AERSP 402B
Detailed Aircraft Design

Electric Light Sport Aircraft
The Flyin’ Lion

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Executive Summary

Electric aircraft are now becoming more popular and feasible. With the improvement of energy density and charge times of batteries, the electric aircraft has developed a niche in the light sport aircraft market. This electric light sport aircraft, known as the Flyin’ Lion, is an all-electric trainer with an innovative design. Concept studies introduced many parent aircraft into this design process. Aircraft ranging from the Piper J-3 Cub to the Pipistrel G2 Taurus were studied to optimize and create the best light sport aircraft possible while continuing to meet the FAA and FAR requirements.

Preliminary sizing was based on the parent aircraft designs to try to come up with an optimal configuration, which resulted in the Flyin’ Lion. With its natural laminar flow airfoils and an aerodynamic shape, the Flyin’ Lions’ strong performance characteristics outperform the competition. The Flyin’ Lion has a very short take-off and landing distance, and performs well for an electric light sport aircraft, having a battery life of two and a half hours with two 200 pound people onboard, with a back-up battery pack that is able to last for ten minutes.

In terms of performance, the Flyin’ Lion is capable of matching most conventional fuel burning aircraft. It has an 81 horsepower power plant that allows the Lion to obtain the maximum speed allowable for a light sport aircraft. This aircraft is a pure joy to fly, and continues to outperform the competition.
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Aircraft Mission Description

The Flyin’ Lion is an electric Light Sport Aircraft (LSA) designed for personal recreational use as well as a trainer, so novice pilots can earn a sport license or recreational pilot’s license. Figure 1 highlights a sample mission that the Lion would fly for a basic training session. This particular mission pushes the limits of the battery’s endurance and takes into consideration worse case scenarios for time of completion of each maneuver.

Figure 1: Sample mission for the Flyin' Lion
Several basic requirements for safety and classification need to be satisfied in order for the aircraft to meet its mission requirements. These requirements are highlighted in Table 1.

The battery performance and electric motor performance were projected three to five years in the future. Current technology limits the possible performance of the aircraft, but using projected trends in the performance of these components, the Lion can achieve these specifications.

The critical mission requirements include meeting the FAR and FAA requirements for a Light Sport Aircraft, and earn a Standard Airworthiness Certification. These requirements drive several of the performance parameters listed in Table 1. Additionally, two seats are needed so that the aircraft can function as a trainer aircraft. Most importantly the aircraft will run only on electricity. With emerging technology in batteries and electric motors, the Lion has the opportunity to take ahold in a new class of light sport aircraft.

Table 1: ELECTRIC Light Sport Aircraft (LSA) Mission Requirements

<table>
<thead>
<tr>
<th>Takeoff and Landing capabilities:</th>
<th>LSA take off will be achieved on a paved runway.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>LSA will be compliant with airport VFR (Visual Flight Rules) <em>only</em> during flight.</td>
</tr>
<tr>
<td></td>
<td>NOTE: No IFR (Instrument Flight Rules) allowed.</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Performance:</th>
<th>Maximum Velocity approximately 120 kt.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Stall speed should be 43 kt.</td>
</tr>
<tr>
<td></td>
<td>Sustained cruise at Steady Level Flight of 100 kt.</td>
</tr>
<tr>
<td></td>
<td>Service ceiling of 10,000 feet.</td>
</tr>
<tr>
<td></td>
<td>Minimum mission endurance of two hours.</td>
</tr>
<tr>
<td></td>
<td>Maximum range of 200 nautical miles.</td>
</tr>
<tr>
<td></td>
<td>Maximum takeoff weight of 1300 lbs.</td>
</tr>
<tr>
<td></td>
<td>Small backup battery system with a ten minute life cycle at 65 kt.</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Critical Requirements</th>
<th>Meets FAR and FAA requirements for a LSA / Standard Airworthiness Certification.</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Two Seat LSA that can be used to as a trainer to earn a sport or recreational pilots license.</td>
</tr>
<tr>
<td></td>
<td>Electric motor power plant.</td>
</tr>
</tbody>
</table>
Expanding on Table 1, the Lion’s takeoff distance was calculated to be 315 feet, well within the minimum 6000 feet length of common paved runways. The landing distance was also calculated to be 255 feet.

Also, the Lion must meet and comply with VFR (Visual Flight Rules). This means that the pilot must have a certain degree of visibility. The visibility will depend upon time of day as well as weather conditions. VFR are dictated by the FAA and the FAR.

The aircraft’s performance requirements come mostly from parent aircraft such as the Piper J-3 Cub, the Yuneec e430, and Pipistrel Taurus Electro G2. The Lion has been designed to be a better performing aircraft than its most similar parent the Yuneec e430, giving it a distinctive edge in the light sport category. These performance parameters include cruise speed and max speed. For example the Yuneec e430’s has a cruise speed of 52 knots and a max speed of 80 knots. The Lion will outperform the Yuneec e430 with a cruise speed of 100 knots and its max speed of 120 knots.

The stall speed for the Lion is 43 knots; the requirement needed to satisfy the FAR and the FAA requirements are 45 knots. The slower stall speed allows for easier landings, an ideal characteristic for a training aircraft.

The service ceiling of 10,000 feet that is imposed on the Lion comes from environmental factors. At altitudes much higher than this, the aircraft must carry supplemental oxygen and at higher altitudes pressurization becomes a requirement for any kind of sustained flight.

The mission endurance of 2 hours comes from the length of a longer than average training session. The range of 200 nautical miles is limited by the endurance of the batteries. Because the aircraft is a trainer, this should be sufficient time and distance for a typical training lesson. These values were confirmed by projected capabilities of the batteries that will be used in the aircraft.

