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1. Executive Summary

1.1 Design Summary

This document details the design, testing, and manufacturing efforts of University of California San Diego Team TLAR Velox in preparation for the 2009-2010 AIAA/Cessna/RMS Design/Build/Fly (DBF) competition. Our team's primary objective is to design the winning DBF aircraft by maximizing total score based on the equation provided by the competition.

1.2 Design Development and Alternatives

Flight score is determined by performance in three missions: Ferry Flight, Internal Payload Flight (Softballs), and External Payload Flight ("Bats"). The Ferry Flight and External Payload Flight focus on the lap time of the aircraft whereas the Internal Payload Flight focuses on the loading time of the Softballs. Both the Ferry Flight and the Internal Payload Flight are normalized by a weight factor while the External Payload Flight is normalized by a factor incorporating the number of "Bats" carried. Design requirements constrain aircraft size to a box with maximum dimensions of 2'x2'x4' and a takeoff distance with a maximum of 100'. A sensitivity study on the flight score reveals total system weight and speed as the most critical design parameters.

Vehicle concepts are developed to meet mission requirements and achieve a maximum flight score. Initial configurations for analysis are based on ball grid patterns including: Conventional (long body/wide body patterns), flying wing, and twin boom designs. Figure of Merit analyses were utilized to select aircraft configuration and components, which yielded a conventional (long body) configuration with a single tractor propeller, and tricycle landing gear. Each component is selected to provide sufficient aircraft performance, while minimizing system weight, drag and ball loading time.

A modular, multi-disciplinary optimization architecture was developed and each engineering team performed analysis on design parameters including: airfoil selection, wing span, propulsion selection, structural efficiency, and flight stability.

1.3 Optimized Design Summary

The optimized system design integrates a removable wing aircraft with easy access to internal payload containers to meet size requirements and provide quick loading time. This combined with a light weight airframe and low drag design give us and optimum flight score. The wings are sized to a 72.25 in. wingspan with a total wing area of 680in2 and a Eppler 214 airfoil to meet takeoff distance requirements. A symmetric airfoil is selected for the horizontal and vertical tails and sized to provide the required stability. Propulsion optimization suggests that a Neu 1506/2Y motor geared to 6.7:1 provides optimal performance for all three missions, and the combination of modular battery packs with 18x200mAhr cells yields the highest score. An open frame carbon fiber box provides the required strength for pre-mission tests and low weight to minimize the weight factor for the Ferry Flight and Internal Payload Flight. The fuselage length is 57in including a 9in removable empennage.

1.4 System Performance and Capabilities

The predicted performance capabilities are as follows. Mission 1(Ferry Flight): total flight time 71 seconds; thrust-to-weight ratio .45; wing loading 1.33 lb/ft2; cruise speed 70ft/s. Mission 2(Internal Payload Flight): payload loading time 10.2 seconds; thrust-to-weight ratio .43; wing loading 2.06 lb/ft2; cruise speed 46.2ft/s. Mission 3(External Payload Flight): total flight time 152.8 seconds; thrust-to-weight-ratio 0.43; wing loading lb/ft2; number of “bats” _____.

2. Management Summary

2.1 Design Team Organization

The 2010 team consists of _____ members ranging from freshmen to seniors. The team organizational chart is shown below.

The team is headed by a project manager and six technical lead engineers who are collectively responsible for the management and coordination of their individual teams. The objective of the performance group is to optimize the aircraft configuration for maximum possible flight score. The aerodynamics group is responsible for deciding and calculating wing, stabilizer, and other aerodynamic parameters. The propulsion team is responsible for the optimization of the propulsion system and the analysis of power requirements. The structures and materials team is responsible for material selection and analysis in order to produce the most structurally efficient configuration. The flight controls team is responsible for stability analysis and CAD modeling for center of gravity analysis. Converting design parameters into the working aircraft is the responsibility of the fabrication team.

2.2 Personnel Assignment Chart

2.3 Milestone Chart

The figure below shows the planned and actual timeline of the

Figure 2.3.1: Timeline of both planned and actual design milestones.

3. Conceptual Design

3.1 Mission Summary

Competition scoring is based on four categories: the written report, and performance on Mission 1 (the ferry flight), Mission 2 (the softball payload flight), and Mission 3 (the bats payload flight). The total score is calculated as:

Total flight score is determined by the sum of the best scores from each mission. Teams are allowed a maximum of either 5 flight attempts or 4 successful scoring flights, whichever comes first. For each mission, the aircraft must take off within 100 feet.

In addition, the airplane must fit into a team-constructed 2' x 2' x 4' case. The weight of the case and airplane are measured before each flight attempt, and the sum of both are reflected in the weight ratio for scoring. Although the aircraft need not be fully assembled to fit in the case, the team must be able to reassemble the aircraft within a 5 minute period. Additionally, the airplane battery pack must weigh less than 4 pounds.

Mission 1

For Mission 1, the aircraft must complete 2 full laps around the course. There is no payload for the mission, so a high score is earned through a low empty aircraft weight and fast flight time. Scoring for mission 1 is calculated as:

where and . The terms and are defined as the lowest time and weight recorded for a finishing team in Mission 1. is the time recorded by the team during Mission 1, while is the heaviest weight for that team from any mission attempt.

Mission 2

For Mission 2, the aircraft must internally carry a random mix of 6 to 10 softballs, of 11" or 12" circumference. This mission emphasizes the need to minimize the amount of time an aircraft spends on the ground. The timed section of the mission includes the loading of the aircraft and 3 full laps. Thus, careful design is necessary to achieve a swift loading time. Scoring for mission 2 is calculated as:

where and . The terms and are defined as the lowest time and weight recorded for a finishing team in Mission 2. is the time recorded by the team during Mission 2, while is the heaviest weight for that team from any mission attempt.

Mission 3

For Mission 3, the aircraft must externally carry a random mix of bats. The bats measure 26 to 30 inches long, with a nominal diameter of 2" and weight of 16 to 20 ounces. The timed section of the mission includes 3 full laps, measured from start of takeoff to completion of the third lap. The mission places emphasis on the need to carry a large payload over a large range. Aircraft and payload weight is not a consideration, so the score is maximized by a fast flight time and large payload. Scoring for mission 3 is calculated as:

where and . The terms and are defined as the lowest time and largest number of bats carried for a finishing team in Mission 3. and are the time recorded and bats carried by the team during Mission 3. The number of bats carried must be at least 1 and less than 5.

3.2 Selection Process

Two main factors were taken into consideration when weighing the different design options: the baseline requirements of the aircraft, and an in-depth look at the flight score breakdown.

Baseline Requirements

There were two main baseline requirements that were established for the aircraft. First, it had to fit in a 4’x2’x2’ box. Second, at the very least it had to be capable of potentially carrying ten, seven ounce softballs.

Flight Score Breakdown

Initially it appears that holding as many bats as possible would provide the best score. The reason for this being that adding each bat made the mission three score a whole factor bigger. However, after further investigation it was realized that saving weight also required less lift, which in turn created less drag, and therefore a lighter plane that could ultimately fly faster as well. Both flight time and weight are each factored into two of the three missions, while the number of bats is only factored into one of the missions. Therefore, after much thought, debate, and attempts to quantify the flight scoring breakdown, it was agreed that focusing on saving weight and increasing flight speed, rather than holding as many bats as possible would provide the best score.

Design Philosophy

Two primary factors were taken into consideration when creating the design philosophy: the baseline requirements of the aircraft, and the concluded scoring breakdown. As previously mentioned, the plane would potentially have to carry 70oz of softballs, and therefore it should just as easily be able to hold 3 baseball bats (potential 60oz). Nevertheless, the flight with the softballs was not a timed flight, and therefore could be taken with caution, while the flight with the baseball bats was a timed flight. Considering all these factors, it was decided to design a plane to fly at high performance while holding three bats and to fly in a cargo configuration while holding 10 softballs.

