing Gear/Layout	0	0	0	0	0	0	0	-	0
Durability	+	+	+	-	+	0	+	0	0
Complexity	0	+	0	-	0	0	-	0	0
Is it Cool?	+	-	+	+	+	+	+	+	0
Total +'s	3	3	2	1	2	1	3	3	0
Total 0's	4	3	5	5	4	8	5	4	10
Total -'s	2	3	2	3	3	0	1	2	0
Overall	1	0	0	-2	-1	1	2	1	0

Figure IV.1

Concept Weighting and Selection Process

Criteria	Weight	Concept I	Concept VI	Concept VII	Concept VIII	Control
Payload Accessibility	30	1	3	4	3	3
Repairability	10	3	3	3	3	3
Ease of Manufacturing	15	4	3	3	4	3
Accessibility of Systems	10	3	3	3	3	3
Durability	20	4	3	4	3	3
Complexity	15	3	3	2	3	3
Score		275	300	335	315	300

Figure IV.2

Justification of Selected Concept

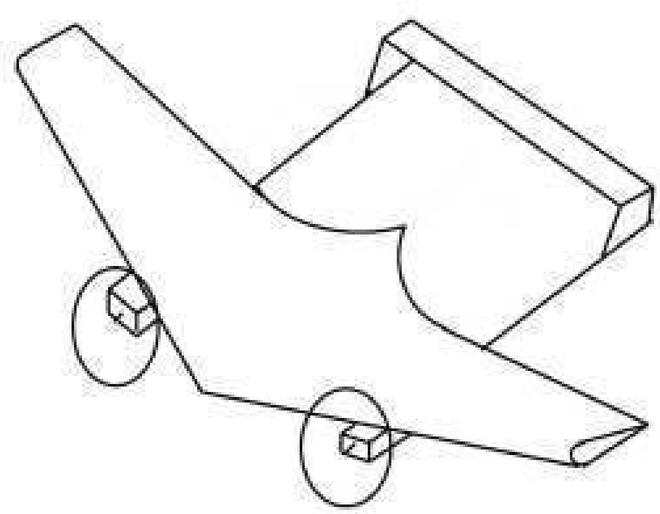
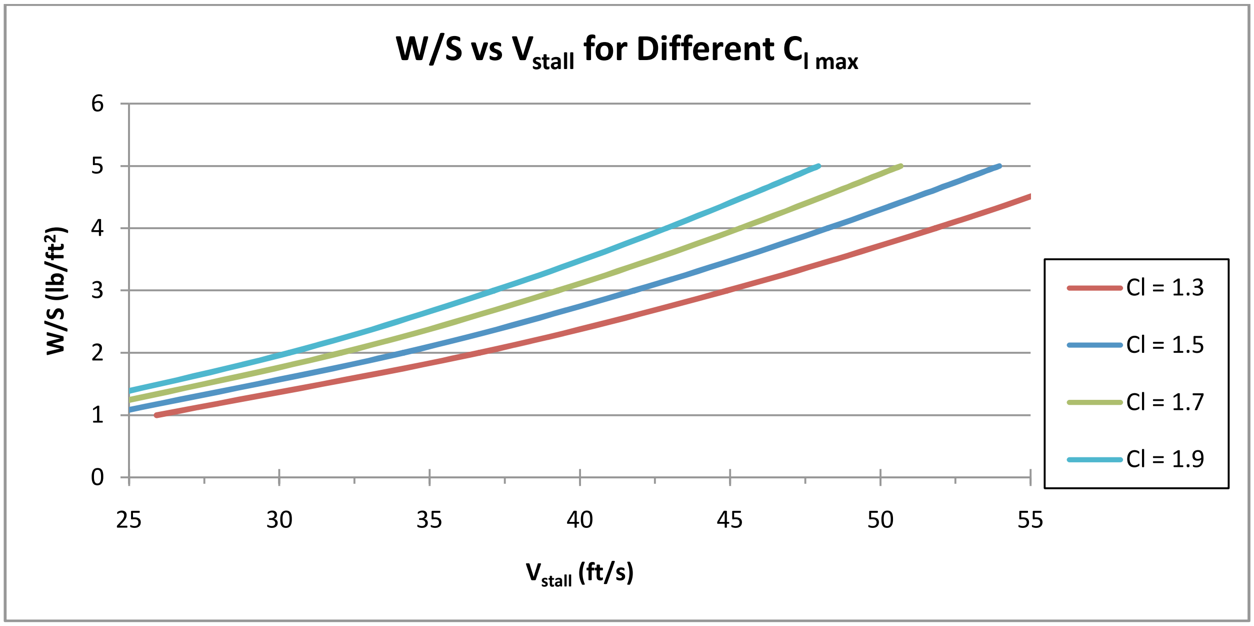
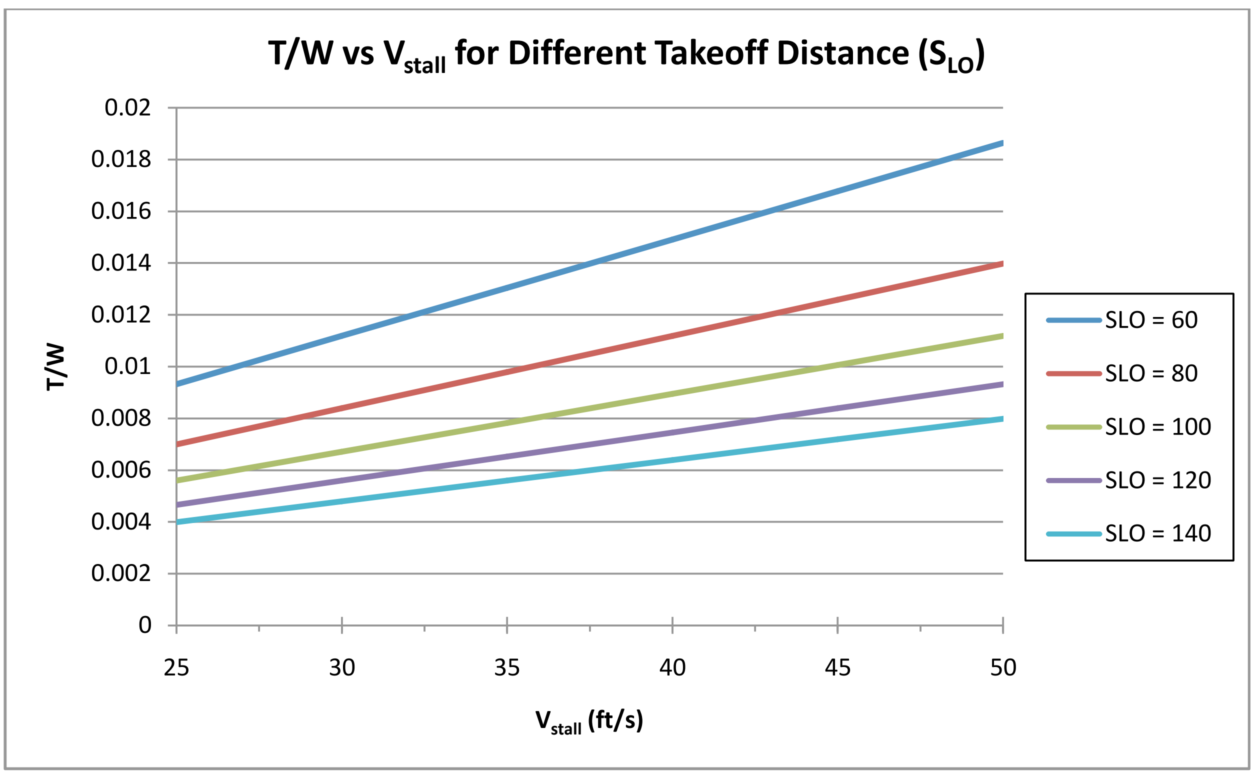


Figure V.1

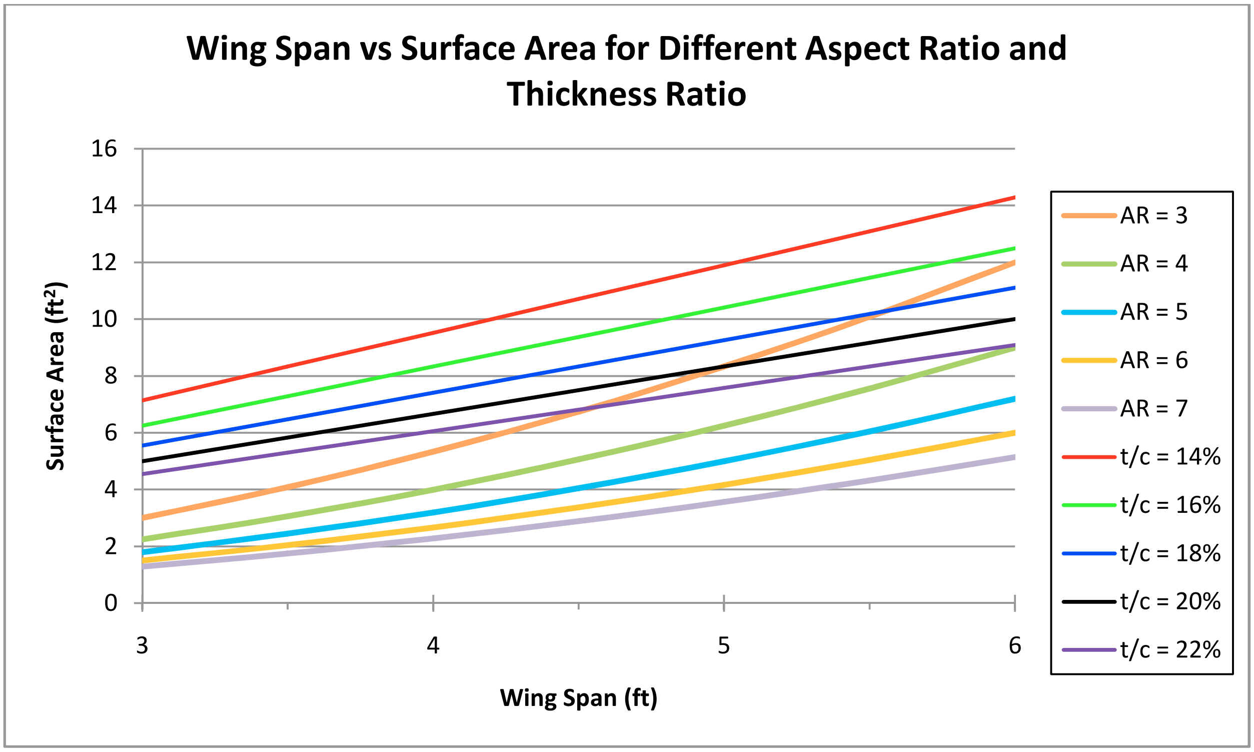
Aerodynamics



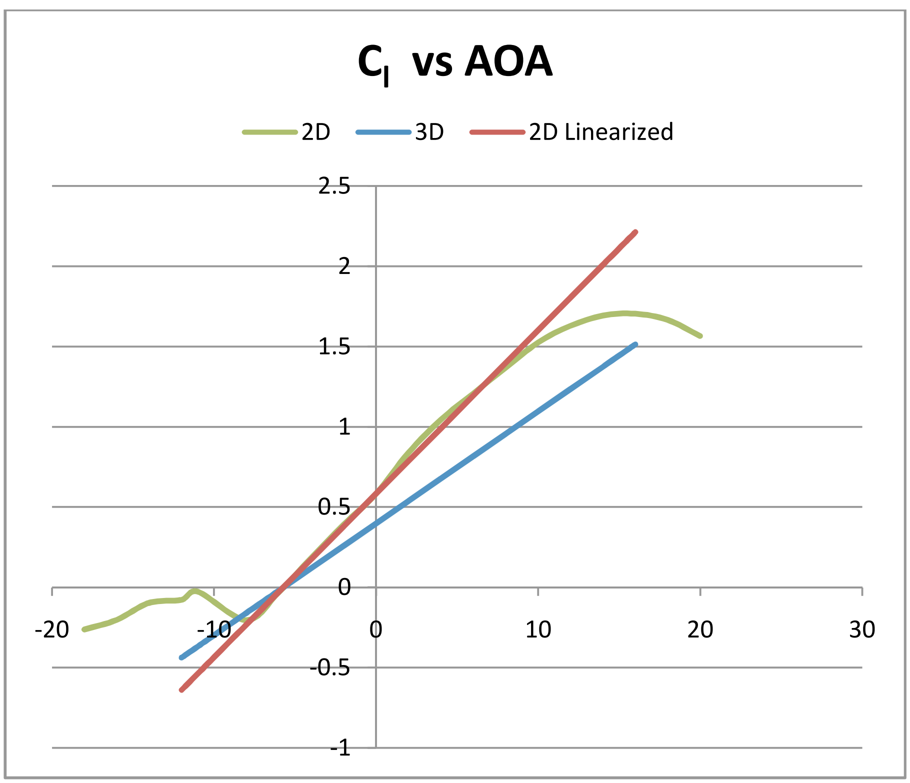
Figure, 1.1 Relationship between Wing Loading and Stall Speed at Different $C_{l\ max}$