The takeoff weight is mostly driven by FAR requirements. In order to be classified as a light sport aircraft the aircraft must be no heavier than 1,320 lbs. The aircraft has a takeoff weight of 1300 lbs. This weight was chosen, so that the Lion could fly with as much weight as possible and still be considered an LSA, as well as matching the performance characteristics. This weight came from the need to satisfy the performance requirements.
As an extra safety feature there will be a backup battery for emergencies. This battery will be capable of powering the Lion for 10 minutes of flight at a cruise speed of 65 knots. This will allow for the capability to safely land the aircraft.

**Concept Studies**

To begin the design process, the group members were initially asked to submit original sketches of what they thought the aircraft should look like. Figures 2, 3 and 4 demonstrate several of these early concepts.

![Figure 2: Early Design Concept for the Flyin’ Lion](image-url)
These original sketches were created as a first step. Even though none of these designs closely resemble the final design, it allowed the group members to think creatively and
brainstorm ideas. The initial sketches lead to more serious discussion and the decision to begin looking for a parent aircraft.

The easiest way to begin an aircraft design is to look at aircraft that have a similar mission to the one at hand. The “parent aircraft” approach gives the designer a launching point, something on which to base preliminary aircraft designs. The mission of the Lion pointed in the direction of several aircraft including the Piper J-3 Cub, Yuneec e430 and Pipistrel Taurus Electro G2, as well as the Pipistrel Virus. Initially, the Piper Cub was selected as the parent aircraft.

![Figure 5: Piper J-3 Cub](image)

The Piper, depicted in Figure 5, was selected because its mission is similar to the mission that the Lion was being designed for. It is also a very safe aircraft which was appealing because the Lion is a trainer. Its proven track record as a trainer made it an appealing parent aircraft.
Figure 6 is the initial design for the Lion using the Piper J-3 Cub as a parent aircraft. However, after a preliminary design review, the idea of having the Piper as a parent aircraft was rejected. Although it is a very good aircraft, the Piper has some drastic differences. Specifically, the Piper is powered using fuel and is constructed from aluminum and fabric. The Lion needs to
be electrically powered and needs to be made of lightweight materials to satisfy weight requirements.

After the initial design was scrapped, more research was put into finding electric light sport aircraft made with more lightweight materials. These are the parent aircraft that the Flyin’ Lion is based on.

Figure 7: Yuneec e430

The Yuneec e430, depicted in Figure 7, is an all composite, electric light sport aircraft. The Yuneec had several aspects that were adopted into the design; however, it was also not a perfect match. The high wing design provides good visibility for the pilot and the winglets decrease the induced drag on the aircraft. Also, the Yuneec is able to hold two passengers side-by-side yet it still has an aerodynamic body shape. The group felt as though the V-tail design and the shape of the wings could be improved upon. This change would help better meet the mission requirements by decreasing the complexity of the control system.
The Taurus, depicted in Figure 8, provided other aspects of the aircraft that the Yuneec could not. The shape of the wing and the T-tail design were inspired by the Taurus. The wing is a bit different in that it has a straight trailing edge and the leading edge is tapered near the wing tips. This improves the load distribution yet still maintains an easily manufactured wing shape.

Another parent aircraft of the Lion is the Pipistrel Virus. The Virus is an ideal trainer, and is also a motor glider. Thus, it has the option of a motor shut off and performing like a glider if the pilot would like to. It has a T-tail design, and is also a high wing. It is incredibly efficient, but is not run by an electric power plant.
Combining aspects from both the Yuneec, Taurus, and the Virus, the Flyin’ Lion’s general design was created and is depicted in Figure 9.
Procedure Outline

Performance

The first calculations of the design were based on the mission requirements and were then compared to the parent aircraft. The group compiled the data from the parameter equations below into a table of specifications that can be found in the Appendix I.

The taper ratio for a partially tapered wing was calculated by weighting the tapered and un-tapered ratios for the sections of the wings. This is calculated using the equation

$$\lambda = \frac{(A_1 \cdot \lambda_1) + (A_2 \cdot \lambda_2)}{A_{Total}}$$  \hspace{1cm} (1)

The taper ratio for the wing was calculated using the root chord and tip chord ratio for the wing of the aircraft.

$$\lambda_i = \left( \frac{C_t}{C_0} \right)_i$$  \hspace{1cm} (2)

The standard taper ratio of a light sport aircraft is typically between 0.6 – 1.

The planform area for a straight-tapered wing can be calculated from the wing span, root chord, and taper ratio. The Flyin’ Lion has a partially tapered wing which changes the method of calculating the planform area. The area was then calculated by adding a rectangular area to the trapezoidal area of the semi-span of the wing.

$$S = \frac{c_0(1 + \lambda)b}{2} = (A_{Rectangle} + A_{Trapezoid}) \cdot 2$$  \hspace{1cm} (3)

Using the determined planform area, the aspect ratio can be calculated.

$$AR = \frac{b^2}{s} = \frac{2b}{c_0(1 + \lambda)}$$  \hspace{1cm} (4)
The importance of the aspect ratio for the overall aircraft will affect the induced drag and the lift to drag ratio of the aircraft during the preliminary calculations. A typical aspect ratio for personal and utility aircraft ranges between 5 and 8.

Next, the weight of the aircraft needed to be determined for the performance specifications. The weight can affect many different parameters in the design process. Such factors affected by weight are the structural integrity, aerodynamic drag, performance, and sizing parameters. It should be noted that the maximum allowable gross weight for an LSA is 1320 pounds, and that the gross weight of the Lion will not vary during flight.

The total gross takeoff weight of the aircraft is composed of the empty structural weight, batteries, passengers and cargo weights.

\[
W_{TOWG} = W_e + W_b + W_{p+cargo}
\]  \hspace{1cm} (4)

Total Gross Weight

The constant weight reflects the removal of the liquid fuel system of a conventional aircraft. The standard gross takeoff weight range of the parent aircraft was found to be between 946 and 1220 pounds.

Using the aforementioned calculations, the first conceptual design estimations for the critical performance parameters were determined. These parameters included the wing loading, lift coefficients, drag coefficients, thrust, lift-to-drag ratio, ground roll, and power estimations. These parameters were influenced by parent aircraft and design specification requirements.