Selection Process

In order to select a design, the design philosophy and baseline requirements have to be unified into a single cohesive idea that maximizes score. The aircraft has to be fast and light, yet still carry at least three bats for mission 3 and at most ten balls for mission 2. The speed requirement led to early eliminating a lifting body fuselage or a dual fuselage design, which would create more drag. Instead a low drag fuselage was chosen.

Minimizing the weight dictated that there should be a minimum amount of detachable parts. This requires that the overall aircraft be smaller and also reduces the weight from connection reinforcements; both of these would significantly decrease the overall aircraft weight. After preliminary dimensioning calculations, it was determined that the fuselage could fit into the box as a solid body with a detachable empennage and a two-part wing.

The last consideration in the conceptual design process was the ball and bat placement. Both the balls and the bats need to be placed so that the effect on the center of gravity is minute. For this reason, the center of mass of the payload must be as close to the aircraft center of gravity as possible. Furthermore, the moment of inertia about the rolling axis should be minimized for the ease of turning. Therefore, bats should be attached as close to the fuselage as possible. However, the exact placement of the bats would be determined after further analyses. Moreover, since the bats were up to 30” long, the center of gravity has to be at least 15” behind the propeller. In order to achieve a stable aircraft, the wings should be located aft of the center of gravity which requires them to be more than 15” behind the propeller. Since the sizes of the balls vary only slightly, the mass variance is considered to be of little significance compared to the variance in number of balls. Figure 3.2.1 shows the final conceptual sketch chosen.

3.3. Propulsion

Motor/Propeller

Multiple motor and propeller configurations were considered in order to determine the ideal setup for the required mission parameters. The competition rules allow for multiple motors; however, due to the weight penalty it would incur, only configurations with one or two motors were researched.

Pusher – Mounted in the rear of the main body, the motor and propeller provide a “pushing” force, propelling the aircraft forward. Advantages of this option include reduction in skin fiction drag and reduction of aircraft fuselage length (Raymer). Potential negative issues with this design include ground interference on takeoff and a lower propeller efficiency due to the disturbed air behind the wing.

Tractor – Mounted near the front of the aircraft, the motor and propeller provide a “pulling” force. With this setup, the incoming air to the propeller is undisturbed resulting in a high propeller efficiency. The tractor setup places a large mass in the front of the aircraft, which tends to shorten the overall length of the plane, allowing for a smaller tail area and improved stability. (Raymer)

Dual – Mounted on the wings, two smaller propellers provide the equivalent thrust of a single, larger motor. This setup limits the location of the bat payloads exclusively to the fuselage and introduces motor-out controllability problems which demand and larger tail size. (Raymer) Weight would also increase over a single motor setup.

Ducted Fan – A ducted fan is capable of providing large amounts of thrust. However, ducted fans tend to be inefficient and problematic to manufacture and fabricate.

With score sensitivity relying heavily on weight, the tractor configuration proved its merit by allowing for a smaller tail size and increasing stability in flight. The tractor arrangement also permits a greater range of propeller options when compared to the pusher configuration with consideration to takeoff.

Motor/Battery

Single Motor- Single Battery – Conventional design that relies on a single motor powered by a single battery to provide thrust. Benefits of this design include low weight, less chance for mechanical or electrical failure, and less fuselage volume.

Dual Motor – Single Battery – Mounts a motor on each wing, powered by a single battery. Benefits include high power-to-weight ratio and more of the fuselage devoted to the payload. This is unrealistic due to the probability of high current draw from two motors, damaging the fuse.

Single Motor – Split Battery – Same mechanics and propulsive power as the Single-Single design. However, splitting the battery allows for fine-tuning center of gravity location, allowing the payload weight to vary considerably.

Dual Motor- Dual Battery – Provides very high power and thrust capabilities. Greatly increases weight and thus, structural weight.

The single motor – split battery configuration displayed flexibility and was relatively lightweight when compared to the other design considerations. Once again, weight factored into scoring sensitivity more than any other parameter, ruling out the heavier, dual motor designs. Split battery allowed for fine-tuning of the center of gravity of the plane for the different payload setups.

4. Preliminary Design

4.1 VeloX-1

For the purpose of testing and validating conceptual fuselage and overall plane geometry, a prototype, VeloX-1, was fabricated based off of initial concepts. Created from an assortment of materials and parts from previous competitions, the prototype was built concurrently with the preliminary design phase. This allowed the team to analyze and test design ideas, with the intention of creating a more robust design for the final plane.

The prototype confirmed the vertically stack softball payload configuration, as well as a single wing and tail. It also elucidated potential stability problems discussed in Section 4.8. Performance of the prototype gave rough validations of estimated final performance. However, the final plane will have increased performance due to reduced aerodynamic drag and optimized take-off weight.

4.2 Trade Studies

Local Winds – Wichita, KS

Wind trade studies have been done by past UCSD teams for McConnell, AFB, covering years 2001-2007. Wind direction proved to be highly unpredictable with gust speeds up to 29 mph, (UCSD DBF Report, 2008). A sampling of wind data from April, 2009, correlates the past data with present conditions at this time of year. It presents some even more challenging wind conditions, proving the need for a robust design and stable plane. Gust speeds of 55 mph were recorded in April, 2009, with average gust speeds at 27mph. (Weather Underground, History). Preliminary design requirements and load factors were created using this data.

4.3 Performance

Mission Model

Mission Model Segments

The timed flight missions are broken down into the following segments: take off, climb to cruise altitude, cruise, 180 degree turn, cruise, 360 degree turn, and cruise. Since the timer stops when the aircraft flies pass the finish line, landing time and distance will be irrelevant to our mission score, and thus will not be modeled. The 2nd mission's flight time does not affect mission score. Therefore, modeling mission performance was based purely on ball loading time. The team designed the aircraft to be able to hold 3 bats while flying as fast as possible for the 3rd mission.

Flight Envelope

According to Section 4.2, the past wind data suggested 27 mph of gust or 40 ft/s. The designed aircraft must be able to perform its mission in a 40 ft/s gust during cruise and 20 ft/s gust during maximum velocity. The following is the flight envelope, where the blue outlines the aircraft's structural limit, the red outlines aircraft's limit with gust, and the black star denotes gust-less cruise condition.

Design Parameters and Configurations

Thrust-to-Weight Ratio and Wing Loading Analysis

An initial constraints analysis helps locate the best possible design points for the propulsion set-up. The graphs below show thrust to weight ratios against the wing loading in three different cases: approach or landing, takeoff, and climb or turn conditions. These constraints take into account the loading that the aircraft can sustain and the velocity at which each maneuver is applied.

In order to meet the mission requirements, it is important to choose a point in the top left area of the graph, near the intersection of the constraint lines (Raymer). From the plots above (Figure 4.3.1-2), the desired Mission 1 parameters are: T/W = 0.45 lb/lb and W/S = 1.33 lb/ft2. Mission 2 parameters are based off of Mission 3 parameters due to equivalent takeoff weights and performance of Mission 2 not being a critical consideration. Taking the upper limits for both mission models yields T/W = 0.43 lb/lb and W/S = 2.06 lb/ft2 for both Mission 2 and 3.

Steady Level Turn Analysis

For a steady level turns, a Matlab script was developed to determine the desirable turn velocity, radius, and g loading. In order to shorten the flight time, turning velocity could be increased or turning radius could be decreased. Both of these scenarios would increase g loading leading to an increase in structural weight. Furthermore, turning velocity and turning radius are inversely proportional assuming constant g loading (Figure 4.3.4). The ideal design point occurs where turning velocity is highest and sensitivity to change in velocity or g loading is lowest. This point is illustrated in Figure 4.3.4.

Flight Velocity Analysis

Another Matlab script was used to determine the most desirable cruise velocity by calculating the maximum operational velocity. The desired flight altitude is approximately 100 ft. With a previously determined maximum load of 5 g's, the data cursor on Figure 4.3.4 shows that the maximum safe operable dive velocity would be 100 ft/s. In order to account for human errors and the uncertainties of wind gust velocities, the operational cruise velocity was set at 70% of dive velocity. For payload flights the weight increases, requiring more lift. The increase in lift creates an increase in induced drag, which slows down the aircraft. Analysis revealed that the increasing the weight proportionally decreases the flight velocity for a given thrust.