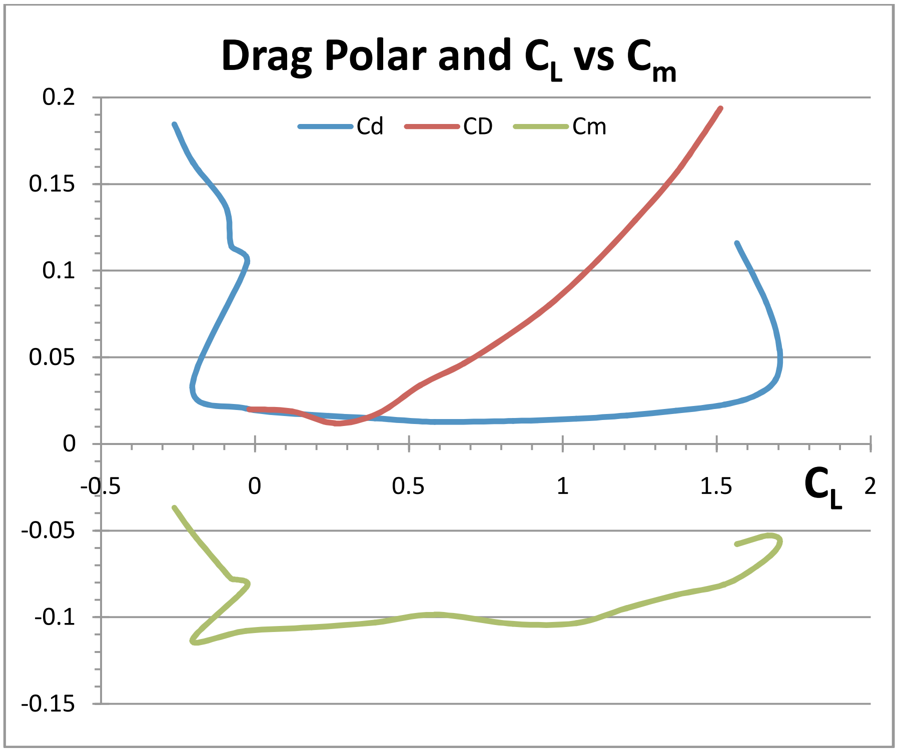
Figure, 1.2 Relationship of Thrust-to-Weight Ratio, Stall Speed, and Takeoff Distance



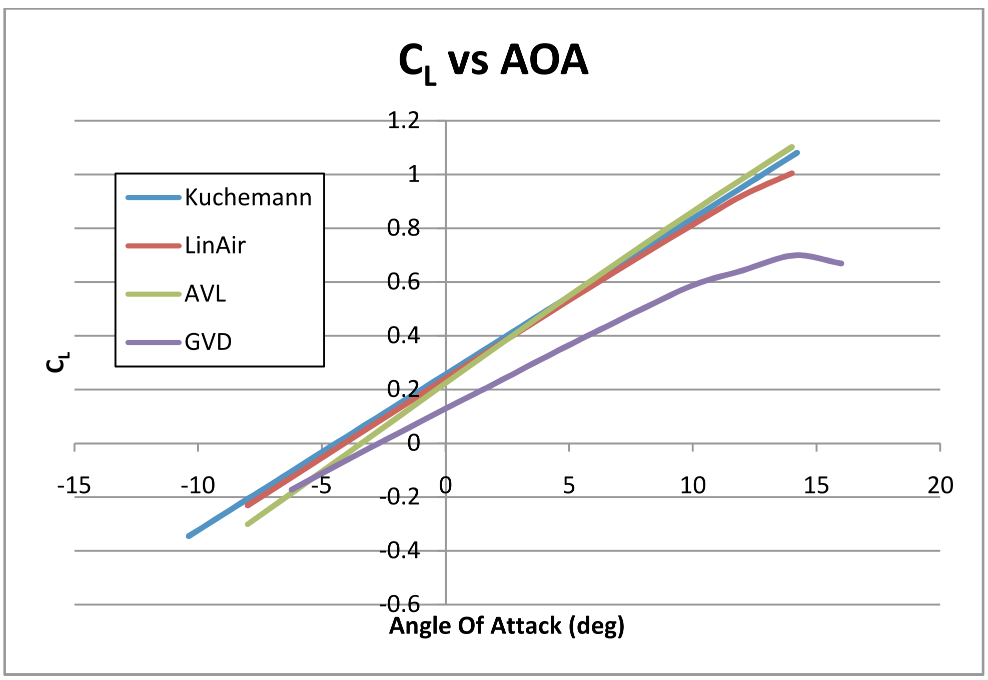
Figure, 1.3



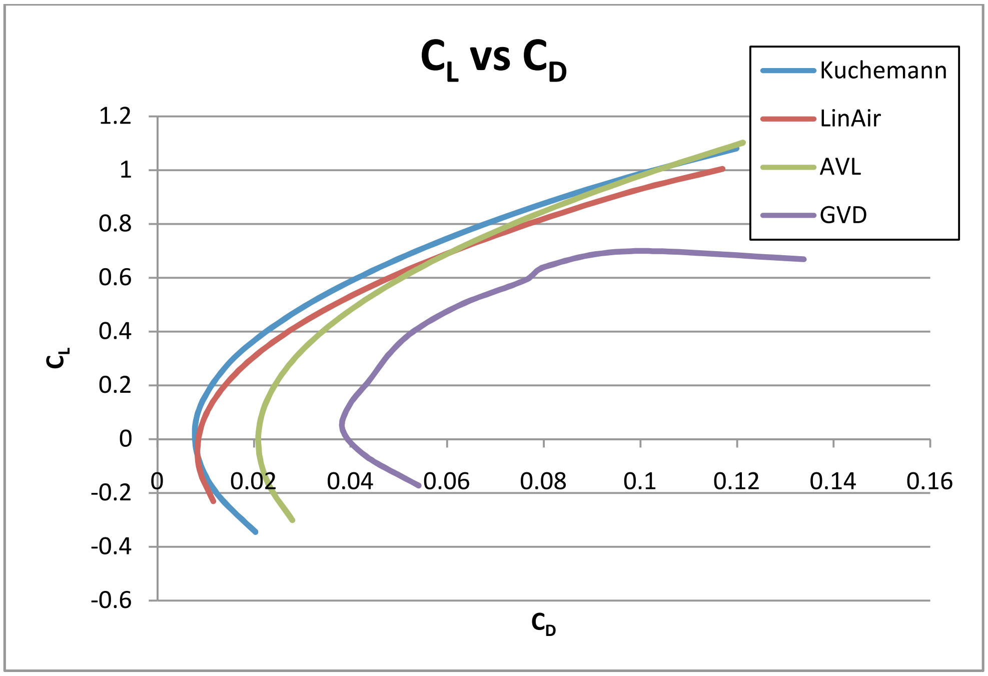
Figure, 1.4 Lift Coefficient Curve for Estimation



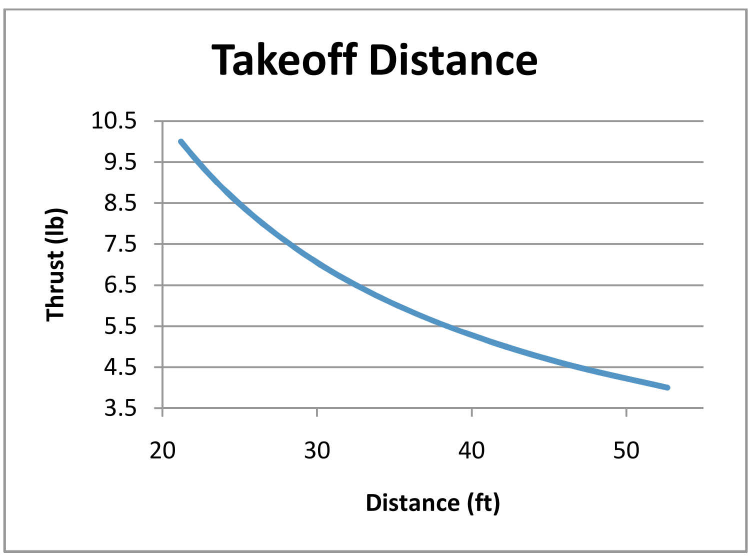
Figure, 1.5



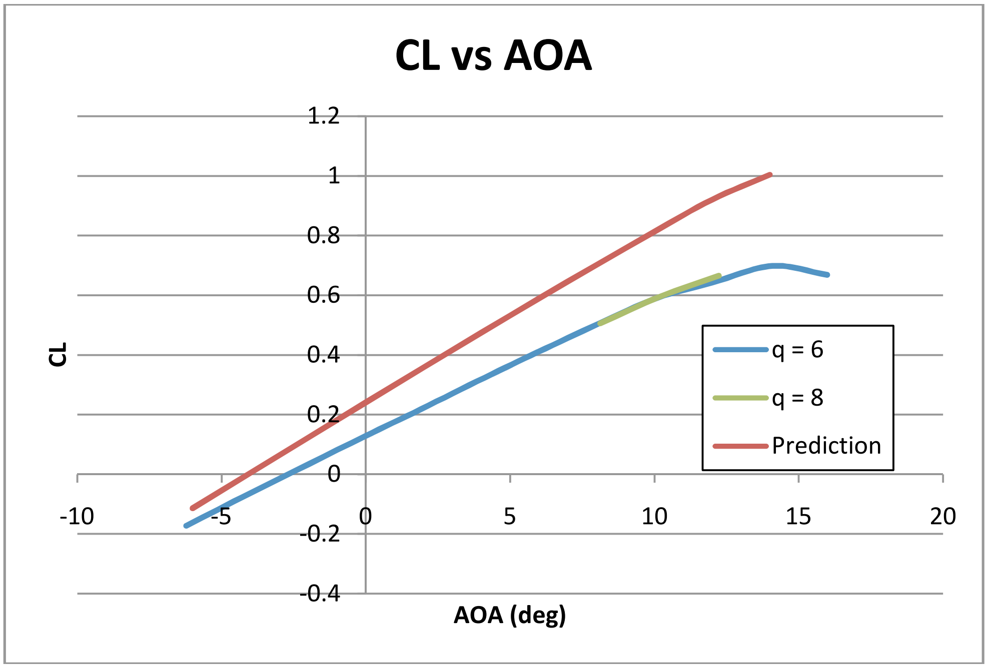
Figure, 1.6 Lift Coefficient Curve Comparison between Ground Validation and Estimations



Figure, 1.7 Drag Polar Comparison of Ground Validation and Estimations



Figure, 1.8



Figure, 1.9 Lift Coefficient Curve at different q numbers from Ground Validation

Elevation (ft)	1300
Density (slugs/ft^3)	2.288×10^{-3}
Viscosity (slugs/ft*s)	3.71×10^{-7}
Airfoil	NACA4418
Reynolds Number	300,000
Takeoff Weight (lb)	7.5
Landing Weight (lb)	5.5
Wing Span (ft)	4.5
Wing Surface Area (ft^2)	5.17
Aspect Ratio	3.91
Aerodynamic Center (%)	24.47
$C_{L\ MAX}$	1.006
$C_{L\ \alpha=0}$	0.278
$\alpha_{\ CL=0}$ (degree)	-4.313
$C_{d,0}$	0.019
$C_{m,0}$	-0.096
$V_{\ cruise}$ (fps)	69.71
$V_{\ stall}$ (fps)	36.64
$V_{\ takeoff}$ (fps)	43.97
$V_{\ landing}$(fps)	47.63
Landing Distance (ft)	85.33
Turn Radius before mission	40.71
Turn Radius after mission (ft)	29.85
Bank Angle (degree)	75.52