Wing Loading

\[
\text{wing loading} = \frac{W_{TOWG}}{S}
\]  \hspace{1cm} (5)

The wing loading can be determined from Equation (5). This is a very important parameter that is based on the weight and planform area of the wing. The wing loading has a direct effect upon many parameters such as the lift, drag, and thrust.

In order to obtain these performance parameters, various equations were utilized to standardize our values. Initially, the temperature was needed at the specific altitudes desired for the flight envelope. This was found using the standard temperature lapse rate.
Temperature Lapse Rate

\[ T = T_0 - 3.57 \left( \frac{h}{1000} \right) \]  

(6)

Next, the temperature ratio was used in order to acquire the density ratio.

Temperature Ratio

\[ \theta = \frac{T}{T_0} \]  

(7)

The standard density ratio, was utilized to ensure uniformity of the calculations.

Density Ratio

\[ \sigma = \frac{\rho}{\rho_0} = \theta^{4.256} \]  

(8)

Using information calculated from the above equations, the total drag can be determined. Compiling drag from all the components of an aircraft is probably the most crucial and perhaps the most difficult step in determining an aircraft’s performance. Several key performance parameters depend directly on the force of drag an aircraft experiences in crucial stages of flight. For example, it would be difficult to determine the time to climb to a certain altitude without drag estimates. Generally speaking, cruise, take-off, turning, and landing behavior cannot be predicted without an accurate estimation of drag at those conditions.

The wing of an aircraft is the largest contributor of drag. At low airspeeds, induced drag is the main factor but profile drag also contributes significantly. X-Foil was used to give simple estimations of the lift and drag coefficients of the airfoil section. X-Foil is a computer program which uses potential flow and boundary layer analysis to determine lift and drag on an airfoil. Refer to Appendix III for X-Foil results on the wing of the aircraft.

The induced drag coefficient was calculated using parameters of the airfoil.

Induced Drag Coefficient

\[ C_{dI} = \frac{C_I^2}{\pi e AR} \]  

(9)
Using the Panel Method for Windows, the span efficiency was calculated as 0.961 for the wing of the Flyin’ Lion. For further discussion on the Panel Method for Windows, refer to Appendix II.

The induced drag was then calculated and added to the X-Foil provided profile drag. The result was the total drag.

\[
D = D_{\text{Induced}} + D_{\text{Profile}}
\]

Drag Equation

\[
\sum_{n=1}^{n} \frac{1}{2} \rho V^2 C_{d0} + \sum_{n=1}^{n} \frac{1}{2} \rho V^2 C_{d0}
\]

This is also a valid procedure for the horizontal tail.

To calculate the drag contributed by the fuselage to the entire aircraft, skin friction drag and pressure drag were taken into account. Induced drag was assumed to be negligible because the fuselage does not produce a significant amount of lift.

To begin, a side view of the fuselage was utilized.

![Side View of Fuselage](image-url)
Figure 11: Fuselage Break-Up

Figure 11 depicts the aircraft discretized into its various shapes. The left diagram is a frontal view depicting the concentric circles of each individual piece. The right diagram depicts the aircraft “un-rolled” into a flat sheet and connected via straight lines to form the trapezoids used to solve for the area of each segment.

To solve for the wetted area, the equations for the circumference of a circle and the area of a trapezoid are used.

\[
\text{Circumference of a Circle} \quad C = \pi D \quad (11)
\]

\[
\text{Area of a Trapezoid} \quad A = \frac{(C_1 + C_2) \times h}{2} \quad (12)
\]

The diameter of the circles can be seen on Figure 10 as the distance “D”. The circumferences of the various circles were then used as the lengths in Equation (12). Solving Equation (12) for every trapezoid on the aircraft provided an approximate surface area for the aircraft’s fuselage.

Next the skin friction coefficient for each trapezoidal section was calculated using laminar and turbulent skin friction coefficients.
Laminar Skin Friction Coefficient

\[ C_{fL} = \frac{0.664}{\sqrt{RE_x}} \] (13)

Turbulent Skin Friction Coefficient

\[ C_{fT} = \frac{0.0576}{RE_x^{0.2}} \] (14)

It was assumed that most of the flow over the fuselage of the aircraft was turbulent; however, due to the aerodynamic shape of the fuselage, there was still a small section at the front of the aircraft that had laminar flow. That section was assumed to be 1.5 feet from the front tip of the aircraft. However without doing comprehensive wind tunnel testing this might not be entirely accurate.

The final skin friction drag was determined by summing all of the contributions of each discretized section.

\[ D_{FP} = \frac{1}{2} \rho V^2 \sum_{i=1}^{n} A_i C_{fi} \] (15)

Given the skin friction drag of the aircraft, the next task was to determine the pressure drag of the aircraft. The pressure drag of any aircraft is very difficult to determine so an approximation was used. Based on the streamlined shape of the fuselage and low velocities, the addition of pressure drag was approximated to add 40 percent to the calculated skin friction drag.

The drag on the vertical tail was calculated using X-Foil. Because the vertical tail does not generate lift, its only major contribution to the overall drag of the aircraft is profile drag.

The total drag value was calculated by adding the components from the tail surfaces, wing surface, and fuselage. These values are sufficiently accurate for the purposes of calculating the performance characteristics of the Flyin’ Lion.

The coefficient of the thrust needs to be calculated to further determine aircraft performance characteristics.

\[ c_T = c_D + c_{w_{TOGw}} \sin(\gamma) \] (16)
Note that the thrust is directly related to drag in the condition of steady level flight, leading to a simplified relationship.

Coefficient of Thrust, Simplified \[ c_T \approx c_D \] (17)

Utilizing these non-dimensional equations, thrust of the aircraft can be calculated. The lift-to-drag ratio can now be determined as well.