4.4 Aerodynamics

The Aerodynamics considerations for the aircraft include design and sizing of the fuselage, the main wing, and the tail. In addition, the Aerodynamics team worked on creating Matlab scripts to accurately predict the loading on the wing due to lift forces as well as performing some calculations to predict drag. These calculations would aid the Structures and Propulsions teams, respectively.

Fuselage Design

Once the decision to have a single wing configuration for the overall type of aircraft was made, several fuselage shapes were considered. The optimal design was the one with the least amount of drag but perhaps more importantly, the easiest way to load the softballs quickly. The lifting body design was eliminated early after basic drag calculations proved too large. In addition the fact that it would have to be wide and large in order to house the softball payload, it was determined to be of no overall advantage.

Next, a cylindrical fuselage was considered. For a cylindrical fuselage, one obvious way to load the softballs would be to have a hatch that swung open. However, this would compromise the torsional stiffness of the body. It was decided to "flatten" the circular cross section and mold it more closely around the softball payload. This would not only reduce the weight, but it would reduce drag as well. Also, since the top would be flatter, the team would simply cut holes into the fuselage to load the softballs and a piece of carbon could securely close over them.

The orientation of the flattened was shape was chosen to be short/wide or tall/skinny. Since a short/wide orientation could result in the moving the CG off of the main axis (in the case that an odd number of softballs was selected), it was decided to use the tall/skinny orientation. The softballs would be dropped through holes in the top and would stack on top of one another. A flap of carbon would be folded over the top to smoothen out the surface.

Main Wing Design

Once the overall fuselage shape was determined, the next step was to design the main wing. The main wing design began with choosing the optimal placement of the wing on the fuselage. For stability purposes, the horizontal placement would be dictated by the location of the CG of the aircraft. The best horizontal location for the wing would maintain that the quarter chord (c/4) of the wing was slightly behind the CG. In order to accurately predict the CG location, 3D weighted models of the aircraft and all of its components were made using SolidWorks.

For the vertical placement of the wing, there were three options considered: a high wing, a mid-wing, and a low wing. A high wing would allow better stability and a stronger wing box, but this would most likely interfere with the softball loading system. A mid-wing design would allow for a stable aircraft as well but would complicate the internal payload structure, as it would have to accommodate the wing spar. A low wing design would result in more risky landings as slight aircraft banking could make the wing strike the ground. A low wing also tends to be slightly less stable, although this can be diminished with positive wing dihedral. However, a benefit of the low wing design would be that the landing gear could be mounted directly on the wing box, thus resulting in a stronger landing structure. All of these considerations were taken into account when it was established that the aircraft would have a mid-wing and the internal payload structure would be built around the wing spar.

The only competition-imposed restraint on the wing size was that it must fit in the 2’x2’x4’ box. When the fuselage shape and size were determined, the team decided to halve the wing into two parts, thus allowing for a span of greater than 4 ft. Ultimately, the size of the wing was dependent on the structural loads that the wing spar could take. Taking estimates from previous years’ competition designs, the goal was to have a wing loading of no more than 40 oz/ft2. Initial estimates for the heaviest configuration of the plane (including payload) were roughly 11 lbs. Therefore, a wing area of at least 4.4 ft2 would be necessary.

For the team’s prototype, a simple rectangular wing was selected in order to expedite the process of building the plane and to start flight tests as soon as possible. For the rectangular wing a chord of 9 in. and a span of 72 in. were selected, thus giving a wing area of 4.5 ft2. This would result in a wing loading of approximately 39oz/ft2. These dimensions also resulted in an aspect ratio of 8, well above the team imposed minimum requirement of 6-7. Further discussion on the wing aspect ratio and final geometry are found in Section 5.2.

In choosing the airfoil to be used on the prototype wing, several airfoils were initially considered. The team made a list of airfoils used in previous years’ competitions and by the UCSD teams in particular. Also some classic airfoils such as the NACA 2412 were considered. In deciding which airfoil to use, the team used basic equations like the stall speed and lift equations to estimate the necessary cl for steady and level flight. A cl-max of 1.4 was necessary and during cruising, a cl of at least 0.5 was required. Still, many airfoils fulfilled these requirements and leading to drag considerations for each airfoil. Since the team’s score would directly be influenced by speed and weight, the teams wanted the thinnest airfoil possible to minimize the drag. Therefore, the cd vs. alpha curves were studied, and the Eppler-214 airfoil was ultimately chosen. Figure 4.4.1 shows the cl vs. alpha and cd vs. alpha curves. For cruising speed at zero angle of attack, cl is 0.55 and the cl-max is 1.4.

With the airfoil selection made and the wing sizing determined, the team ordered a foam wing from FlyingFoam.com. The relatively simple design made it easy to lay up and quickly ready for flight tests. The team would continue to work on optimizing the wing size and geometry for the final design, as discussed in Section 5.2.

Empennage Design

The functions of the aircraft's tail are mainly to provide for stability, control, and trim. Trim is achieved through creating a slight lift force with the horizontal stabilizer and elevators which, when coupled with the distance from the main wing, can control the aircraft's pitch. The vertical stabilizer and rudder control the yawing movement of the aircraft and can also help counteract the uneven forces on the fuselage due to the way the air is spun around the aircraft by the propeller.

Several tail configurations such as those in Figure 4.4.2 were considered during the preliminary design phase. Configurations that had more than two basic stabilizers, such as the Twin-tail or the H-tail, were eliminated. They were predicted to add more weight to the aircraft with little benefit to the stability of the plane. The "two-part" tail was pursued for the prototype as it would allow a lot of control from the horizontal stabilizer/elevator.

[Figure 4.4.2. This is the figure of the tail configurations made by Matt A. using Solidworks. If possible and time permitting, please add "twin tail" and "h-tail" to the diagram since I refer to them in the paper. If not, I can re-write that section.]

Accurate tail sizing is needed to ensure the effectiveness of the tail to control the aircraft motion. The horizontal and vertical stabilizer sizes were initially calculated with the tail volume coefficient method outlined by Raymer. Based on the main wing size and the distance between the main wing and tail, a vertical stabilizer of 0.4 ft2 and a horizontal stabilizer of 0.9 ft2 were required. Due to limited supplies the team could not tailor the aspect ratio of the tail to ideal conditions and therefore the aspect ratio of the tail for the prototype was about 1. Flight tests would determine whether sufficient control was achieved using these estimates for tail sizing, and if not, a larger tail would be used. For the final design, aspect ratio of the tail would also be improved upon.

4.5 Structures

Fuselage

The fuselage was chosen to be molded out of carbon fiber and reinforced with bulkheads. It would be two complete halves running from the front to the tail which would then be glued together along the center. There were several reasons for this decision.   First, carbon fiber is considered one of the stiffest, lightest materials for aerospace applications. On top of that, the Fabrication Team was mainly experienced in composite manufacturing and felt more capable of making a high quality composite fuselage than using any other material. Furthermore, being able to mold the fuselage out of carbon fiber allowed for a more contoured shape. This made it possible to only put material where material was needed and hence reduce the weight. It also allowed the Fabrication Team to follow more precise fuselage dimensions. Bulkheads were strategically placed in the fuselage to be able to both hold the balls in place as well as reinforce areas such as where the landing gears were mounted. Moreover, the fact that the fuselage was all carbon fiber made it easily to mount the motor, as the fuselage itself provided a mounting base.

Landing Gear

A tricycle landing gear design was chosen due to its proven historical aircraft usage, which pointed to a reliable design. Considering the wings were mounted high on the aircraft, it was necessary to mount the main wheels to the fuselage. The main wheels needed to be mounted in a place where there was enough load support for a 5g landing. Also, they needed to be placed aft of the wings far enough to prevent tipback of the tail, yet close enough to allow takeoff. In order to conserve weight, it was desired to integrate the landing gear with one of the bulkheads already in place for the balls. This left limited choices, but the bulkhead behind the wing was found to be a sufficient placement of the main wheels.