Table1, Data List

	NACA	LinAir	ROSKAM	GVD
$C_{D\ Total}$	0.028	0.029	0.055	0.040

Table 2 Drag Data List

Stability and Controls

Component	Weight (lbs)	x location (in)	Aircraft cg(in)	Item CG location relative to aircraft CG	moment about aircraft cg(lb.in)
Motor + Prop	0.80	-6.00	6.52	-12.52	-10.02
Wing	1.75	10.00		3.48	6.09
2 aileron servos	0.18	10.00		3.48	0.63
Nose gear(including wheels)	0.30	5.30		-1.22	-0.37
tail dreagger(including wheels)	0.03	43.84		37.32	0.93
Booms	0.25	21.55		15.03	3.76
Horizontal tail	0.08	43.84		37.32	2.99
Vertical tail	0.06	43.84		37.32	2.24
Rudder servo	0.06	43.84		37.32	2.09
Elevator servo	0.09	39.50		32.98	2.89
batteries	0.85	0.50		-6.02	-5.12
reciever/control rods	0.25	1.00		-5.52	-1.38
bomb bay servos	0.09	2.00		-4.52	-0.41
speed controller	0.16	1.00		-5.52	-0.88

Table 2.1: components weights and their distances relative to the aircraft cg

CL,0	0.28	CL,δe	0.73	Cl,β	-0.17	Cl,p	-0.81	Cn,δa	-0.06
CL,α	3.82	CM,δe	-1.13	Cn,δr	-0.092	Cn,p	-0.59	Cy,r	0.97
CM,0	0.0962	Cy,β	-0.42	Cl,δr	-0.03	Cy,δa	0	Cl,r	-0.35
CM,α	-0.38	Cn,β	0.14	Cy,δr	0.19	Cl,δa	0.65	Cn,r	-0.09

Table 2.2 Static and dynamic stability derivatives

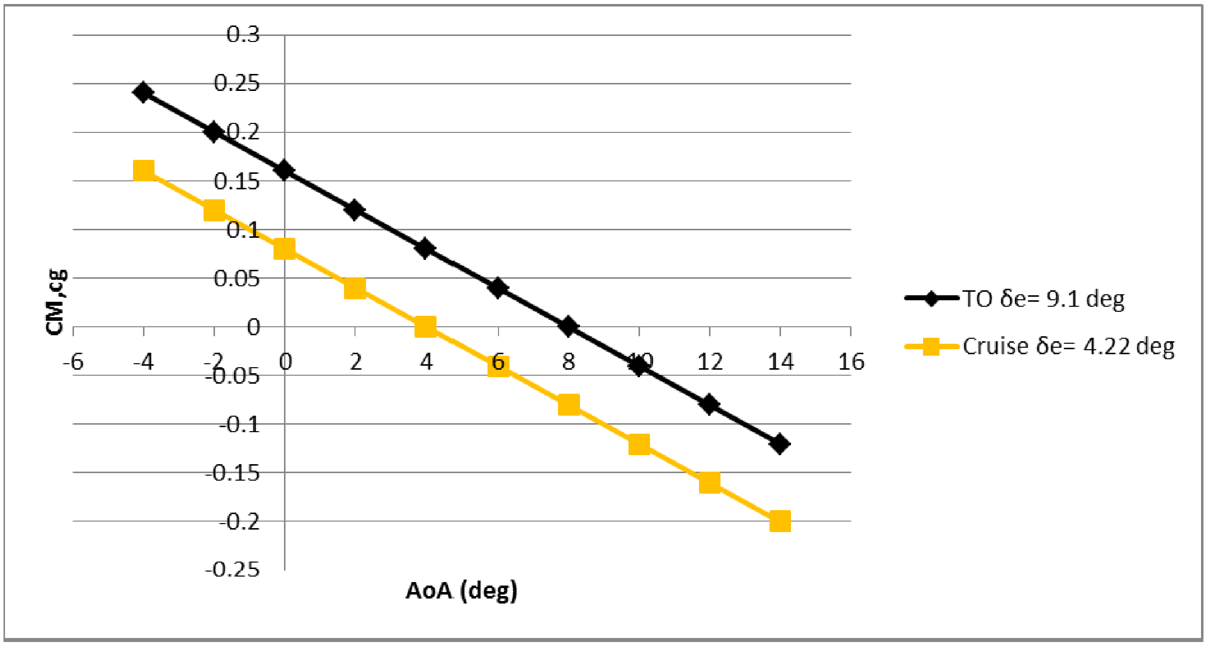


Fig. 2.1 trim plot for cruise conditions

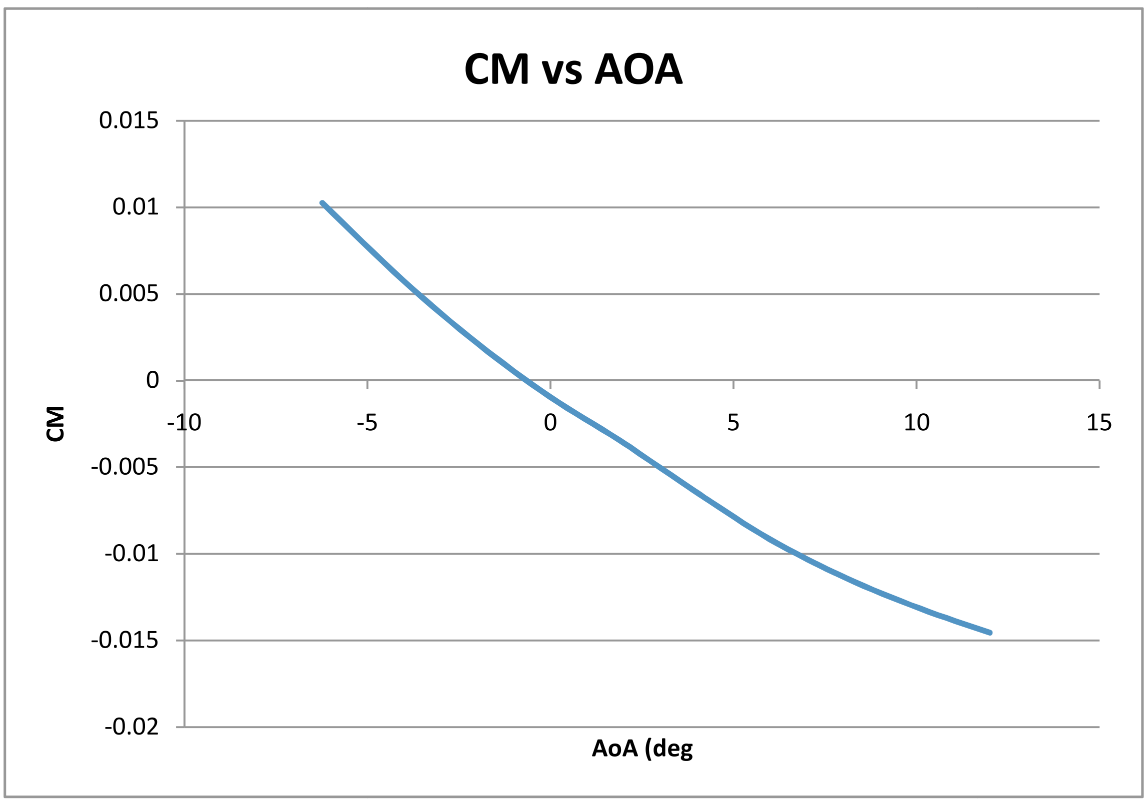


Fig. 2.2 Wind tunnel data trim plot

Component	Weight (lbs)	x location (in)	Aircraft cg(in)	Item CG location relative to aircraft CG	moment about aircraft cg(lb.in)
Motor + Prop	0.40	-3.00	7.70	-10.70	-4.28
Wing	1.63	8.78		1.08	1.76
2 aileron servos	0.18	8.78		1.08	0.19
Nose gear(including wheels)	0.26	5.30		-2.40	-0.62
tail dreagger(including wheels)	0.03	43.84		36.14	0.90
Booms	0.37	0.00		-7.70	-2.85
Horizontal tail	0.08	43.84		36.14	2.89
Vertical tail	0.06	43.84		36.14	2.17
Rudder servo	0.06	43.84		36.14	2.02
Elevator servo	0.09	39.50		31.80	2.78
batteries	0.95	0.50		-7.20	-6.84
reciever/control rods	0.25	1.00		-6.70	-1.68
bomb bay servos	0.18	2.00		-5.70	-1.03
speed controller	0.30	1.00		-6.70	-2.01