Lift/Drag Ratio \[ \frac{lift}{drag} \text{ ratio} = \frac{c_l}{c_d} \] (18)

This is necessary to predict the maximum glide range in case of engine out conditions. This is a vital parameter because the aircraft will be used primarily as a trainer. The estimated lift-to-drag ratio is a key factor used to select an airfoil for the Flyin’ Lion.

From the mission requirements and parent aircraft, the desired thrust range was determined for the different flight parameters. The thrust is used to calculate the rate-of-climb, ground roll, and thrust-to-weight ratio.

Thrust \[ T = D + Wy \] (19)

Note that Equation (19) is for a small angle approximation.

Now a thrust-to-weight ratio can be calculated. Note that this directly affects the take-off and landing ground roll for the LSA.

Thrust/Weight Ratio \[ \text{Thrust} - to - \text{Weight} = \frac{T}{W} \] (20)

The distance of the ground roll is important to meet the mission requirements that were established. It can be crucial to the design goals of the aircraft.
Ground Roll Distance

\[ S_g = \frac{1.21 \left( \frac{W}{S} \right)}{g \rho \omega c_{l_{\text{max}}} \left( \frac{L}{W} \right)} \]  \hspace{1cm} (21)

Note that Equation (21) does not consider ground effect lift.

The minimum power required and power available are vital parameters that can be utilized to help find rate-of-climb and other important specifications.

Power Required

\[ P_{\text{reqd}} = DV \]  \hspace{1cm} (22)

Power Available

\[ P_{\text{avail}} = \eta \times BHP \]  \hspace{1cm} (23)

Where \( \eta \) is the prop efficiency and \( BHP \) is the break horse power in Equation 23.

The rate-of-climb can be calculated after power available and power required are determined.

Rate of Climb

\[ \frac{R}{C} = \left( \frac{P_{\text{avail}} - P_{\text{reqd}}}{w} \right) \]  \hspace{1cm} (24)

The rate-of-climb must be sufficient enough to clear the required ground obstacles on takeoff. It is also essential to reach an optimal rate of climb to arrive at cruise altitude; thus, maximizing flight time at cruise and reducing the overall power consumption.

Next, the calculations of the endurance and range were performed in order to satisfy the mission requirements. The endurance calculation of an electric LSA is different than that of a normal aircraft because it is not fuel powered. However, the premise is still the same. The simple idea of endurance is how much fuel the aircraft has in relation to how much power it takes to fly at a desired speed. Thus, the endurance was derived.

Endurance Equation

\[ E = \frac{C W_{\text{bat}} \eta_{\text{batteries}} \eta_{\text{controller}} \eta_{\text{prop}}}{P_{\text{reqd}}} \]  \hspace{1cm} (25)
It is important to note that the energy density of the batteries is noted as $C$ in this equation. Some energy is lost due to lack of efficiency between the batteries and the output from the engine. The goal for the Flyin’ Lion was to fly for at least 2 hours, which can be achieved with 400 pounds of batteries.

The fuel efficiency for the airplane was based on the electrical power system. The fuel was considered as the electrical current which was available from the Lithium Ion battery packs and delivered through the charging system. The fuel efficiency remained constant throughout flight at various power settings since the battery controller regulated the current flux through the system. The two main efficiency factors which effect the electrical system were the battery efficiency, 90%, and the controller efficiency, 95%. These efficiencies are reasonable values based upon current technology.

The range can then be determined from the endurance. Simply multiply the desired airspeed by the endurance to obtain the range of the aircraft

\[ R = EV \]  

(26)

**Stability & Control**

After the basic calculations for sizing the aircraft and obtaining the performance parameters, the empennage was sized using the equations for the horizontal tail volume and vertical tail volume ratios.

Horizontal Tail Volume Ratio

\[ \bar{V}_h = \frac{S_h \bar{T}_h}{S \bar{C}} \]  

(27)

Vertical Tail Volume Ratio

\[ \bar{V}_v = \frac{S_v \bar{T}_v}{S \bar{C}} \]  

(28)

The important factors were the horizontal and vertical tail areas, volumes, and locations. The size of the tail sections helped determine the minimum amount of drag on the aircraft, and stabilize and control the aircraft.
In order for the Lion to be able to achieve flight the first stability quantity that needs to be calculated is the neutral point of the aircraft.

Neutral Point
\[
    h_n = h_{nwb} + \frac{a_t}{a} \bar{V}_h \left(1 - \frac{d\varepsilon}{d\alpha}\right)
\]  
(29)

The neutral point is the aft limit of the cg location. The neutral point of the aircraft is calculated using parameters from the aircraft including the lift slope of the aircraft, the lift slope of the tail, the horizontal tail volume coefficient, the neutral point of the wing body, and the downwash parameter. All of these factors play into the neutral point of the Lion.

Next the forward limit of the center of gravity must be determined. The forward limit is determined by the flight conditions at takeoff. This is when \(C_L\) is at its max and the aircraft is in ground effect.

\[
\left(\frac{x_C}{c}\right)_{fwd} = \frac{x'_a}{c} + \frac{\bar{V}_h}{1 + F} \frac{a_t}{a_w} \left(1 - \frac{d\varepsilon}{d\alpha}\right) - \frac{1}{C_{L_{max}}} \left( C_{m_{a}} + \frac{\bar{V}_h}{1 + F} a_t \left(i_w - i_t - \tau\delta_{max}\right) \right)
\]

Forward C.G. Location \hspace{1cm} where \(F = \frac{a_t}{a_w s_w} \left(1 - \frac{d\varepsilon}{d\alpha}\right)\)  
(30)

The equation takes into account many aspects of the aircraft. This includes the lift slopes for the tail wing and aircraft, the surface areas and inclination of the tail and wing, the effectiveness of the tail, the location of the fuselage aerodynamic center, and the maximum lift coefficient.