Wing Connection

A wing joint or connection became necessary due to the length of the wing desired, along with the box constraints. It was decided that the wing should be connected to the fuselage at the wing root for ease of fabrication purposes. The idea of making the wing connections midway down the wing where stresses were lower was also proposed. However, that would have required either making four separate wing pieces or cutting wings in half. It was decided that this would ultimately lead to a less structurally sound and less aerodynamic wing, and therefore the idea was thrown out.

Wing

There were three main ideas considered for the structure and material of the wing: a carbon fiber wrapped foam core, a fiberglass wrapped foam core with a main spar, and a balsa-carbon hybrid wing. After initial estimates of the weights, the balsa-carbon hybrid wing design was predicted to be slightly lighter than the other two alternatives. However, the Fabrication Team had no experience in manufacturing a balsa wing. Moreover, initial structural analysis estimates showed that the fully composite wings had much more strength than their balsa counter-parts. On top of that, using a foam core composite made it possible for a more detailed aerodynamic design of the wing, because the foam could be ordered with the exact specifications desired. Therefore, it was decided that slight weight savings in this case would be traded for structural reliability and aerodynamic precision.

4.6 Propulsion

The propulsion system for this design consisted of a critical items such as a motor, flight pack, and propeller. Non-critical items, not crucially affecting weight or overall system performance, included the 40 amp fuse and the electronic speed controller (ESC). The mission parameters were examined and integrated into design parameters which provided the guidelines for designing the propulsion setup. In order to cut down assembly time in competition, the decision was made to use one motor setup for all missions with the possibility of different battery packs tailored to mission specific need. However, the total weight of the heaviest configuration normalizes each mission score. This directed the search for the lightest and most powerful propulsion setup capable of powering Mission 3, predicted to be the most taxing of the missions.

Past UCSD DBF teams thoroughly tested different types of motors and battery cells. The decision was made early on to focus the battery search on NiMH technology, which can provide higher current loads with less voltage drop than NiCAD technology. No major changes had been made in terms of NiMH chemistry, making the data reasonably up-to-date. Using this data, this year's team used both optimization software and testing to further analyze and validate system performance. MotoCalc, a propulsion optimization code with a large database of user contributed data on a large selection of motors, batteries, and propellers, provided rough estimations on reasonable propulsion setups. This iterative process determined propeller sizing, motor selection, and battery configuration while using a rough model of the plane. Estimations of current draw, voltage drop, flight speeds, and power loss give a reasonable starting point for the optimization of the different missions. The predictions of performance parameters including thrust and pitch-speed at various flight-speeds provided preliminary flight performance validation.

After narrowing down to a few different selections, initial testing was achieved through use of a static motor bench. Combined with data acquisition equipment provided by a sponsor, EagleTree Systems, performance data was gathered. The EagleTree collected real-time data on current draw, voltage, motor and propeller RPMs, while the test bench gave a measure of the static thrust of the system. Further evaluation through prototype flight testing related airspeed data to voltage drop and current draw.

Motor and Battery Trade Study:

Inrunner brushless motors have a proven track record of providing excellent power-to-weight ratios. Generally, lower kV (Rpm/Volt) generates less heat due to higher efficiencies. Motors coupled with a gearbox spin propellers at optimum RPMs. Team sponsor, Neu Motors, has always provided excellent products which this team has used to provide propulsion to the plane. They have a wide selection of high quality brushless motors where this search focused.

Battery selection played a critical role in determination of flight speed and weight—both crucial to the overall flight score. However, total power requirements for the different missions depended on the number of cells and maximum current provided by the selected cells. Preliminary optimization pointed to a 1500 mAh, 2/3A sized cell and a 2000 mAh, 4/5A size cell manufactured by Elite, a major producer of high quality NiMH cells, as candidates for a propulsion pack. Energy densities computed as 1.85 Ah/oz for the 1500 mAh and 1.72 Ah/oz for the 2000 mAh.

Static motor testing showed that the 2000 mAh cell provided a higher continuous current while maintaining better voltage under load than the 1500 mAh cell. For the highest loaded case with the 16x12 propeller, current draw for the 2000 mAh battery was 37.3 amps, while the current draw for the 1500 mAh battery was 36.3 amps. Overall thrust capabilities benefited from the 2000 mAh cells' ability to maintain higher voltages under load. A four-cell lithium polymer battery was used as a control in testing.

Table 4.5.1 displays the predicted power-to-weight ratio, pitch-speeds, and other data for the various motors considered. Brushless motors have an optimum range between 30000-45000 RPM where the highest efficiencies can occur. All motors considered would run in this range at flight speeds and flight loadings. Propeller RPMs are reduced using a gearbox. Efficiencies were quite similar for all motors, pointing to choosing the best power-to-weight ratio and fastest pitch speed combination. All values were estimated using a 15x10 propeller and 16 cell 2000 mAh battery. The optimization software did not account for battery current limitations. This was validated through historical data and testing.

Table 4.5.1: Highlighted motor eventually chosen as ideal component for final plane. It exhibited the highest power-to-weight ratio while maintaining a high pitch speed

4.7 Stability and Control

By emulating commonly used designs, an engineering team can produce good first round estimates for structural, aerodynamic, performance, propulsion, and stability parameters. Since the first estimates can be historically approximated, there are fewer design iterations required to produce a fully developed aircraft. Being limited in both time and money, the VeloX-1 prototype aircraft was designed using this philosophy.

Without knowledge of the static margin or center of gravity location, it was decided that a large elevator surface would be necessary to accommodate the relatively unknown tail forces. It is for this reason that a stabilator design was chosen for the VeloX-1. The horizontal and vertical stabilizers were sized using standard tail volume coefficients for transport aircraft. The rudder size was arbitrarily chosen to be 1/3 the chord of the vertical stabilizer, and the full span in length. These tail sizing parameters are not a far deviation from other similar aircraft, but they proved to be ineffective at stabilizing the VeloX-1.

In the interest of time, production of the prototype proceeded without proper understanding of dynamical properties. The plane was stabilized by adjusting the center of gravity after fabrication.

4.8 VeloX-1 Crash Analysis

The tragic crash of VeloX-1 occurred on February 5th at 12:30 pm. The flight lasted approximately 5 minutes and ended in a 150-foot plummet into a cinderblock fire pit. The prototype was destroyed but the information gained from this flight assured the successful development of VeloX-2.

From the moment the aircraft took off it was apparent that the pilot, despite his uncommon skill, was flying the aircraft at the limit of stability. After an extensive study of video and photographs taken during the flight, a collection of first hand accounts, and sifting through every piece of wreckage it was determined that the stability issue was related to but not directly the cause of the final crash.

The stability issue presented itself in both a lateral pitching mode and a lack of yaw control. Figure 5.8.1 shows one period of oscillation along the pitching mode. The cause of this instability was determined to be a zero or slightly negative static margin. This was caused by an improper placement of the flight battery pack prior to takeoff. In addition the Center of Gravity (CG) of the aircraft was approximately 2 inches above the wing spar, causing an inverted pendulum effect (as described by figure 5.8.2). When combined with the large forces generated by the stabilitor tail, the poor CG placement before takeoff made steady level flight impossible for the pilot.

In addition to the CG instability, the drive motor and propeller where located 14 inches ahead of the wing and 3 inches above the CG. This caused a pitching torque of nearly 2 ft*lbs when the motor was at 100% thrust. This caused the aircraft to pitch down whenever power was applied, making it even harder to control.

The aircraft also experienced a lack of yaw control during the flight. This is primarily attributed to the small vertical stabilized and small rudder. Since the fuselage was tall and skinny, it provided some damping to yaw oscillations but meant that the VeloX-1 required a much larger rudder in order to affect any yaw influence. The tail on the VeloX-1 had been designed using traditional tail sizing methods without much consideration of the unique fuselage shape. Without yaw control the pilot was unable to initiate any turns in standard fashion. Normally a rudder is required to initiate a turn because the act of banking an aircraft causes a yawing moment in the direction of the raised wing due to increased lift instigating increased induced drag. Without yaw control the pilot was able to perform crude maneuvering using a lateral nose over technique, in which the aircraft is banked until decreased lift causes the nose to pitch down. When the aircraft is pulled out of this maneuver, an uncoordinated but effected turn is initiated. Images of one such maneuver can be seen in figure 5.8.3.