Table. 2.3 refined weight build up and CG estimate

Propulsion

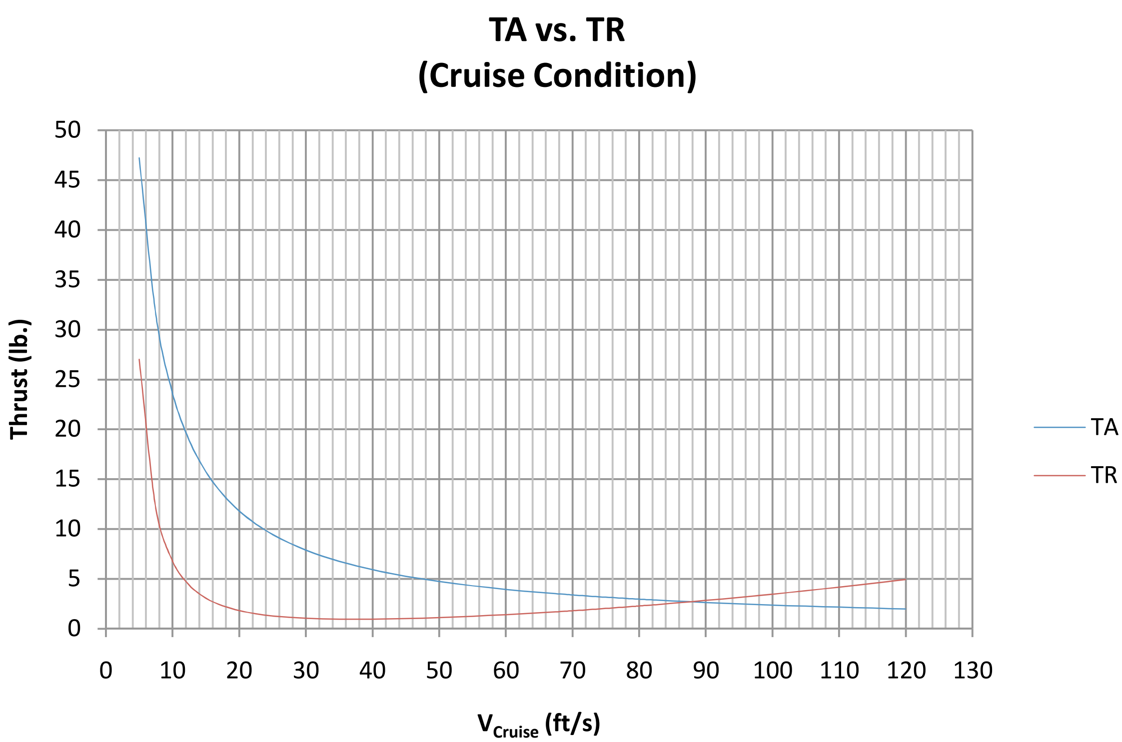


Figure 3.1

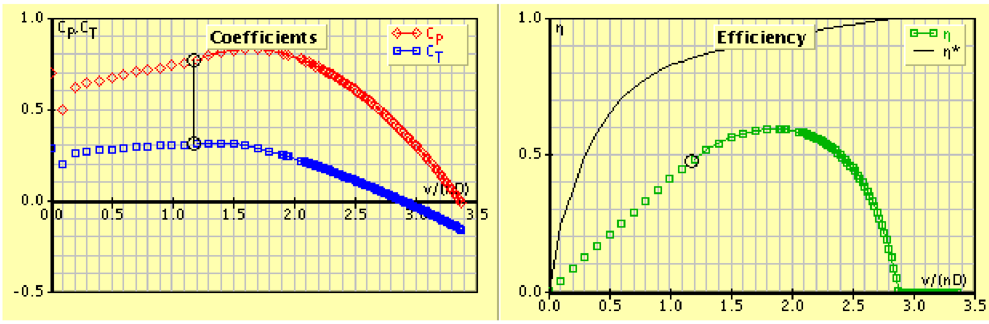


Figure 3.2
Javaprop Analysis at 5,000 RPM with APC 9x4.5E Propeller

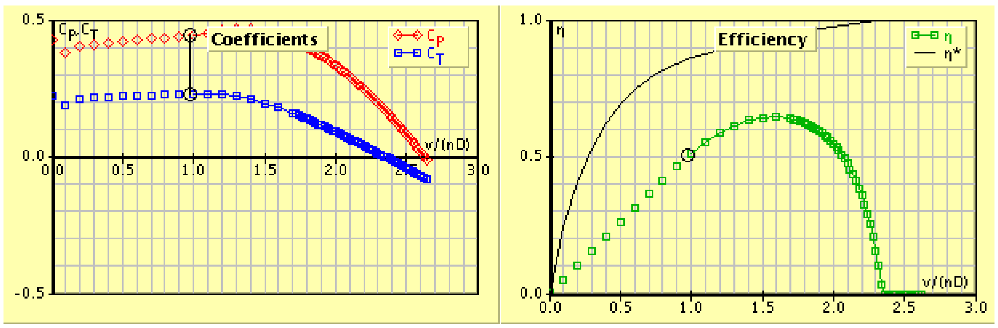


Figure 3.3
Javaprop Analysis at 6,000 RPM with APC 9x4.5E Propeller

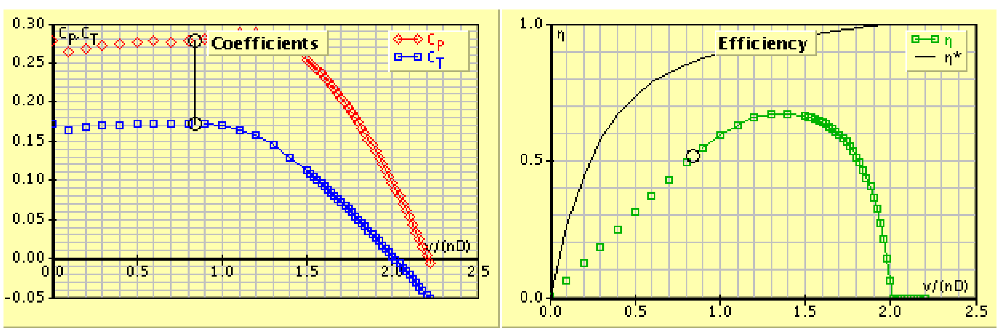


Figure 3.4
Javaprop Analysis at 7,000 RPM with APC 9x4.5E Propeller

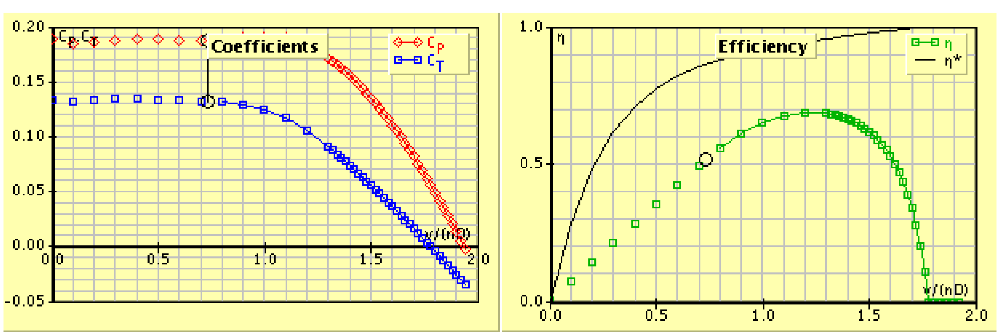


Figure 3.5
Javaprop Analysis at 8,000 RPM with APC 9x4.5E Propeller

Item	Weight
Propellers	1.3 oz.
Motors	6.5 oz.
Motor mounts	0.4 oz.
ESC’s	2.2oz.
Battery Packs	13.8 oz.
Fuse	0.25 oz.
Wires	3.8 oz.
TOTAL W_P	28.25 oz. ~ 1.76lb.

Table 3.1
Propulsion System Individual and Total Weight

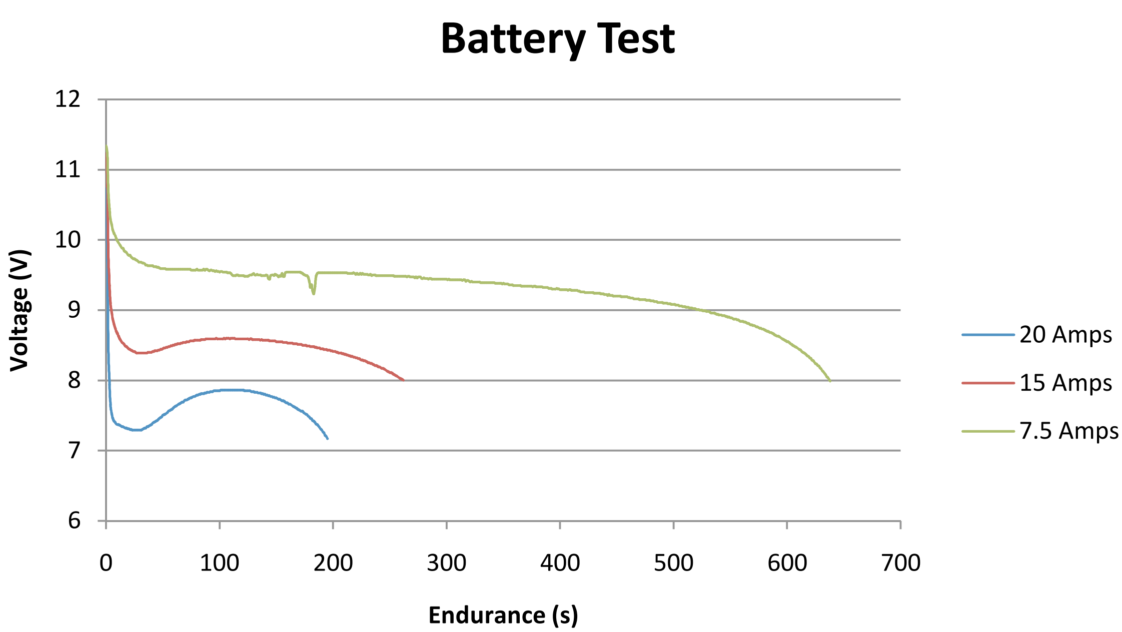


Figure 3.6

Name	Q (lb/ft^2)	Thrust (lb.)
Static Thrust	0	1.0755
Cruise	6	2.6238
Maximum	9	3.8175
Landing	1	1.0251

Table 3.2
Wind Tunnel Thrust Test Data

Structures

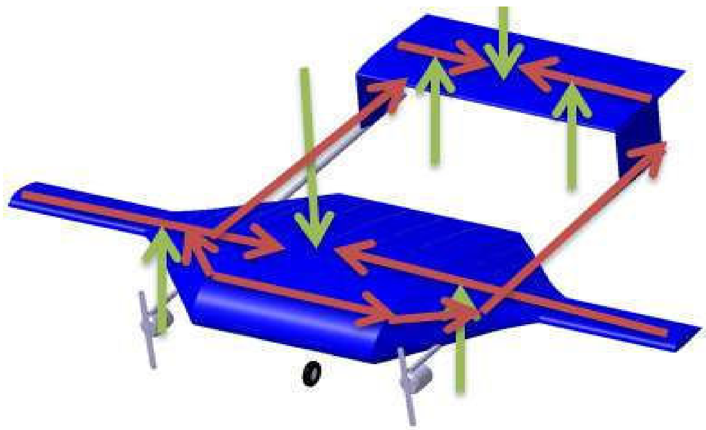


Fig:4.1 Major Load Paths on the aircraft.

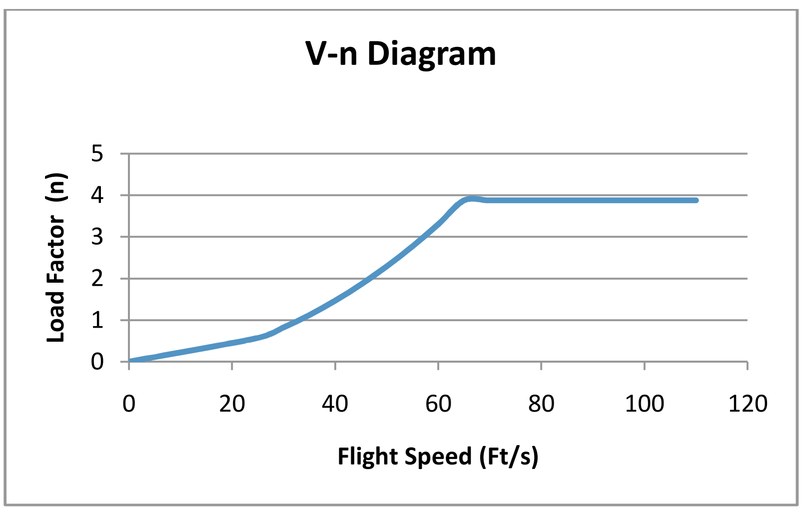


Fig: 4.2 V-n Diagram and load factor estimations.

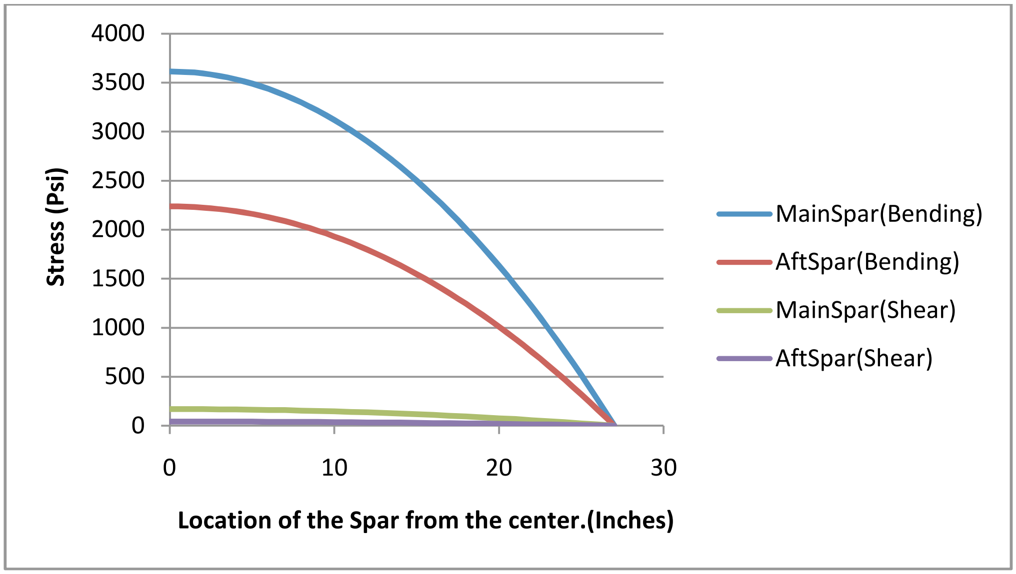


Fig: 4.3 Bending and shear stress distribution on the spars.

Bending along the span at the maximum load situation.						Main SPAR		
Section Properties								
Material		I-Beam	Area(sqr in.)	Volume	Weight	MOI(x)	MOI(y)	Deflection(in)
Spruce	b	0.7500	0.3125	16.8750	0.2700	0.0212	0.0043	1.1259
	h	0.5000						
	H	0.7500						
	t	0.2500						
Bending along the span at the maximum load situation.						Aft Spar		
Section Properties								
Material		I-Beam	Area(sqr in.)	Volume	Weight	MOI(x)	MOI(y)	Deflection(in)
Spruce	b	0.4375	0.1484375	8.015625	0.12825	0.005694		1.045958691
Balsa	h	0.3125						
	H	0.5625						
	t	0.125						

Table 4.1

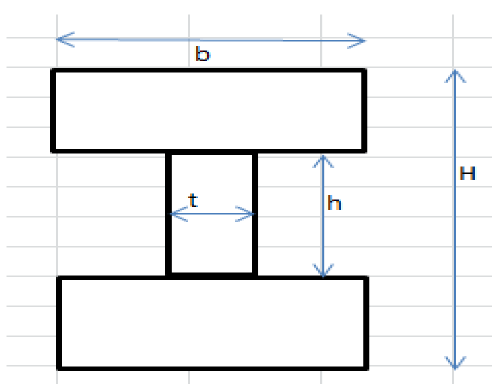


Fig 4.4 Section properties and deflections on the main and aft spars.