Now that the forward and aft limits for the aircraft have been determined the internal components can be placed inside of the Lion so that its center of gravity is within the limits. To determine the center of gravity each major contributor of weight was given a position in the fuselage and then a moment calculation was used to determine the equivalent location of the center of gravity.

\[
\text{Aircraft C.G. Location} \hspace{1cm} \frac{\sum (\text{weight} \times \text{distance from datum})}{\text{total weight}} = \text{CG location from datum}
\]  
(31)
**Aileron Sizing**

After viewing case studies of the Flyin’ Lions parent aircraft, an average roll rate was determined to be about 40 deg./sec. To obtain this roll rate, the ailerons size needed to be determined based on this specification at the cruise velocity of 100 knots. The ailerons were limited to 10 degrees of deflection due to the parent aircrafts average aileron deflection. Then the outboard location of the aileron was decided to be 22 feet from the centerline of the aircraft, as seen in Figure 9. From these details, it was determined that the inboard aileron should begin at 12 feet from the centerline of the aircraft. This gives an approximate roll rate of 45 deg./sec. This makes the Flyin’ Lion an exciting and exhilarating aircraft to maneuver. To see the Matlab code used to calculate the aileron sizing, view Appendix IV.

**Rudder Sizing**

Rudder and vertical tail sizing is necessary to ensure lateral directional stability of an aircraft. An adequately sized tail is needed to ensure proper control of the aircraft during gusts and sideslips. The rudder is used to coordinate turns as well as to correct heading. The major driving variables for tail and rudder sizing are vertical tail lift slope, the weathercock stability derivative, and the vertical tail volume coefficient. Another factor that influences the effectiveness of the tail is the size and shape of the fuselage. The geometry of the fuselage can induce side wash effects which alter the effective angle of attack and airflow the tail sees.

In order to ensure static lateral directional stability, the aircraft needs a positive weathercock stability derivative. This physically means that if a gust yaws the aircraft to the right, the aircraft will automatically return to the trimmed state, and vice versa. Assuming the side wash coefficient is 0.17, based on the Morelli document, the weathercock stability derivative results in

$$C_{n\beta} = a_v \bar{V}_v \left(1 - \frac{\partial \sigma}{\partial \beta}\right)$$
where $a_v$ is the vertical tail lift slope, $\bar{V}_v$ is the vertical tail volume coefficient, and $\frac{\partial \sigma}{\partial \beta}$ is the side wash coefficient. Due to the destabilizing effect of the Lion’s fuselage, the total weathercock stability is reduced by a value of 0.02.

FAR Sec. 23 states that the aircraft must be capable of stabilizing in a maximum sideslip of 11.5°. This is the minimum amount of sideslip the aircraft can be exposed to, although, it is desirable for the aircraft to operate in higher gust conditions. The need to meet these conditions led to our rudder size. The Flying Lion’s rudder power results in

$$c_{n\delta v} = a_v \tau \bar{V}_v$$

where $\tau$ is an efficiency for the vertical tail. With rudder power, rudder deflection to meet FAR requirements is calculated as

$$\delta_v = \frac{c_{n\beta}}{c_{n\delta v}} \beta$$

where $\beta$ is the sideslip angle. Once the rudder deflection was obtain, a factor of safety was obtained based on a maximum rudder deflection of 30°.

**Motor, Controller, and Batteries**

![Figure 12: Yuneec Power Drive 60 Motor](image)
The Yuneec Power Drive 60 in Figure 12 was chosen as the electric motor for the Flyin’ Lion. The following are the specifications for this motor.

**Table 2: Engine Specifications**

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
<th>Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power</td>
<td>81</td>
<td>Horsepower</td>
</tr>
<tr>
<td>Rotational Speed</td>
<td>2400</td>
<td>Revolutions/Minute</td>
</tr>
<tr>
<td>Weight</td>
<td>66</td>
<td>lbs</td>
</tr>
<tr>
<td>Diameter</td>
<td>11</td>
<td>inches</td>
</tr>
</tbody>
</table>

Because electric battery technology is improving at an exponential rate, the specifications for the battery system are estimated using projections 3-5 years into the future. The motor will be powered by Lithium-Ion batteries weighing 400 lbs and having an energy density of 240 WHr/kg. These parameters will be used later to calculate endurance and range of the aircraft.

A Yuneec power controller will accompany the batteries and the motor. The Power Block 60 is encased in a plastic, fire-retardant material. It weighs around 22 lbs.

To reduce the overall temperature of the motor, battery, and controller system, a cooling duct and system was designed for the aircraft. Electrical controllers and battery systems can reach high temperatures and need to be lowered since the air will be recirculating in the cabin. These temperatures can create malfunctions in sensitive avionics and electronics of the aircraft.

**Specifications**
Figure 3: Scissor Diagram for Flight Constraints

Figure 13 shows the scissor diagram for the Lion. In order to set the design limits for the electric LSA, a scissor diagram will be used to ensure that the parameters were feasible. The diagram allows for a quick comparison of different design specifications chosen for the Lion. Several major components are for the take-off, landing, and level turn constraints.

To calculate the take-off constraint, the ground roll equation was utilized. The wing loading was incremented by small values to allow for proper comparison of the thrust to weight ratio. The landing constraint ensures that the wing loading is within a range to endure the stresses of landing the aircraft. The level turn load factor was used to allow a factor of safety for operational flight. The load factor that was used for this diagram was two.