As stated previously, the review of the crash data does not indicate that the instabilities noticed during the flight were directly responsible for the crash. Since both the limited video of the crash and all eyewitness reports show the aircraft to suddenly lose lift and falling to the ground in a fully developed bi-axial spin, it is assumed that there was some structural, electrical or mechanical failure.

After the crash, inspection revealed delaminating of the fiberglass on both the right wing and tail surfaces. Since it cannot be determined if this occurred before or after the crash, it is neither excluded nor implicated as the crash result, but remains as a cause for concern on VeloX-2. The inspection of the electronics found them to be working in proper condition, and the distance from transmitter to aircraft was measured to be .45 miles at the time of crash. This excludes the electronics as cause for the crash because they were not damaged and well within line-of-sight range for the microwave band transmitter. Finally the stabilator servo mechanism was inspected and found to be slightly dislodged and no longer capable of produced sufficient torque to actuate the stabilator under normal loading.

It is for these reasons that the crash of the VeloX-1 is primarily attributed to structural failure of the stabilator servo mechanism. The pitch and yaw instabilities and moderately unfavorable weather conditions were contributing factors to the unusually large loads on the servo.

5 Detail Design

5.1 VeloX-2

Immediately following the crash of VeloX-1, planning began for the VeloX-2. Just as VeloX-1 was fabricated as a prototype, the VeloX-2 has been designed as the competition aircraft. This required some major changes from the prototype to be implemented in order to assure performance.

First off the wing spar was moved up to 2 inches from the top of the fuselage and the motor was moved down to 2.5 inches from the top of the fuselage. By placing the center of lift above the center of gravity, the aircraft will act as a stable normal pendulum instead of the unstable inverted pendulum. With the motor mounted below the center of lift and above the cg while still being close to both, it will provide a minimal pitching moment during full throttle operation.

Next the stabilator tail was redesigned as a conventional tail with larger horizontal and vertical stabilizer areas. In addition the rudder size was increased to twice that of a normal aircraft of this type. This is necessary because the large side profile of the fuselage resists yawing against the relative wind. The larger stabilizer areas will provide greater pitch and yaw damping for the aircraft.

In addition, the aircraft design was carefully rebalanced to assure a CG location ahead of the center of pressure to insure a positive static margin. The redesign allows for the static margin to be controlled to within 5% under any loading factor.

Lastly, the aircraft was designed to utilize advanced materials not available for the prototype. Items such as a carbon fiber fuselage and tapered wings with twist were incorporated to increase performance and reduce weight of the aircraft.

With lessons learned from VeloX-1, the VeloX-2 promises to be an efficient high-performance aircraft with a variety of innovations and great potential for the competition.

5.2 Aerodynamics

Fuselage Design

The main concern with the fuselage body was its relatively large side area. This was a concern because it is known that in Wichita, Kansas in April there can be high gust winds (as described in Section 4.2) . However, the UCSD team performed flight tests on a rather windy day, with wind speeds estimated at ------- knots. Also, according to the pilot, the aircraft did not seem to be pushed around by the wind. Instead, it was hard to control due to the insufficient tail size. Therefore, the team decided to keep the same fuselage shape as used on the prototype.

Main Wing Design

The rectangular wing used on the prototype provided sufficient lift throughout the flight. In addition, the take-off distance was measured as roughly 70 ft., well under the 100 ft. limit. However, the Aerodynamics team wanted to make improvements to the wing geometry to optimize its performance.

In selecting the specific geometry for the final wing, several characteristics were considered including aspect ratio, taper, and twist. For the final design, an aspect ratio of at least 6-7 was desired to reduce the effects of induced drag. The larger the aspect ratio of the wing, the less the wing is negatively affected by wingtip vortices. Wing sweep was not a priority due to the relatively low velocities at which the aircraft would perform. Wing taper, however, would have huge benefits. Tapering the wing would ensure that more lift was produced near the root of the wing, and less at the tip. This is beneficial for two purposes: first, it would reduce the amount of induced drag experienced due to wingtip vortices, and secondly, it would decrease the bending moment due to the lift forces at the root of the wing. Raymer suggests a taper ratio, λ, of 0.45. However, upon observing planes from previous years' competitions, it was decided that a taper ratio of about 0.6-0.7 should be used. Wing twist would also be implemented to ensure that stall (if it should occur) would occur at the root instead of the tip of the wing. This is desirable since tip stall is hard to recover from, and wings that stall at the root generally give the pilot enough time to correct the flight path. Raymer suggests twist angles of 0-5 degrees.

To decide upon the final geometry of the wing, the UCSD Aerodynamics team created its own Matlab program to calculate the lift forces that different geometries would produce. In order to produce more lift near the root, the team selected a chord length of 12 in. at the root. The chord at the tip would be reduced to 8 in., thus resulting in a taper ratio of 0.67. This would also give a mean chord length of 10 in. Since the prototype wing provided sufficient lift, the same wing area was maintained. Therefore the span could be reduced slightly to 68 in. Finally, since the team would order the final wing again, it was decided to utilize a wing twist of 3 deg with 2 deg angle of attack at the root and -1 deg angle of attack at the tip.

Wingtips were also a consideration for the Aerodynamics team, as the induced drag on the wing could drastically be lowered with the proper selection and implementation of wingtips. Various wingtip designs that were considered are shown in Figure 4.X. Ultimately, the team decided to forego adding extra weight to the wingtips and opted for a sharp edge wingtip such as the Hoerner tip. This design inhibits the amount of high pressure air below the wing from to spill over to the top of the wing while also being very easy to fabricate.

[Figure 5.2.1 Wingtip picture from Raymer]

The process of finalizing the airfoil selection was again the most extensive part of the wing design. A myriad of airfoils were initially considered as the team began by compiling a list of airfoils used by UCSD teams in previous years' competitions. In addition to those, a few other standard airfoils were considered such as the NACA 2412.

It was fairly obvious from the start that several airfoils and wing geometries could achieve take-off within the specified length of 100 ft. Therefore, the team decided to prioritize high speed and low weight, because these factors had a direct impact on the competition score. First, the cD as a function of angle of attack graphs were studied (Figure 4.X). By observing these graphs, the team quickly decided to eliminate those airfoils with a cD greater than 0.015, such as the E591.

Next, the drag polar graphs were studied, such as in Figure 4.X below. As a preliminary estimate, the team determined the wing would be set anywhere from -2 to +2 degrees angle of attack. The lift coefficient was determined at these values of alpha and the drag polar graphs were used to see which airfoils had a higher cD at these specific cL values. From this, the team further eliminated the SD7003 and the NACA 2412.

The two final airfoils considered were the E214 and the E392. The decision was ultimately made to use the E214 due to its higher cL vs. alpha curve. More lift could be generated at low angles of attack by the E214 than the E392.

Empennage Design

During preliminary flight testing it became apparent that the vertical stabilizer was improperly sized, greatly hindering control of the aircraft. In an effort to improve performance, the area of the vertical stabilizer was increased roughly by a factor of two. The larger stabilizer, and so larger rudder, would correct the poor performance observed in the flight test. The horizontal stabilizer area was increased but by a factor of about 1.5. Based on historical data found in Raymer's book, the vertical stabilizer generally has an aspect ratio of about 1.3 to 2 and the vertical stabilizer has an aspect ratio of about 3 to 5.

The final tail design would also be changed to the conventional tail, instead of the "two-part" tail that was tested on the preliminary aircraft.