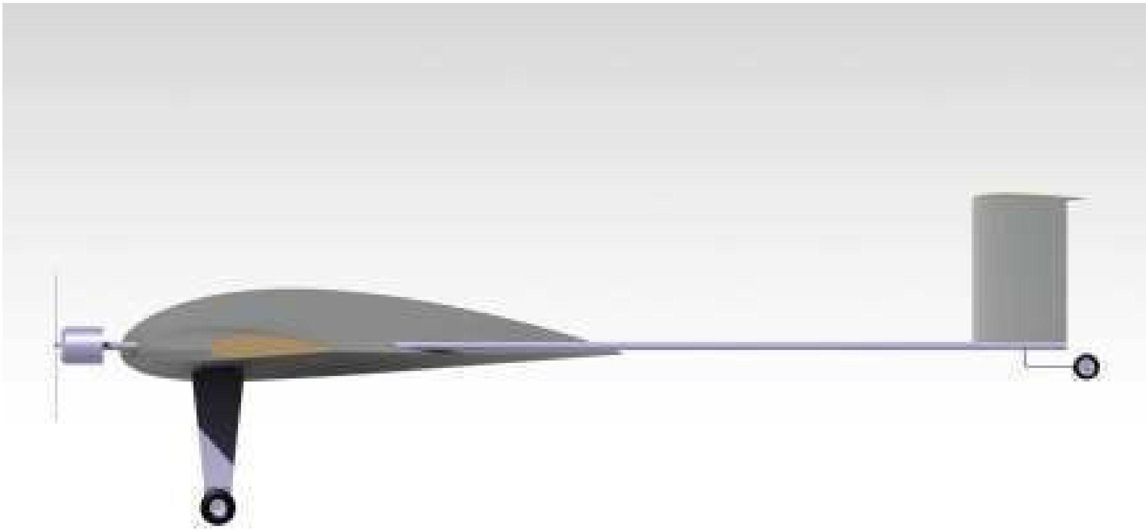


Figure 4.5 Side view of the Aircraft showing landing gear layout

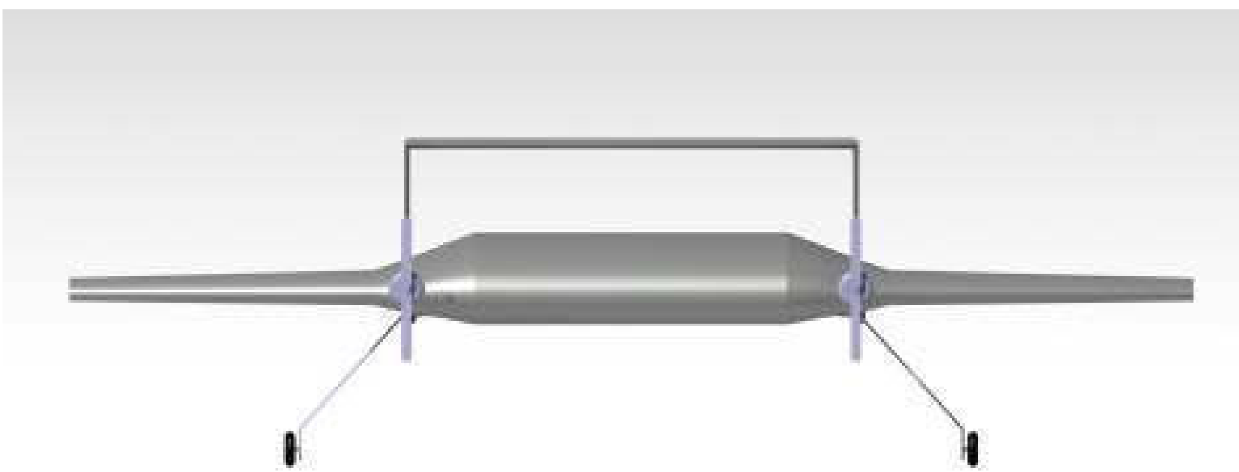


Figure 4.6 Front view showing landing gear layout

Payload

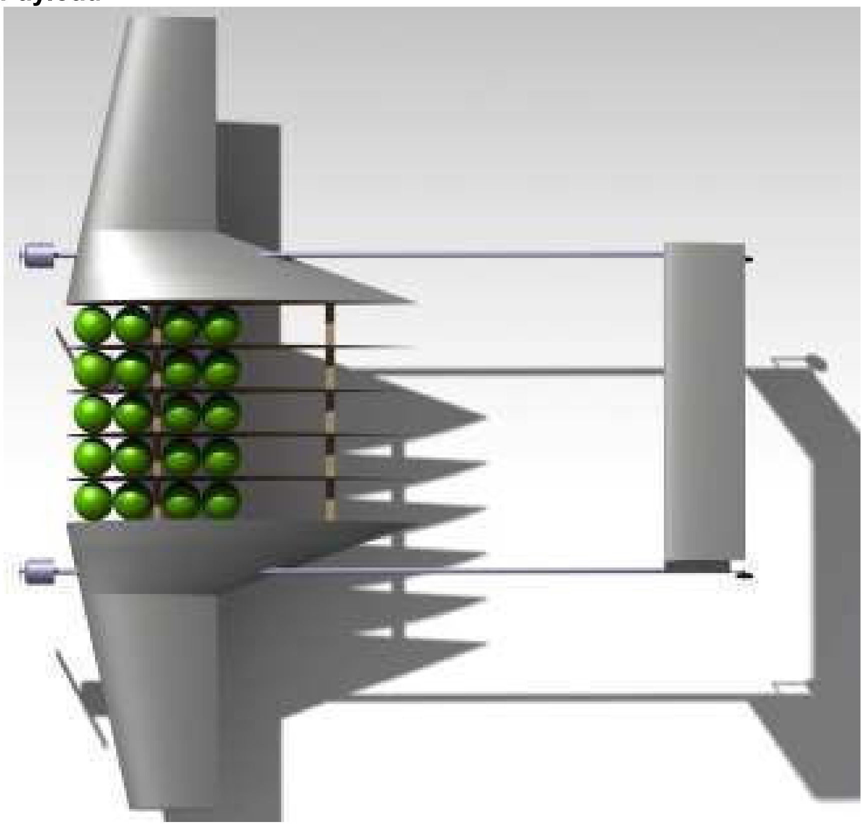


Figure 5.1-payload layout

Figure 5.2 Payload Bay Mockup Open

Figure 5.3 Payload Bay Mockup Closed

Project Costs

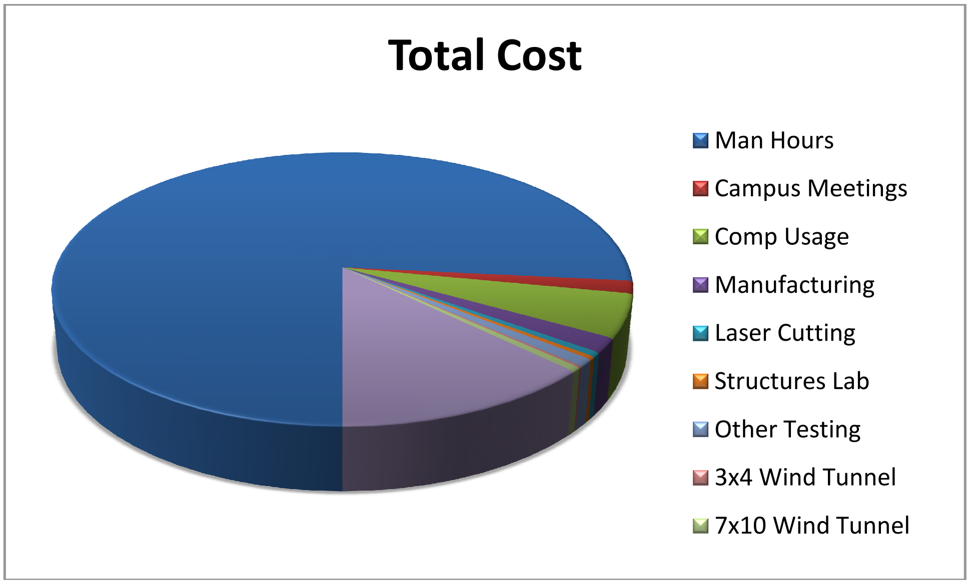


Figure 7.1 visual breakdown of total project costs

Department	Total Cost
Man Hours 18 hrs/wk=600 hrsx5 members Total: 3000 hrs.	$225,000
Campus Meetings 175 Hours	$4,375
Comp Usage 120 hrs./member=600 Hrs. Total	$15,000
Manufacturing 50 Hours	$5,000
Laser Cutting 15 Hrs.	$1,500
Structures Lab 10 Hrs.	$1,000
Other Testing 25 Hrs.	$2,500
3x4 Wind Tunnel 5 Hrs.	$500
7x10 Wind Tunnel 3 Hrs.	$1,500
Specialized Software 75 Hrs.	$37,500
Total	$293,875
Grand Total (With Operating Costs)	**$440,812.5**

Table 7.1 Breakdown Project Cost Estimates

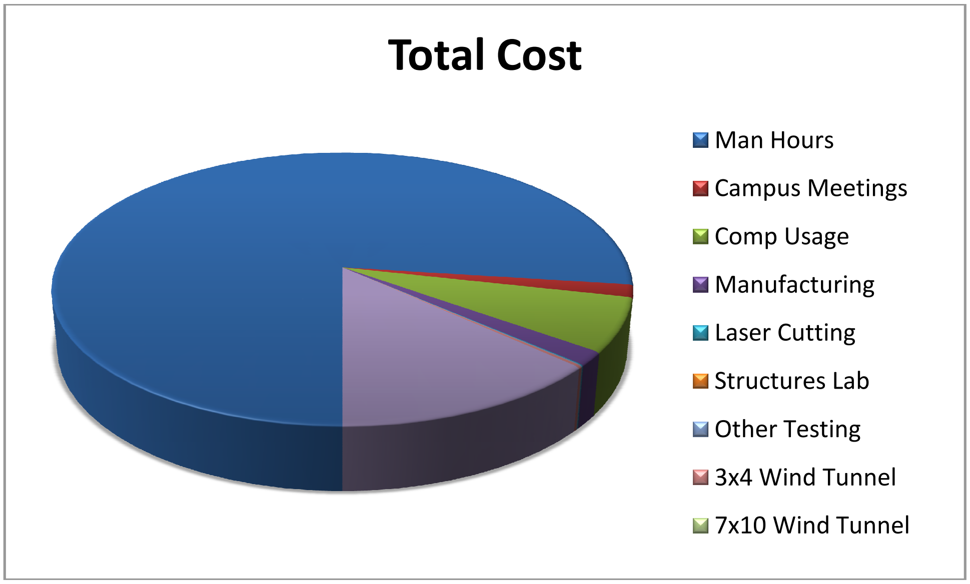
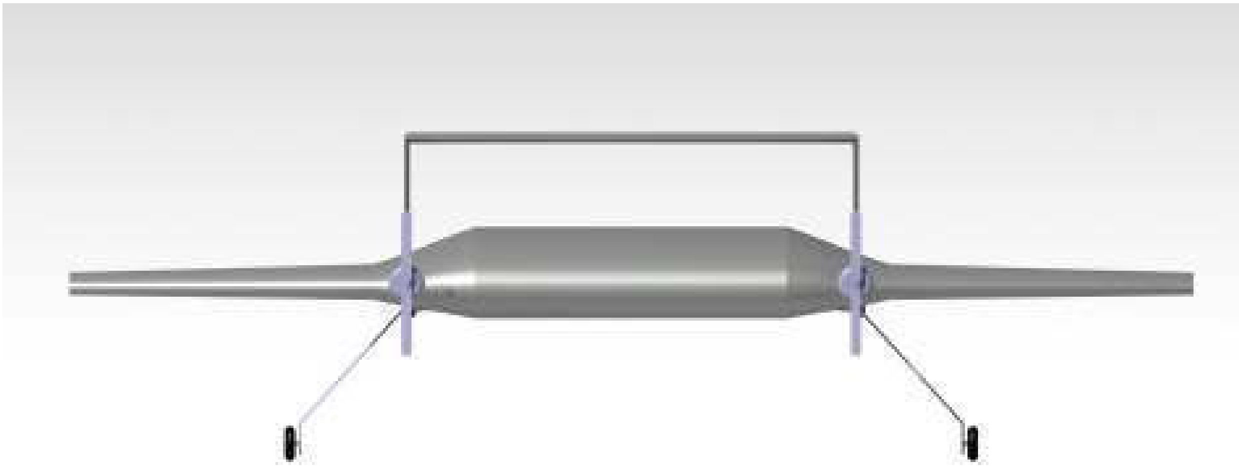