This diagram is a critical tool for the successful design of this aircraft. Every time that a parameter has been updated, the scissor diagram has been referenced to ensure that the aircraft remains inside the design area. As it can be seen, the current data fits well into the design space constraints. Future scissor diagrams will be used to help specify the design parameters for stability and control of this aircraft.
Table 3: Aircraft Design Geometry and Specifications

<table>
<thead>
<tr>
<th>Parameters</th>
<th>Specifications</th>
</tr>
</thead>
<tbody>
<tr>
<td>Taper Ratio</td>
<td>0.94</td>
</tr>
<tr>
<td>Chord (average)</td>
<td>3.25 ft</td>
</tr>
<tr>
<td>Length</td>
<td>23 ft</td>
</tr>
<tr>
<td>Wingspan</td>
<td>46 ft</td>
</tr>
<tr>
<td>Height</td>
<td>9 ft</td>
</tr>
<tr>
<td>Wing Area</td>
<td>156 ft²</td>
</tr>
<tr>
<td>Aspect Ratio</td>
<td>13.56</td>
</tr>
<tr>
<td>Empty Weight</td>
<td>900 lbs</td>
</tr>
<tr>
<td>Battery Weight (lithium Ion battery Packs)</td>
<td>400 lbs</td>
</tr>
<tr>
<td>Max T.O. Weight</td>
<td>1300 lbs</td>
</tr>
<tr>
<td>Energy Density</td>
<td>240 WH/kg</td>
</tr>
<tr>
<td>Wing Loading</td>
<td>6.8 lb/ft²</td>
</tr>
<tr>
<td>$V_h$</td>
<td>0.82</td>
</tr>
<tr>
<td>$S_{ht}$</td>
<td>16 ft²</td>
</tr>
<tr>
<td>$b_{ht}$</td>
<td>8 ft</td>
</tr>
<tr>
<td>$c_{ht}$</td>
<td>2 ft</td>
</tr>
<tr>
<td>$AR_{ht}$</td>
<td>4.0</td>
</tr>
<tr>
<td>$V_v$</td>
<td>0.042</td>
</tr>
<tr>
<td>$S_{vt}$</td>
<td>13.5 ft²</td>
</tr>
<tr>
<td>$b_{vt}$</td>
<td>4.50 ft</td>
</tr>
<tr>
<td>$c_{vt}$</td>
<td>3.00 ft</td>
</tr>
<tr>
<td>$AR_{vt}$</td>
<td>1.5</td>
</tr>
</tbody>
</table>

Table 3 shows the specifications for the geometry of the aircraft. The majority of the parameters above have been modified to reflect an update in the design iteration process. The appropriate changes were made after consideration for all the design areas. Several conflicts occurred during the development of the aircraft, which resulted in the current values shown above (Table 3).
Drag

Table 4: Total Drag Build-Up

<table>
<thead>
<tr>
<th>Airspeed (kt)</th>
<th>Landing Gear (lbs)</th>
<th>Fuselage Drag (lbs)</th>
<th>Wing Drag (lbs)</th>
<th>Empennage Drag (lbs)</th>
<th>Total Drag (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>40</td>
<td>0.64</td>
<td>3.46</td>
<td>53.17</td>
<td>0.29</td>
<td>57.56</td>
</tr>
<tr>
<td>43</td>
<td>0.74</td>
<td>3.94</td>
<td>47.11</td>
<td>0.32</td>
<td>52.12</td>
</tr>
<tr>
<td>45</td>
<td>0.81</td>
<td>4.28</td>
<td>43.8</td>
<td>0.34</td>
<td>49.23</td>
</tr>
<tr>
<td>50</td>
<td>1</td>
<td>5.17</td>
<td>37.44</td>
<td>0.4</td>
<td>44.01</td>
</tr>
<tr>
<td>75</td>
<td>2.25</td>
<td>10.73</td>
<td>26.19</td>
<td>0.76</td>
<td>39.93</td>
</tr>
<tr>
<td>100</td>
<td>4</td>
<td>18</td>
<td>28.45</td>
<td>1.24</td>
<td>51.69</td>
</tr>
<tr>
<td>120</td>
<td>5.76</td>
<td>24.99</td>
<td>34.33</td>
<td>1.72</td>
<td>66.8</td>
</tr>
<tr>
<td>140</td>
<td>7.84</td>
<td>32.98</td>
<td>42.57</td>
<td>2.29</td>
<td>85.68</td>
</tr>
</tbody>
</table>

Table 4 is a summary of the total drag build-up for the Lion.

Figure 14: Drag vs. Airspeed at Cruise
Figure 14 demonstrates the drag versus airspeed at the cruise altitude. Note that this is curve is created for standard gross weight. The stall of the aircraft is shown to be right around 43 knots, which fits within the specifications of a light sport aircraft. It also can be seen that the drag significantly increases with airspeed, which is an expected effect for this aircraft. It can also be seen that the maximum L/D occurs at a value of around 65 knots.

**Power**

![Power Required vs. Velocity](image)

Figure 15: Power vs. Velocity at Cruise

The power available curve in Figure 15 is based on the total power available for the Lion. After taking into account the efficiency losses of the batteries and the control system, a general power available was determined. Since the power plant is purely electric, there are no efficiency losses with altitude except for that of the propeller. Thus, calculating the propeller efficiency was necessary for the power available curve. To calculate power available, the advance ratio (J) must be determined.
Advance ratio

\[ J = \frac{V}{ND} \]

Note that it is purely dependent on velocity (V), the engine speed in revolutions per second (N), and the diameter of the propeller (D).

**Figure 16: Propeller Efficiency versus Advance Ratio**

From this calculation, Figure 16 was used to determine the propeller efficiency. The Flyin’ Lions propeller is a fixed pitch version of the Vario, a propeller made by Pipistrel. Since efficiency data on the Vario was not able to be directly obtained, Figure 16 is a subsequent fixed pitch propeller for the Cessna 172R. This curve is a reasonable approximation for the propeller used on the Flyin’ Lion. As the speed of the aircraft increases, the efficiency of the propeller changes, giving the Flyin’ Lion the power available as shown in Figure 15.

Power Required: To obtain the power required, one must find the total drag of the aircraft at different velocities. It makes sense to think of the power required as the total power needed to
overcome the drag at a certain velocity. Thus, Equation 22 shows the simple relation of power required to drag.

Thus, the curve obtained for the required power of the Flyin’ Lion is shown in Figure 15. Note that it follows the same basic trend of a drag curve. It can be seen that the power required does change with different altitudes, unlike the power available. This is due to the fact that the drag changes with altitude, making the power curve shift/rotate to the right and up as shown in Figure 15. To obtain the power obtained at different altitudes, the drag coefficient needed to be determined. To do this, a plot of drag coefficient versus lift coefficient was created, as shown in Figure 17.