5.3 Structures

Wings and Wing Connection

The wing has a foam core wrapped with fiberglass. Down the quarter chord is a main spar. The main spar is composed of two unidirectional carbon fiber spar caps and a 1/8" balsa shear web. The spar caps are designed to take a maximum tensile and compressive stress of 15.4 ksi at the root from a 5g turn, while the spar's shear web is designed to take a maximum shear stress of 3.2 ksi. The fiberglass laminate will help reduce both the shear stress and tensile stress in the spar for an additional factor of safety.

The wing also has two connections at the wing root to connect the wing to the fuselage. The main spar on each wing extends past the wing root. This extrusion is then inserted into a hollow carbon fiber rectangular tube which is molded to the interior of the fuselage. Two pins run through the hollow tube and the spar to ensure that the wing does not slip out during flight. The second connection is fairly similar. A carbon fiber tubing runs through the bulkhead near the trailing edge of the wing. This tubing is adhered to the bulkhead. A hole is cut out of the wing at this location where the tubing can then be inserted into the wing.

The main spar connection is also designed to take the maximum stress that each spar is expected to take. The two connections together are made to take a twisting moment of 7.75 lb-ft from the wing's aerodynamic moment.

Landing Gear

The main landing gear is composed of a carbon fiber weave to create a solid spring landing gear. The front landing gear consists of wheel and spring to absorb shock loads. The nose wheel is actuated by a servo motor for taxi steering. The front gear is mounted to the front bulkhead, while the main gear is mounted to the bulkhead just behind the wing's quarter chord. The main landing gear is designed to take a 5g landing load of 55.5 lb.

Fuselage

The fuselage is composed of one layer of carbon fiber weave. Five holes are cut into the top of the fuselage for the balls to fit through. Seven bulkheads are then evenly spaced along the center to create slots in between which the balls will be placed—except for the two bulkheads which are on each side of the main spar wing connection. The bulkheads are all 1/8" balsa sheets, except for the bulkheads connected to the landing gear. The landing gear bulkheads are 1/8" balsa sandwiched in carbon fiber for landing support. Additional layers of carbon are also added around the landing gear bulkheads to help the fuselage hold the landing loads. The center of each bulkhead has plastic flaps which hold the bottom balls in place in case no second ball is loaded above them. Plastic flaps also cover the top of the fuselage to keep the balls in place during flight.

On the outside of the fuselage there are pins where each bat's CG is to be placed. There is one pin on the top of the fuselage, and two pins on each side of the fuselage for a total of five pins. There are C-clips with a 2" diameter on each side of each pin to restrain the balls from moving. The rationale for having a mounting system for 5 bats is that although the plane is designed to fly only with 3 bats, it is possible that the aircraft may be overdesigned, in which case more bats can be added if the design proves capable of handling the extra weight and drag.

The motor is mounted to the fuselage via two screws drilled through the front of the fuselage. Additional layers of carbon fiber have been placed around these screws to make sure the fuselage can withstand a 6.25 lb-in. torque from the motor.

The carbon tail boom has a 2.5" diameter, as that proved sufficient for withstanding a 3.5 lb-ft moment from the tail.

5.4 Propulsions

General Propulsion

Based on predictions and static motor testing, the team decided on the Neu 1506/2y with a 6.7 gearbox as the optimum motor choice. When compared to the various Neu 1509 motors Table 4.5.1, a slightly larger motor capable of providing more power, this motor performed similar or better in most significant categories while weighing a few ounces less. Placing the motor along the center line of the wing to reduce any thrust induced pitching moment effect. This system achieved admirable efficiency and thrust numbers while minimizing weight. Batteries configurations and placements in the plane played a huge role in the determination of the center of gravity. The team decided on using batteries in a stick configuration (see Figure 5.7.1),which allowed for the placements in any payload compartment along the fuselage. Fine-tuning the center of gravity proved critical for obtaining optimum flight conditions.

Mission 1

Mission 1 called for the lowest achievable 2-lap flight time. The heaviest total weight , W_team, was determined from any mission attempt, not necessarily this mission. Therefore, system weight, including the battery weight, for Mission 3 determined the W_team for all missions, since the takeoff weight for Mission 3 calculated as the heaviest. This led to the optimization of flight time and thus cruise velocity for Mission 1. The lightest takeoff weight occurs for Mission 1, making required static thrust less of a priority then pitch-speed. An APC electric propeller with a high pitch, 14x12, was selected on this basis. Maximum flight speeds of 70 mph derive from this setup. Further flight testing will be conducted to verify this decision. The battery pack chosen for this mission also reflected the same methodology. A 16 cell 2000 mAh Elite flight pack optimized the cruise velocity while not increasing the overall total weight, W_team. Since drag increases as a function of velocity squared, increasing the number of cells and thus, the weight of the plane, has limited benefit. Therefore, using the 16 cell battery pack optimized the maximum velocity of 70mph, without increasing the W_team derived from Mission 3. Predicted flight times were not an issue with the higher capacity of the 2000 mAh cell. The predicted max current draw of 33 amps gave a good margin of safety between flight loads and the maximum allowed current by the fuse.

Mission 2:

In order to account for the possibility of a varying takeoff weight, propulsion optimization for Mission 2 assumed the maximum number of softball, and thus, the maximum payload weight. Flight time and flight speeds are not an issue in this mission. A propeller with a 15x10 diameter and pitch gave the necessary thrust for takeoff within 100 feet. An adequate margin of safety was given between required thrust and generated thrust to account for any unexpected winds or other unpredicted.

The same battery pack for Mission 1 powers the Mission 2 flights. This did not increase the overall weight factor and easily made the three lap mission profile at cruise speeds of 45 mph.

Mission 3

Team philosophy predicted the optimum number of bats as four. Maximum payload weight for this mission equaled that of Mission 2. However, drag increased substantially due to the blunt objects mounted on the aircraft. Takeoff distance increased for this mission, requiring slightly more thrust than needed in Mission 2 Figure 5.4.2 . A 16x12 propeller gives the plane adequate thrust for takeoff and a much higher maximum velocity of 65 mph.

As mentioned previously, the propulsion design centered on Mission 3. The highest flight loads and battery loads result from this Mission, therefore the battery optimization determined the battery for all three missions. A 16 cell 2000 mAh Elite battery pack once again provided optimum power.

Testing of the prototype indicated a maximum velocity of 60mph and takeoff distance well below the 100 ft limit. From this data, battery cell count was reduced to 16 cells further testing of the actual plane will allow better propeller optimization for varying wind conditions and unexpected loading. 

5.5 Performance

Aircraft Performance

The following summarizes the desired aircraft performance for ferry and payload flights. The aircraft is designed to be able to take -2 to 5 g loads with a minimum height of 40 ft for pulling out from a negative 60 degree dive at maximum velocity. As mentioned previously, Mission 2 and 3 can be viewed congruently.

Performance Parameter	Mission 1 (no payload)	Mission 2 & 3 (10 balls or 3 bats)
CLmax	1.3	1.3
Oswald Efficiency Factor	0.8	0.8
CD0	0.05	0.07
(L/D)max	8.68	8.68
Takeoff Weight (lbs)	8	11.75
W/S (lb/ft^2)	1.33	2.06
T/W (lb/lb)	0.45	0.43
Takeoff Distance (ft)	30	80
Steady Turn Radius (ft)	15.85	13.45
Cruise Speed (ft/s)	70	46.2
Stall Speed (ft/s)	29.38	35.6

Steady Turn Speed (ft/s)	50	33
Maximum Speed (ft/s)	100	66
Climb Rate (ft/s)	50	33

Mission Performances

Mission 1: Ferry Flight

	Distance (ft)	Velocity (ft/s)	Time (s)
Take Off	30	0-50	6
Climb	60	50	1.2
Cruise	410	70	5.9
180 Turn	49.8	50	1.0
Cruise	1000	70	14.3
540 turn	149.4	50	3.0
Cruise	500	70	7.1
Total 1st lap time (s)	**38.5**		
Cruise	500	70	7.1
180 Turn	49.8	50	1.0
Cruise	1000	70	14.3
540 turn	149.4	50	3.0
Cruise	500	70	7.1
Total 2nd lap time	32.6		
Total mission 1 time (s)	**71.0**		

Mission 2: Internal Payload Flight

The scoring factor for Mission 2 is described in section 3.1.2. With the aircraft's internal column design, the balls could be loaded into the columns at up to 2 balls at a time. From the ball configurations shown in Figure 4.3.5, the vertical column loading method is expected to be faster than single ball loading by at least 1.4 times. The expected loading time parameter T2 would be at least 0.9.