Figure 7.2 Breakdown of actual Total Project Costs

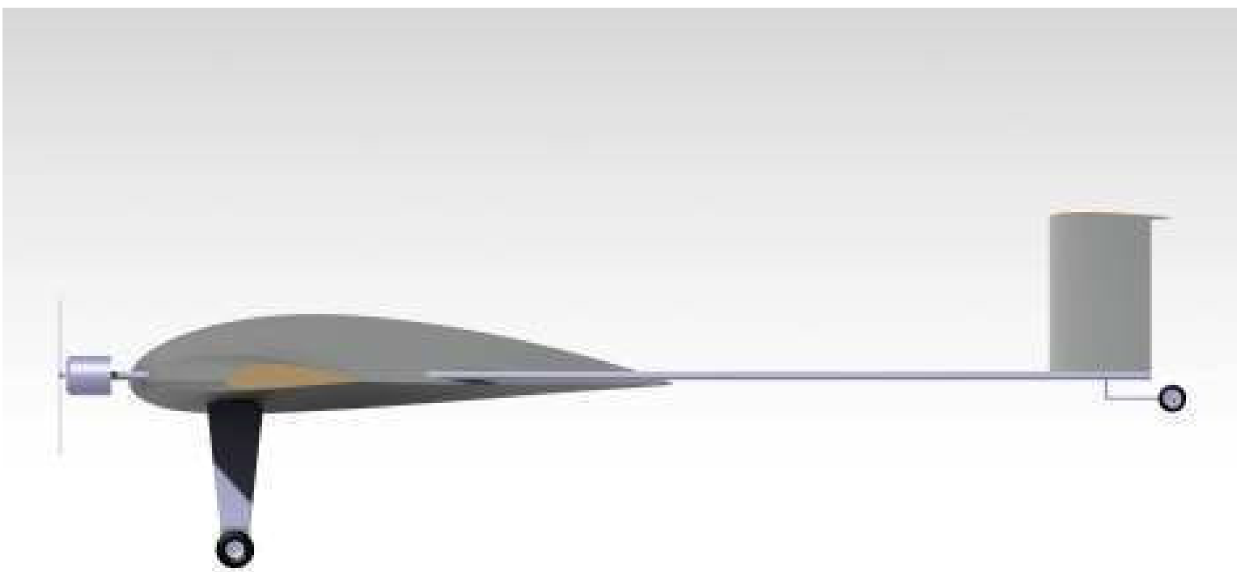
Department	Total Cost
Man Hours 18 hrs/wk=604 hrsx5 members Total: 3000 hrs.	$226,500
Campus Meetings 172 Hours	$4,300
Comp Usage 149.6 hrs./member=748 Hrs. Total	$18,700
Manufacturing 46 Hours	$4,600
Laser Cutting 2 Hrs.	$200
Structures Lab 0 Hrs.	$0
Other Testing 2 Hrs.	$200
3x4 Wind Tunnel 6 Hrs.	$600
7x10 Wind Tunnel 0 Hrs.	$0
Specialized Software 78 Hrs.	$39,000
Total	$294,100
Grand Total (With Operating Costs)	**$441,150**