![CD vs. CL](image)

**Figure 17: Cd versus Cl curve**

Next, it was necessary to find the lift coefficient at the different altitudes of interest, since they change with density. However, the lift coefficient and drag coefficient can still be found directly with Figure 17. Thus, the drag coefficient can be determined. From this, the Power required for different altitudes can be found.
Power Required at different altitudes

\[ P'_{reqd} = W' \sqrt{\frac{2 (\frac{W}{S}) \left( \frac{C_D}{\sqrt{\frac{\beta}{C_{L}}}} \right)}{\rho}} \]

The next important step is to determine the rate of climb.

**Rate of Climb**

From the power available and power required, the rate of climb of an aircraft can be determined. Subtracting the power required for the aircraft to perform at a given setting from the power available at that setting, the excess power is found. The excess power of the aircraft is directly correlated to the rate of climb as shown in Equation 24.

The excess power can be visually represented using a power versus velocity graph such as the one below.

**Figure 18: Visualization of Excess Power**
Calculating the rate of climb using Equation 24, Figure 19 was created.

Figure 19 shows the rate of climb at various altitudes and airspeeds and is used to determine the absolute ceiling of the aircraft. As the aircraft increases its altitude, it cannot continue to climb without increasing its airspeed or it will stall. Therefore, each set of blue dots represents a new airspeed. Due to the large amount of available power of the aircraft, the absolute ceiling is limited by the speed requirements of an LSA. The Flyin’ Lion will reach its maximum allowable airspeed of 120 knots and stall around 53000 feet (making 53000 feet the absolute ceiling) even though the aircraft still has power available.
Data obtained from Figure 19 can also be used to calculate the time to climb to any given altitude

\[
\text{Time-to-Climb} \quad t = \frac{h_{abs}}{(R/C)_0} \ln \left[ \frac{1}{1 - \frac{h}{h_{abs}}} \right]
\]

where \( h_{abs} \) is the absolute ceiling of the aircraft, \( h \) is the desired altitude, and \( (R/C)_0 \) is the rate of climb of the aircraft at sea level. Using Equation 32, it was determined that the Lion could climb to a cruising altitude of 5000 feet in approximately 4.5 minutes.

**Stall Speed**

Using Figure 19, Figure 20 was created. It summarizes the stall speeds at various altitudes.

![Stall Speed vs. Altitude](image)

*Figure 20: Stall Speed versus Altitude*
Figure 21 shows the lift to drag ratio versus airspeed at the cruise altitude and total gross weight. Note that as airspeed increases, the lift to drag ratio increases and peaks at approximately 65 knots. The cruise speed of the Lion fits well into the optimal lift to drag ratio. By cruising within the optimal lift to drag ratio, the Lion will get optimal endurance during cruise.
**Range**

The plot below depicts the range of the Lion at various airspeeds.

![Range vs. Airspeed](image)

**Figure 22: Range versus Airspeed**

Figure 22 shows the range at cruising power of 100 knots was determined to be 222 miles and the maximum range peaks at 300 miles. This maximum range occurs at a speed of 65 knots which is approximately the speed for maximum excess power of the Lion.
Lift Curves and Drag Polars

Figure 23: Lift vs. Angle of Attack

Figure 23 depicts the lift coefficient versus angle of attack at various flight conditions. The lift coefficients at the given flight conditions meet mission specifications for stall speed and takeoff.
Figure 24: Cl vs. Cd

Figure 24 depicts the lift coefficient versus the drag coefficient. Since the Lion usually operates in a lift coefficient range of 0.2 – 0.65, the drag bucket for the given mission and flight conditions is favorable.
Table 5: Aircraft Specifications

<table>
<thead>
<tr>
<th>Parameters</th>
<th>Specifications</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max airspeed</td>
<td>120 kts</td>
</tr>
<tr>
<td>Cruise Airspeed</td>
<td>100 kts</td>
</tr>
<tr>
<td>Stall Airspeed</td>
<td>43 kts</td>
</tr>
<tr>
<td>Max CL</td>
<td>1.31</td>
</tr>
<tr>
<td>T.O. Cd</td>
<td>0.016</td>
</tr>
<tr>
<td>Max L/D</td>
<td>33</td>
</tr>
<tr>
<td>Powerplant @ 2,400 RPM</td>
<td>81 hp / 60 kW</td>
</tr>
<tr>
<td>Yuneec Power Drive 60</td>
<td></td>
</tr>
<tr>
<td>(Thrust/Weight) T.O.</td>
<td>0.314</td>
</tr>
<tr>
<td>Thrust T.O.</td>
<td>408 lbs</td>
</tr>
<tr>
<td>Power Available @ T.O.</td>
<td>64.8 hp</td>
</tr>
<tr>
<td>Power Required @ T.O.</td>
<td>7.5 hp</td>
</tr>
<tr>
<td>T.O. Distance</td>
<td>315 ft</td>
</tr>
<tr>
<td>Rate of Climb</td>
<td>1235 ft/min</td>
</tr>
<tr>
<td>Service Ceiling</td>
<td>10000 ft</td>
</tr>
<tr>
<td>Range</td>
<td>200 nm</td>
</tr>
<tr>
<td>Endurance</td>
<td>2.5 hrs</td>
</tr>
<tr>
<td>Landing Distance</td>
<td>250 ft</td>
</tr>
</tbody>
</table>

Table 4 shows the continuation of parameters and specifications for the aircraft. All specifications were compared and verified to be in the established range of performance trade studies for a light sport aircraft. Through a constant iteration process, the performance specifications were optimized to meet the mission and design requirements.
Turning Performance

An analysis of turning performance is necessary to develop a flight envelope for the aircraft. There are two key turning performance analyses which must be evaluated; instantaneous pull-up and push-over, and steady level banked turn.