Mission 3: External Payload Flight

	Distance (ft)	Velocity (ft/s)	Time (s)
Take Off	80	0-50	8
Climb	60	33	1.8
Cruise	360	46	7.8
180 Turn	42.3	33	1.3
Cruise	1000	46	21.7
540 turn	126.8	33	3.8
Cruise	500	46	10.9
Total 1st lap time (s)	**55.4**		
Cruise	500	46	10.9
180 Turn	42.3	33	1.3
Cruise	1000	46	21.7
540 turn	126.8	33	3.8
Cruise	500	46	10.9
Total 2nd lap time (s)	**48.6**		
Total 3rd lap time (s)	**48.6**		
Total mission 3 time	**152.8**		

5.6 Stability and Flight Controls

Static Stability

The static stability of an aircraft requires the balance of wing lift, tail lift, weight, thrust and drag. Figure 5.6.1 shows schematically how these forces are applied on the VeloX-2 design.

Without wind tunnel data it is very difficult to measure the drag force of the aircraft. In addition, component weights of the finished design are often different from those estimated in design models. It is for these reasons that the aircraft center of gravity will need to be positioned after final fabrication using weighting Included in the design is the ability to position the flight battery packs in various locations around the aircraft to distribute their significant weight with different moment arms. This allows rough tailoring of the CG location after fabrication.

For fine positioning of the CG location, the payload configuration is taken into consideration. The bats are designed to be mounted symmetrically on the fuselage at the CG location, and with the same loading as the 10-ball case. This allows us to simplify our CG calculations to 5 different loading conditions; 0-balls/ unloaded, 6-balls, 7-balls, 8-balls, 9-balls, and 10-balls/3-bats. Figure 5.6.2 shows the position of the balls (nose being on the left, tail on the right) for loadings of 1-10 balls in order to keep the static margin in the desired 5-10% range. The loadings for 1-5 balls is included as a likely testing payload, though it will not be seen in the competition.

Table 5.6.1 shows the cargo loading, CG location, Static margin, and tail loading for each configuration shown in Figure5.6.2. The static margin offset is calculated based on the distance from the CG to the propeller. These calculations were assuming the batteries where split into two symmetric packs, one in the nose and the other in the tail. Notice that the static margins are all acceptable except the unloaded condition. This demonstrates the need for modular battery packs that can be positioned in a variety of locations to help balance the aircraft. For the unloaded condition, the rear battery pack will be moved into one of the cargo slots to correct the static margin.

Cargo		Center of Gravity		Static Margin		Tail force (lbs)	
Load (Softballs	Weight (lbs)	X (in)	Y (in)	SM offset	SM	0% Thrust	100% Thrust

)							
0	8.00	13.99	4.6	-3.70%	-0.40%	0.055	-0.030
1	8.44	13.38	4.48	-3.58%	5.82%	0.165	0.081
2	8.88	13.29	4.37	-3.56%	6.74%	0.191	0.106
3	9.32	13.2	4.45	-3.54%	7.66%	0.217	0.133
4	9.76	13.38	4.18	-3.58%	5.82%	0.191	0.106
5	10.20	13.34	4.27	-3.57%	6.23%	0.208	0.124
6	10.64	13.26	4.18	-3.55%	7.05%	0.235	0.150
7	11.08	13.23	4.26	-3.55%	7.35%	0.252	0.167
8	11.52	13.34	4.33	-3.57%	6.23%	0.235	0.151
9	11.96	13.31	4.4	-3.56%	6.54%	0.252	0.167
10	12.40	13.25	4.46	-3.55%	7.15%	0.276	0.192

Table 5.6.1: Center of gravity, static margin, and tail force for different cargo loadings of the aircraft. Center of gravity is measured from the bottom of the first bulkhead.

Figure 5.6.3 shows the static margin plotted versus the cargo loading. It shows that as the weight of the cargo increases, the static margin becomes more stable. This is because the cargo weight is a significant fraction of the aircraft weight, and as it increases it slowly moves the aircraft CG to the cargo CG.

Figure 5.6.4 is a plot of the Required tail force versus the cargo loading. Note that a positive force is in the downward direction by standard definition. The force required by the tail is limited to a range of -0.1lbs to 0.3lbs. This range is well within a respectable limit for a horizontal tail surface and small enough to allow for steady level flight with light trimming. The thrust effects the tail force because it is offset from the cg and hence generates a moment.

Control Surfaces

The ailerons, rudder, and elevator sizes for the VeloX-2 were chosen from historical data for aircraft of this type. Using data from Raymer, the Ailerons where chosen to be 40% of the wing span, and 25% of the wing chord. Similarly the rudder was chosen to be 40% of the vertical stabilizer chord, and the elevators chosen to be 45% of the horizontal stabilizer chord. Figure 5.6.5 shows the Raymer charts used to size the control surfaces of the VeloX-2.

It should be noted that the aircraft was simulated as a single engine General Aviation aircraft despite the fact that it carries a large cargo and travels at high speeds. The reason for this decision is manifold; the aircraft is experiencing prop wash similar to a single engine GA aircraft, the body layout is similar to a GA aircraft, and the crash of the VeloX-1 would suggest a large tail is beneficial to this design. In addition, it is useful to consider the similarities between the VeloX-2 and the Cessna Caravan in terms of thrust-to-weight ratio, wing aspect ratio, center of lift, center of gravity, engine mounting, and general shape. These factors justify the parameters chosen for control surface sizing.

Dynamic Stability

The Dynamic stability of the aircraft was analyzed in two different fashions. First the moment of inertia tensor was calculated for different loading conditions. The values for unloaded, 6 balls, and 10 balls are listed in table 5.6.2.

Table 5.6.2: Moment of Inertia tensors for different loading configurations.

There is very little change in the tensor depending on loading because the center of gravity of the cargo is located very near to the center of gravity of the aircraft. The Moment tensor is a dominant diagonal matrix which allows the dynamic stability of the X,Y, and Z axes to be treated independently. This also tells us that the standard front right down coordinate system is closely aligned with the principle axes of the aircraft.

The second dynamical analysis performed for the VeloX-2 was the estimation of the aircraft stability coefficients. This was accomplished using a Vortex Lattice Method (or VLM) written by Jacob Kay of the Aerospace and Ocean Engineering Department of Virginia Tech in 1992. The code is called JKayVLM and utilizes a trapezoidal approximation for aircraft shapes including fuselage, wings, horizontal stabilizers, vertical stabilizers and control surfaces. The output parameters are listed in table 5.6.3.

Stability Coefficients					

General		Lateral		Ailerons	
CL0	0.64438	Cy-beta	-1.12185	CL-delta [3 2]	1.79169
Cm0	-1.99669	Cn-beta	0.26389	Cm-delta [3 2]	0.97264
Cdi(at alpha =5)	0.08672	Cl-beta	-0.05517	Cl-delta [3 2]	-0.18755
Cm/CL	-0.07244	Cy-r	1.09144	Cn-delta [3 2]	0.00413
Cdi/CL^2	0.0565	Cn-r	-0.44698	Elevators	
CL (at alpha =5)	1.52156	Cl-r	0.06492	CL-delta [5 2]	1.32035
Cdi(at alpha =5)	0.1308	Cl-p	-1.10171	Cm-delta [5 2]	-4.41261
		Cn-p	-0.7979	Cl-delta [5 2]	-0.10423
		Longitudinal		Cn-delta [5 2]	-0.13337
		CL-q	20.68985	Rudder	
		CM-q	-43.2414	Cy-delta [3 2]	0.45934
		CL-alpha	10.05174	Cl-delta [3 2]	0.03595
		Cm-alpha	-0.72818	Cn-delta [3 2]	-0.21684
		CL-epsil *	0.49614		

Table 5.6.3: Stability Coefficients calculated using JKayVLM.
*Lift curve slope of the tail from downwash from the wing.