Table 7.2 Breakdown of Actual Total Project Costs

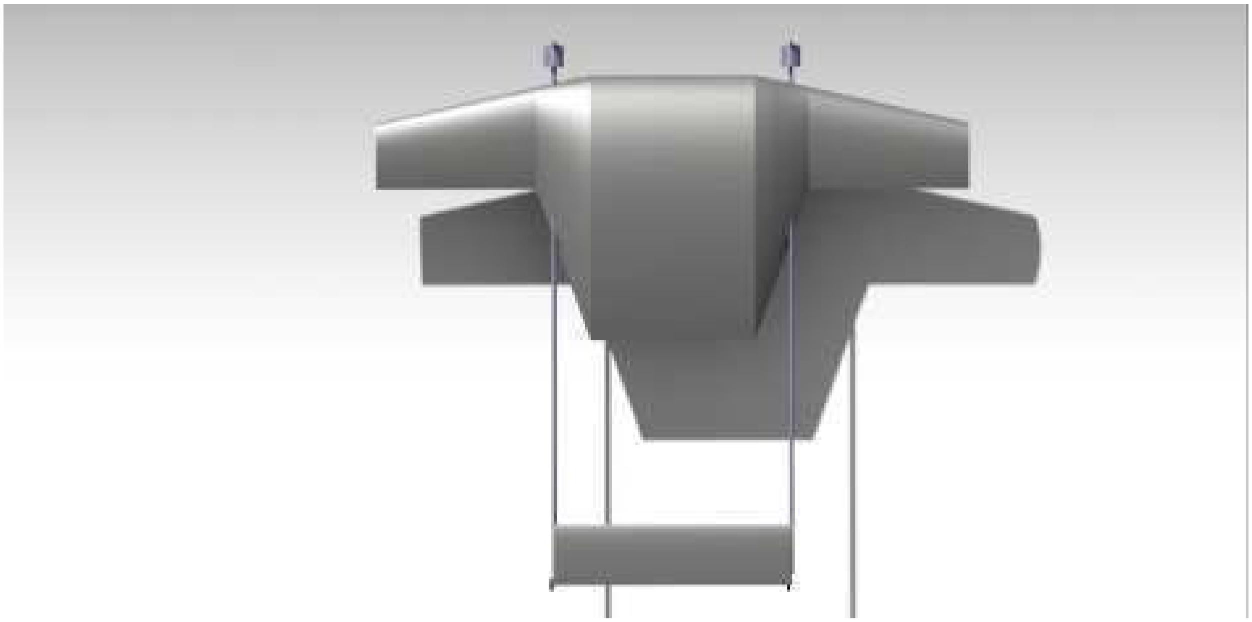
Vehicle Drawings



Assembled Front View

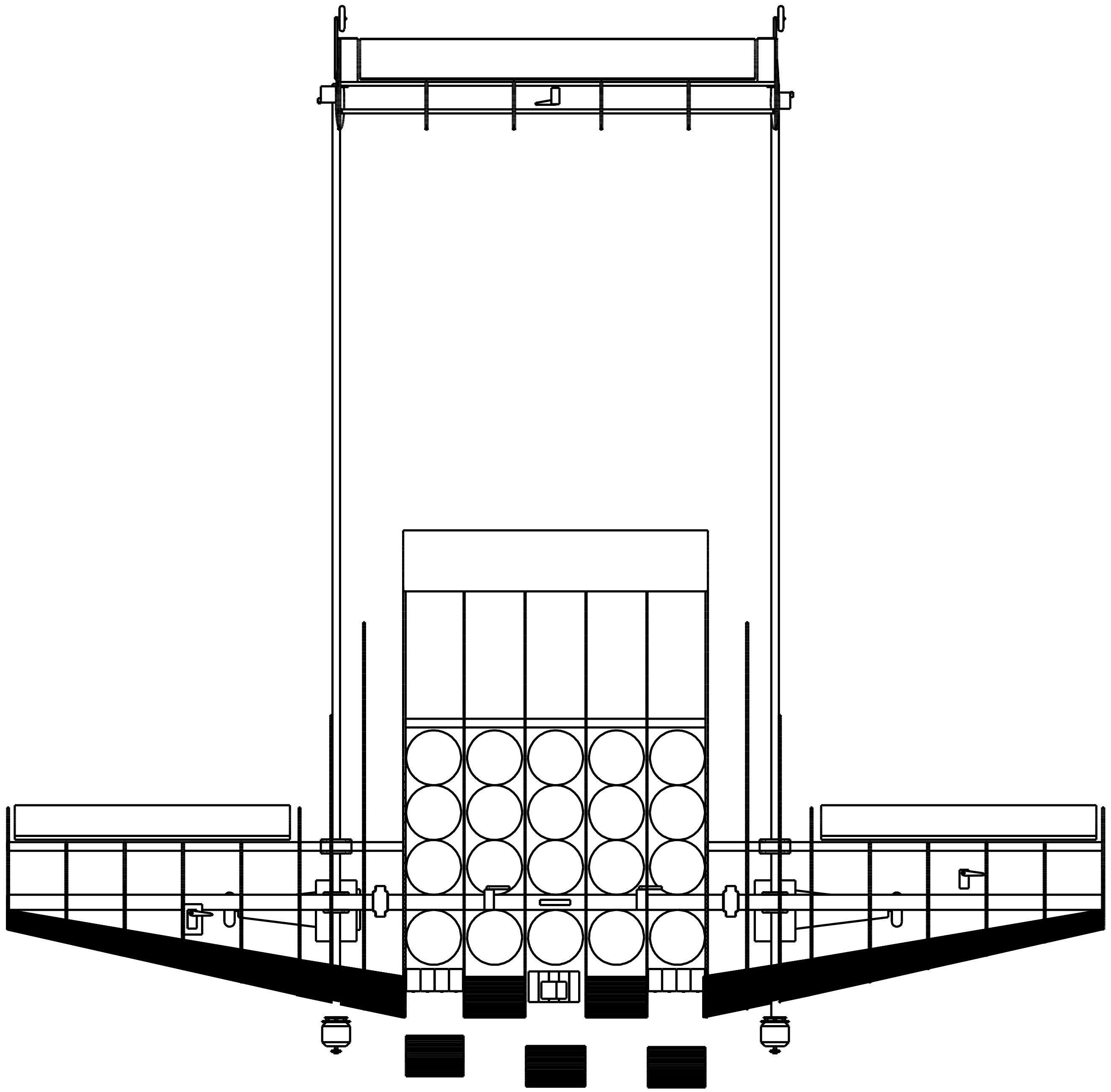


Assembled Side View

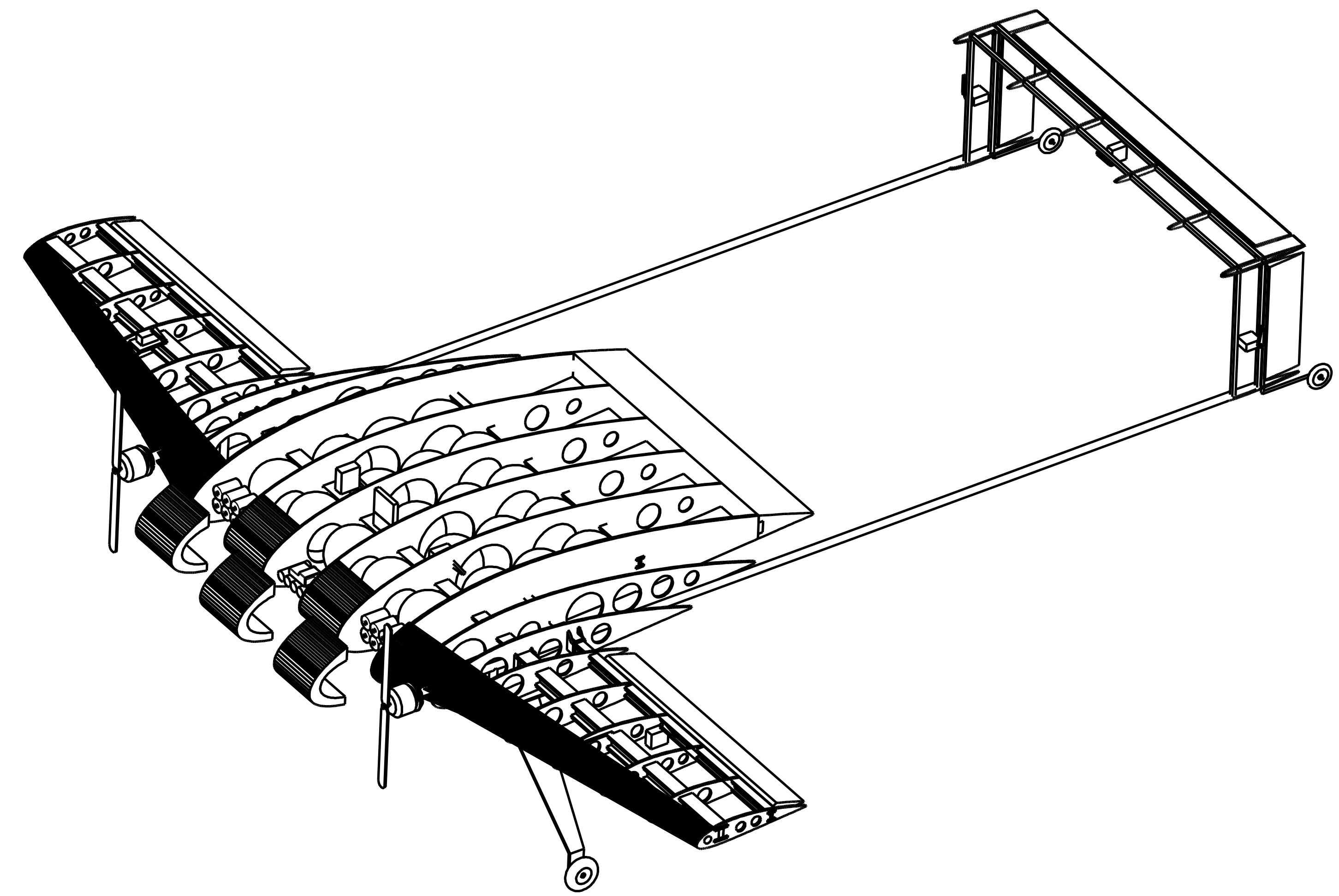


Assembled Top View

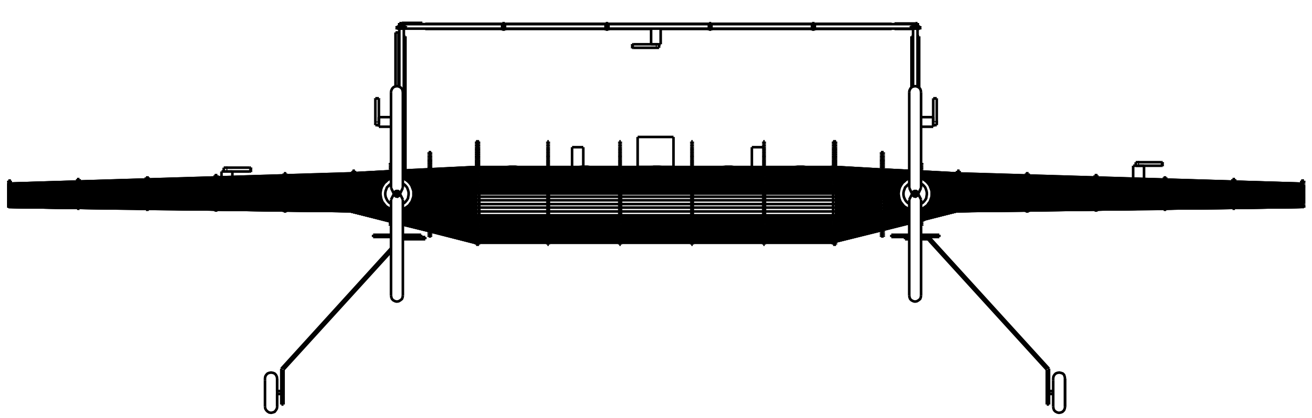
Assembled Isometric View



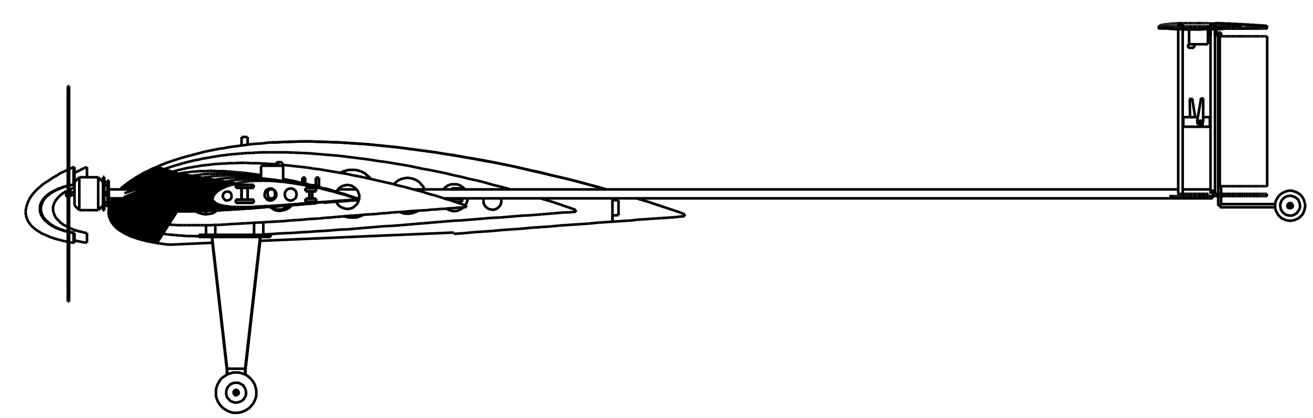
Top view
Scale: 1:8



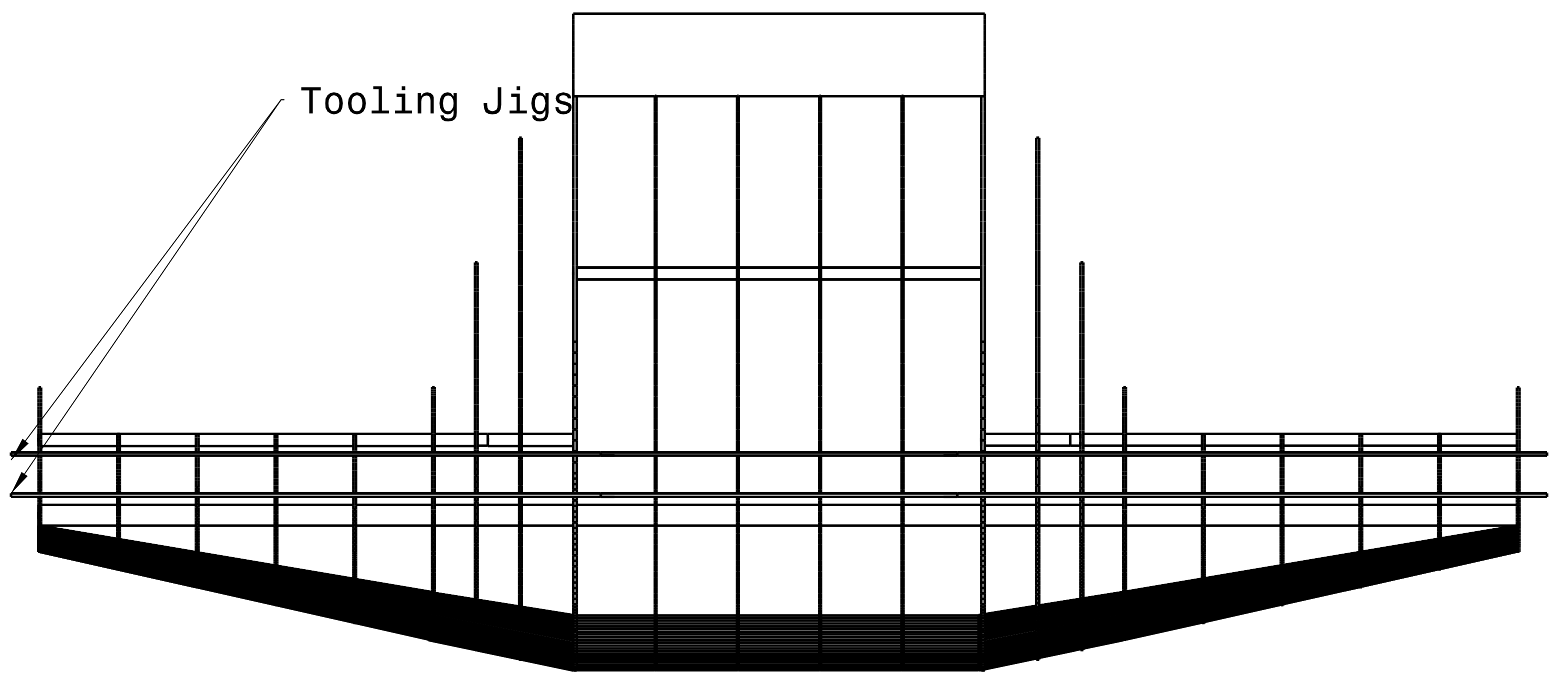
Isometric view
Scale: 1:8



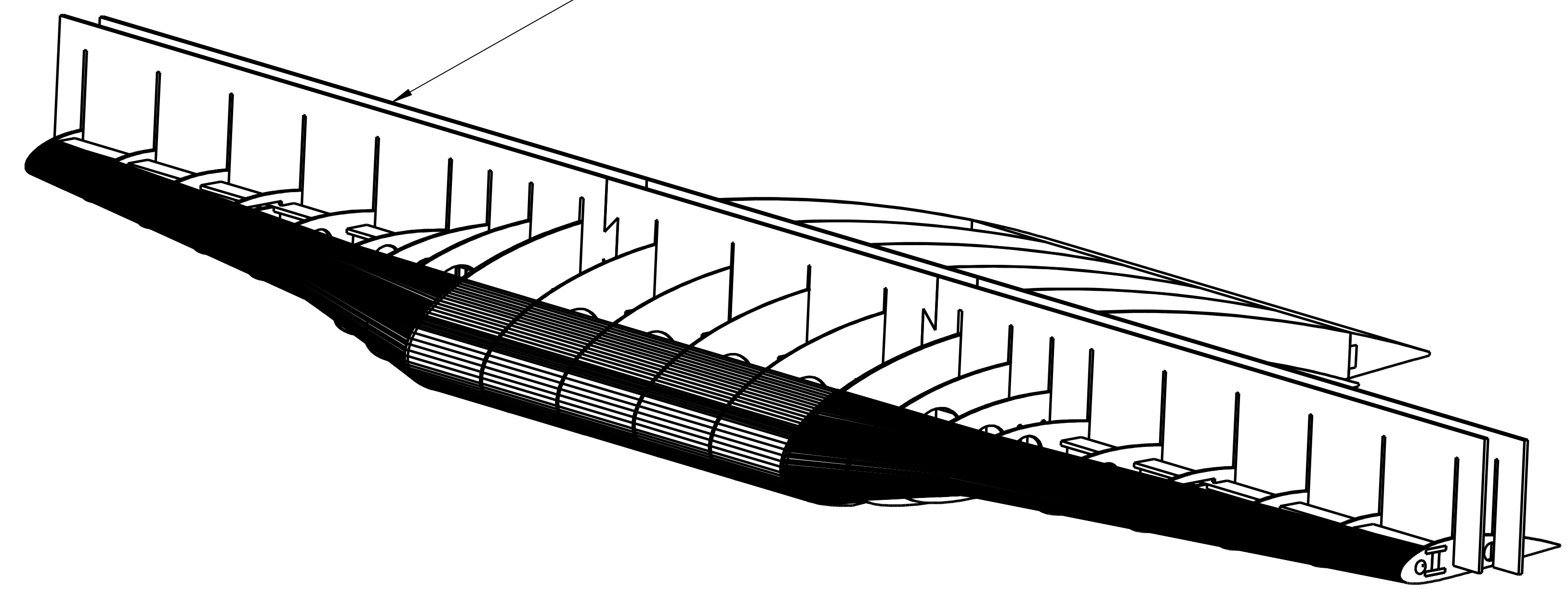
Front view
Scale: 1:8



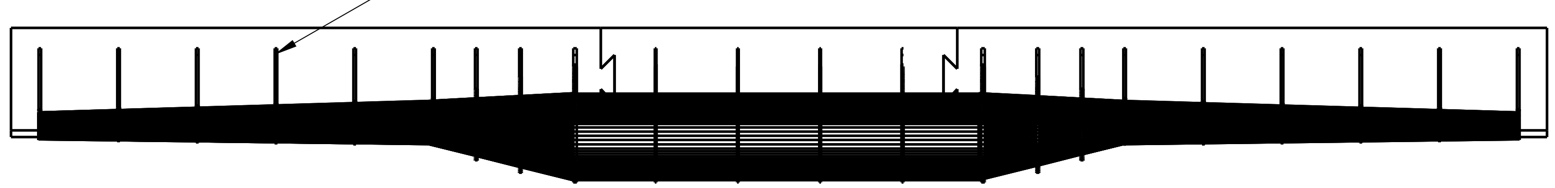
Right view
Scale: 1:8



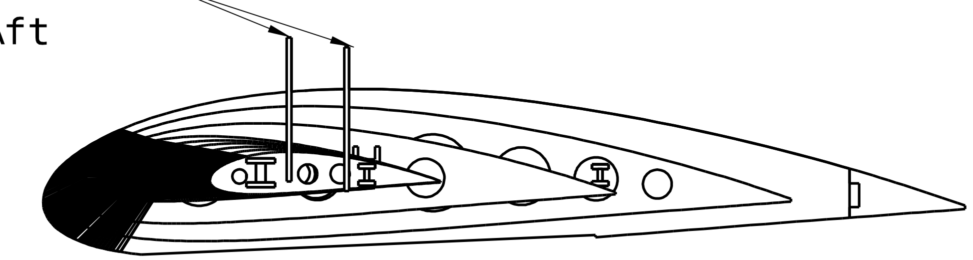
Top view
Scale: 1:7



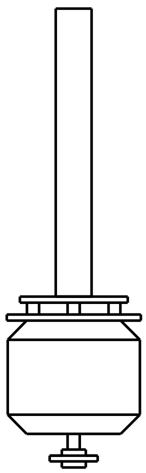
Isometric view
Scale: 1:5



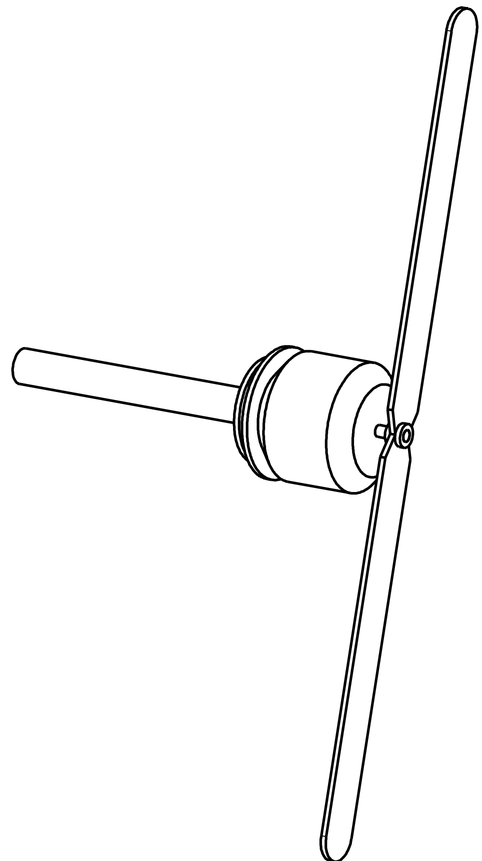
Front view
Scale: 1:7



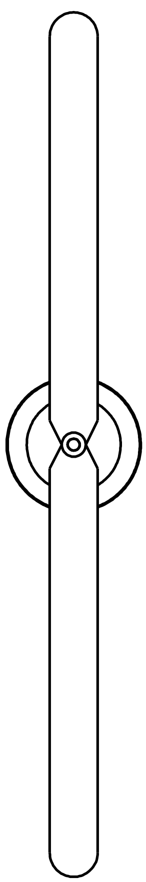
Right view
Scale: 1:5



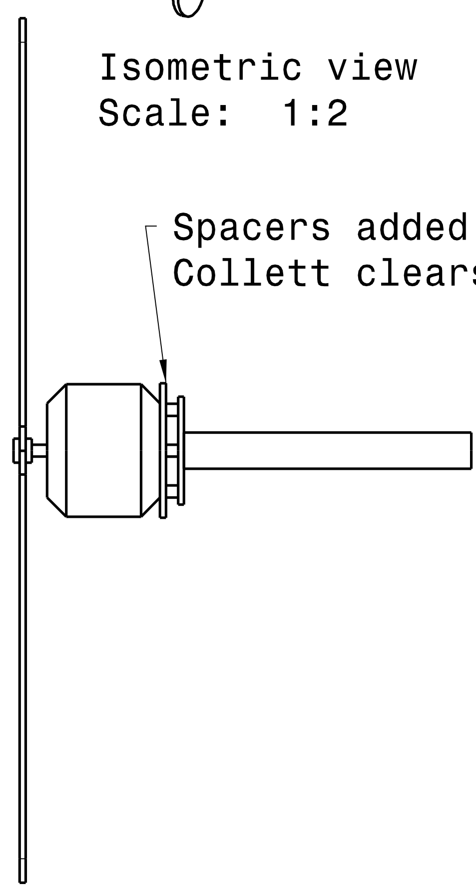
Top view
Scale: 1:2



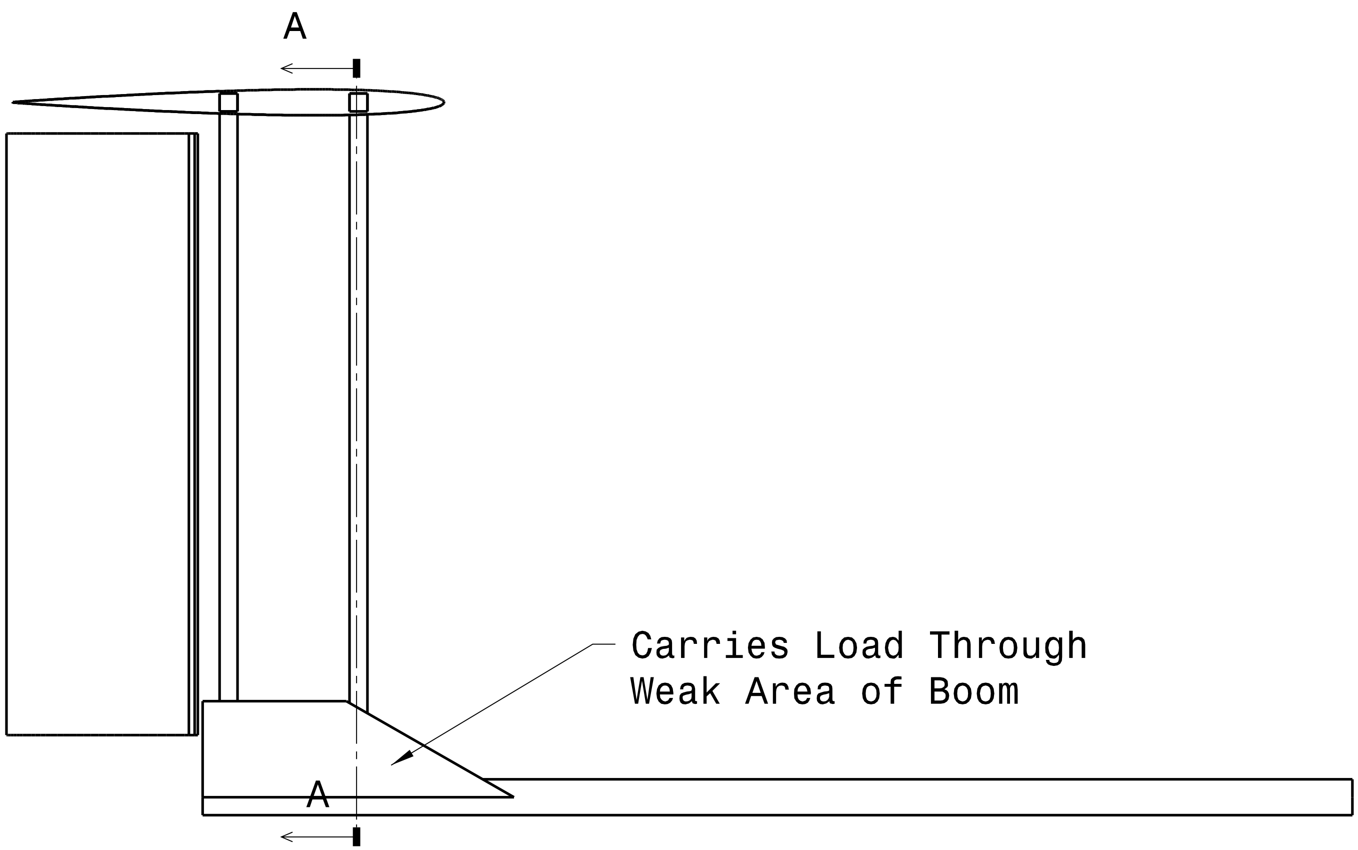
Isometric view
Scale: 1:2



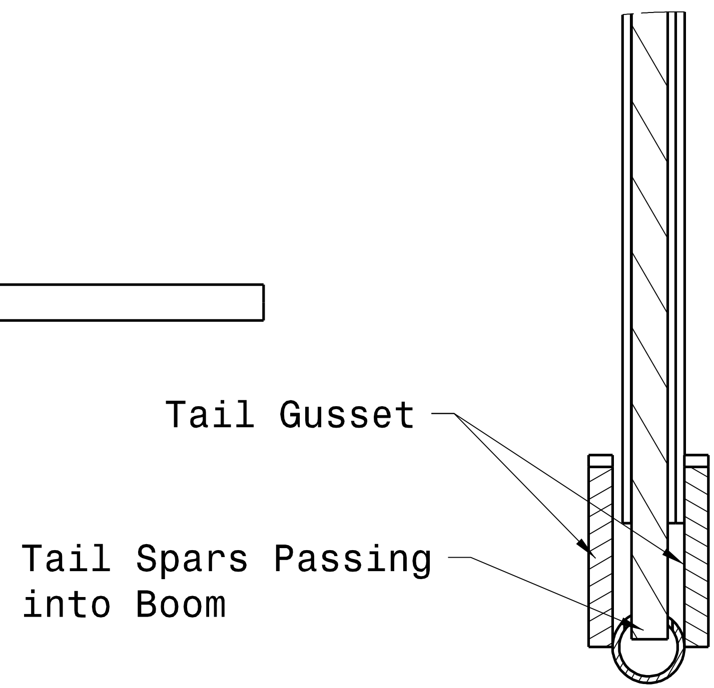
Front view
Scale: 1:2



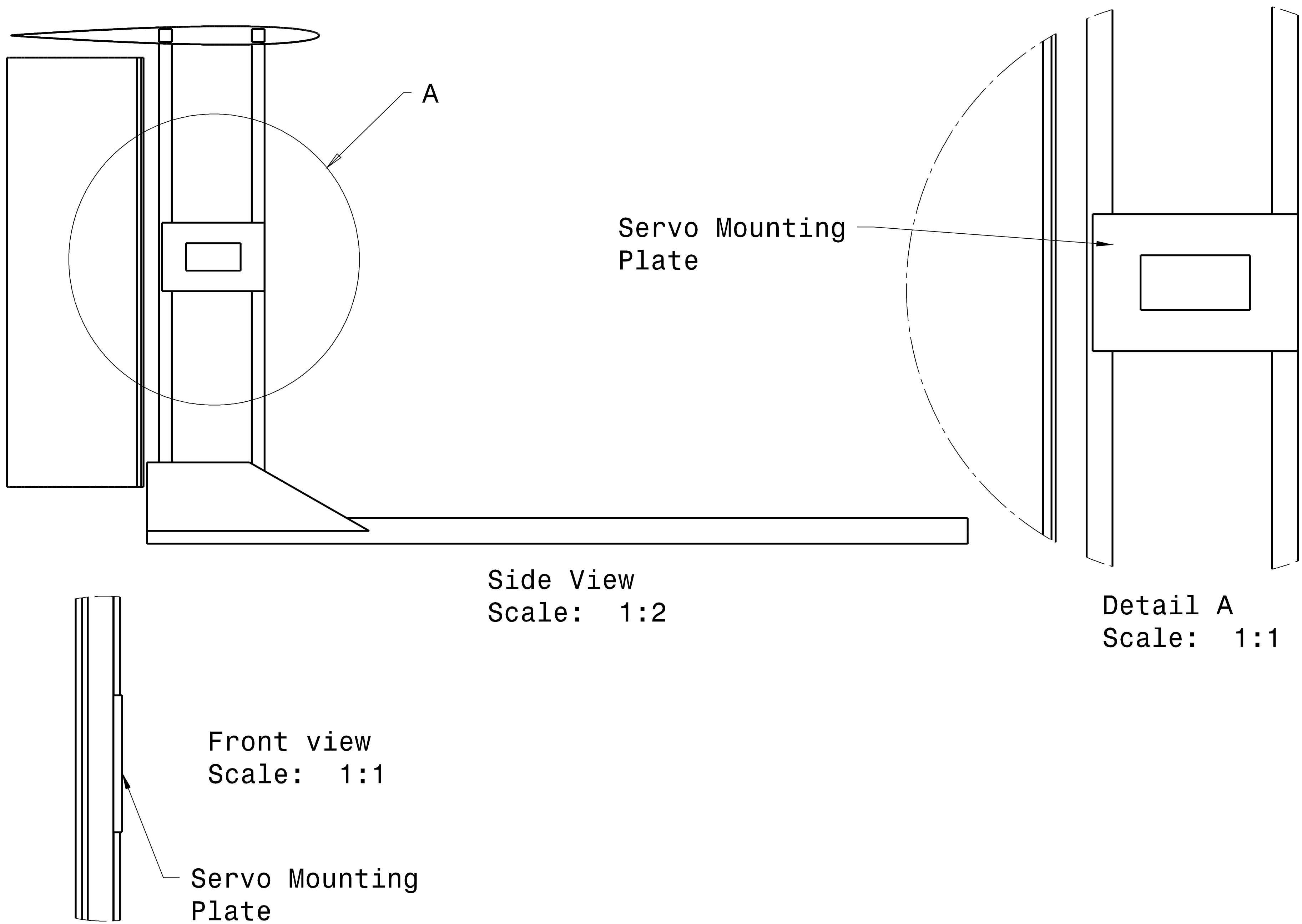
Right view
Scale: 1:2



Front view
Scale: 1:2



Section view A-A
Scale: 1:1


A
Servo Mounting
Plate
Side View
Scale: 1:2
Detail A
Scale: 1:1
Front view
Scale: 1:1
Servo Mounting
Plate