The term Cm/CL in table 5.6.3 is equal to the negative of the static margin by definition.

The KJayVLM code calculates a static margin of 7.24% which is very well centered in the 5-10% range defined for stability, and very close to the calculated static margins for different cargo loadings.

Some of the stability coefficients are compared to a reference aircraft (taken from Roskam) of similar span, chord, and CG location in Table 5.6.4. This comparison indicates that the VeloX-2 will perform in a similar manner to the reference aircraft with the exception of Cl-p and CM-q, which are larger because the VeloX-2 is high wing (Aircraft F is low wing).

Reference Comparison			
VeloX-2		Airplane F	
Lateral		Lateral	
Cy-beta	-1.12185	Cy-beta	-0.883
Cn-beta	0.26389	Cn-beta	0.1739
Cl-beta	-0.05517	Cl-beta	-0.1381
Cy-r	1.09144	Cy-r	0.448
Cn-r	-0.44698	Cn-r	-0.2
Cl-r	0.06492	Cl-r	0.1166
Cl-p	-1.10171	Cl-p	-0.566
Cn-p	-0.7979	Cn-p	-0.0501
Longitudinal		Longitudinal	
CL-q	20.68985	CL-q	7.5
CM-q	-43.2414	CM-q	-22.4
CL-alpha	10.05174	CL-alpha	5.55
Cm-alpha	-0.72818	Cm-alpha	-1.18
CL-epsilon	0.49614	CL-epsilon	0.58

Table 5.6.3: Comparison of select stability coefficients with a reference aircraft taken from Roskam.

In addition, when the coefficients are checked for correct sign (+/-) using the conventions set up in Roskam, all values imply stability. Table 5.6.4 shows the values and their sign convention required for dynamic disturbances to have a negative feedback. These sign convention check does not imply a well damped system, but does show that perturbations will decrease over time instead of growing.

Stability Check			
Cy-beta	-1.12185	< 0	PASS
CL-alpha	10.05174	> 0	PASS
Cl-beta	-0.05517	< 0	PASS
Cl-p	-1.10171	< 0	PASS
Cm-alpha	-0.72818	< 0	PASS
CM-q	-43.2414	< 0	PASS
Cn-beta	0.26389	> 0	PASS
Cn-r	-0.44698	< 0	PASS

Table 5.6.4: Stability check by sign convention taken from Roskam.

When all stability factors of the VeloX-2 design are taken into consideration, the aircraft presents a statically and dynamically stable design. In addition the control surfaces are of reasonable size to prevent surface stalling and still provide enough force to maneuver the aircraft. Together these conclusions confirm the final check before the VeloX-2 moves into production.

5.7 Secondary Components:

The team chose a Castle Creations Phoenix 80 electronic speed controller, which provided up to 80 amps of continuous current and up to 25.2 volts, giving a large factor of safety, thus not constraining performance in any way. This controller covered all possibilities of battery and motor configurations considered in the design, while having a relatively low weight of 2.1 oz.

A JR X9503 2.4 ghz radio provided control for the plane. The 2.4 ghz radio effectively prevents "glitching" by switching frequencies many times per second. This also prevents interference from other transmitters used at the competition. Transmitter weight factors into the final W_team, affecting all missions. However, data on transmitter weight proved hard to come by. From a general sampling of radio brands, it was determined that the average weight per transmitter did not greatly vary. Therefore, the decision to use pre-existing team hardware resulted from trust in the equipment and budgetary constraints.

A 400 mAh NiMh receiver pack provided power to the servos and radio receiver. In order to minimize the weight of the receiver pack, experimentation and calculations determined a maximum current draw of 1.5amps. When coupled with the expected flight times, a maximum energy usage of 200 mAh is predicted. 400 mAh cells give a factor of safety of 2, allowing for any discrepancies from the predictions.

6. Manufacturing Plan and Processes

6.1 Fuselage

To minimize weight of the fuselage, carbon fiber was selected as the main material. A process using a custom mold and vacuum bagging was selected. This process allows for minimum weight and optimal strength due to improved matrix distribution during curing. For the bulkheads a carbon/balsa sandwich method was chosen for a higher strength than pure balsa with minimal weight addition. These high strength bulkheads allow us to use less material in the shell of the fuselage and easily distribute high point loads where necessary.

6.2 Wings/Empennage

Two methods of wing fabrication were chosen in order to flight test and analyze the strengths and weaknesses of both. The first method chosen is a foam core with a fiberglass skin. This method produces a very robust wing and has in the past been very reliable for UCSD’s DBF teams. The second method involves making most internal components out of balsa with a monocoque skin. This method creates a much lighter wing which would help optimize flight scores. Our fabrication team has little experience with this type of fabrication but chose to utilize the design for analysis on performance.

6.3 Landing Gear

To minimize weight a composite layup method was utilized in the fabrication of the rear landing gear. A custom mold was made from foam and the gear consists of purely carbon fiber.

6.4 Box

Rectangular carbon tubes were used as the frame of the box and were connected using a carbon fiber layup. Cross braces were installed for additional support. Internal mounting points were added to secure the plane and flight gear.

7. Testing Plan

7.1 Propulsions

Static motor testing equipments included the static thrust bench, propulsion system, and EagleTree system. The thrust bench had a motor mount attached to a 16 in. vertical lever arm, which was connected to a 14 in. horizontal lever arm. This system translated motor thrust into a proportional force measured by a scale. Readings from the scale were multiplied by a scalar, giving the thrust in terms of ounces.

The EagleTree data acquisition system recorded current flow, voltage, power output, and propeller rpms. Data were then analyzed on a computer.

Figure 7.1.1: Static motor bench with Neu 1506/2y and 14x12 APC propeller

7.2 Stability and Controls

In order to test for stability, two methods will be used. The first method is a tried and true process used by test pilots since World War II called t he Cooper-Harper rating scale. The scale is shown in figure 7.2.1 and provides a quantitative communication between pilots and engineers.

The second testing method involves the installation of an IMU device (Inertial Measurement Unit) in order to monitor the 3-axial motion, 3-axial attitude, and control surface deflections of the aircraft. This will provide enough data to formulate a state space transfer function and Nyquist plot of the aircraft. With this information the stability of the aircraft can be determined with substantial accuracy.

7.3 Aerodynamics

To reduce the wing tip vortices, which would increase the speed performance; different wing tip designs are being considered. With the recent acquisition of a 3D Printer (1200es Dimension Printer) by the UCSD Mechanical and Aerospace Engineering Department, scaled models of various wing tip designs will be made and tested in the UCSD wind tunnel. Drag force for each wing tip configuration will be measured and used to decide the final wing tip design.

7.4 Structures

There are plans to test various parts of the fuselage along with the main spar. Spars that are specifically built for testing will be loaded to failure in order to ensure the predicted strength. Composite sheets will also be fabricated in order to ensure that the fuselage design will be able to withstand the motor torque and landing loads.

8. Performance Results

8.2 Aircraft Performances

The completed prototype went through a ferry flight test and the resultant maximum velocity came out to be 81 ft/s, which is 15.7% higher than the expected mission 1 value. The maximum velocity was 95 ft/s, which is 5% lower than the expected value of 100 ft/s. The take off distance was estimated to be about 20 feet, which was also higher than the expected mission 1 value. There are few reasons the discrepancies. The fuse that limits 40A of current draw from the batteries was not installed, and the peak current draw from the test flight was 65A. The aircraft was using more 65% more than the allowable current. Therefore the prototype performed better than expected at take off distance and cruise velocity. The weighted unloaded prototype was 10.5 lbs, which was 31% above the projected empty weight. So the maximum speed might be limited due to this. An alternate wing structure is currently being built that would save up to 50% of wing weight.

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