Instantaneous pull-up is a maneuver in which the aircraft is initially in steady level flight followed by an instantaneous constant change in pitch attitude. In order to calculate the turning radius in an instantaneous pull-up or push-over, the load factor needs to be found.

\[
\text{Load Factor} \quad n = \frac{L}{D} * \frac{T}{W} \quad (32)
\]

Once the load factor is found for a range of airspeeds, the instantaneous turn radius can be determined.

- **Instantaneous Pull-Up Radius**
  \[
  R = \frac{V^2}{g(n - 1)} \quad (33)
  \]

- **Instantaneous Push-Over Radius**
  \[
  R = \frac{V^2}{g(n + 1)} \quad (34)
  \]

A range of instantaneous turn radii for the Lion are shown in Figure 25 below.

![Instantaneous Turn Radius vs. Velocity](image)

**Figure 25: Instantaneous Turn Radius versus Velocity**
A steady level turn is defined as a sustained constant speed, constant altitude maneuver. Similar to the instantaneous turning maneuvers, a steady level turn radius is dependent on the load factor as defined in equation 32. With the load factor, the steady level turn radius can be evaluated.

\[
R = \frac{V^2}{gV^n^2 - 1}
\]  

(35)

A range of steady level turn radii for the Lion are shown in Figure 26 below.

---

**Figure 26: Steady Level Turn Radius versus Velocity**

Combining the two turning radii plots with structural limits on the aircraft will lead to a flight envelope. The structural limits of the Lion have yet to be determined.
Flight and Gust Envelope

The flight and gust envelopes for the Flyin’ Lion are shown in Figure 27. The accelerated flight conditions are calculated using

$$n = \frac{\rho V^2 SC_{L_{max}}}{2W}$$

where n represents the load factor on the aircraft. Exceeding this limit would stall the aircraft.

To avoid structural damage, the FAR limits the aircraft to a positive load factor at 3.8 Gs and negative load factor -1.5 Gs. Note that these are structural yield limits. The ultimate structural limits would have a certain factor of safety associated with it allowing the pilot to fly above 3.8 G’s without breaking the aircraft. According to FAR standards, the limit load factor in the negative region varies linearly from cruise speed to a load factor of zero at the dive speed.

Although the Lion is limited, as a light sport aircraft, to a maximum airspeed of 120 knots, it is still possible for the aircraft to achieve a greater airspeed. To ensure safety at higher
airspeeds, which may occur during a dive, the limit airspeed was determined as 50% higher than
the cruise speed and the aircraft was designed to handle such loads.

The red lines in Figure 27 indicate the gust envelope. In the case where the aircraft
encounters a gust, the flight condition of the aircraft will shift to the corresponding gust flight
condition. According to Figure 27, the Lion cannot handle a gust of 66 feet per second at any
time during flight. However, this condition rarely ever happens and the Lion should not be
flying in these sorts of conditions to begin with. The Lion would also have difficulties handling
a 30 feet per second flight condition, although it fairs slightly better especially if the aircraft
experiences a negative load factor as a result of the gust. At gusts of 15 feet per second the Lion
is well within the flight envelope for most airspeeds, meaning it is safe to fly. To determine the
gust envelope, the aircraft mass factor must first be calculated.

$$\mu = \frac{2(W/S)}{g\vec{c}\rho}$$

Using the mass factor, the gust alleviation factor can be calculated.

$$K_g = \frac{0.88\mu}{5.3 + \mu}$$

The gust alleviation factor is simply an empirical formula that is used to compensate for the gust
gradient, the airplanes response, and the lag in the change in lift due to the change in the angle of
attack. Finally, the load factor can be calculated:

$$n = 1 \pm \frac{K_g U_{de} V_c \rho_0 \alpha}{2(W/S)}$$

where $U_{de}$ is equal to the gust speed and $V_c$ equal to the aircraft’s equivalent airspeed. By
changing the airspeed of the aircraft to the values corresponding to those in Figure 27, the red
lines can be made.

**Weight and Cost Estimation**

In order to estimate costs, the aircraft was broken down into various sections. Several of
the sections were: the engine system and nacelle, the main fuselage and interior cabin, wing
structure, and the empennage. Each of the sections were divided into a detailed component listing which contained the manufacturer specifications, component unit weights, number of parts, and part costs. The total kit price estimate was determined from the entire parts breakdown. The overall kit price was estimated to be $46,000 for all of the aircraft components. The price was competitive compared to a parent aircraft, the Pipistrel Virus, which cost $76,509 for the entire 400 hour aircraft kit. Various components were approximated and may be over/under estimated. The aircraft skins and structural components were the main components with the highest estimation errors. The cost of carbon fiber materials was estimated based on the price per ft$^2$. Several items will need to be added into the overall cost as miscellaneous hardware parts, total electrical system wiring, and cooling systems. These additional items will continue to increase the total cost. Once the final kit cost is determined, the estimates for production quantities will be calculated. Variables such as manufacturing costs and labors will need to be added. Generalized airframe material and equipment costs can be comparatively estimated for non-pressurized aircraft using general aviation costing trends and graphs.
### Table 6: Weight and Cost Breakdown

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight (lbs)</th>
<th>Cost ($)</th>
<th># of Parts</th>
<th>Total Weight (lbs)</th>
<th>Total Cost ($)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Propeller</td>
<td>15</td>
<td>2500</td>
<td>1</td>
<td>15</td>
<td>2500</td>
</tr>
<tr>
<td>Electric Engine (80HP)</td>
<td>42</td>
<td>8200</td>
<td>1</td>
<td>42</td>
<td>8200</td>
</tr>
<tr>
<td>Battery Pack (Li ion)</td>
<td>191</td>
<td>900</td>
<td>2</td>
<td>382</td>
<td>1800</td>
</tr>
<tr>
<td>Power Block 40 Controller</td>
<td>15.5</td>
<td>1789</td>
<td>1</td>
<td>15.5</td>
<td>1789</td>
</tr>
<tr>
<td>E Charger</td>
<td>15.4</td>
<td>900</td>
<td>1</td>
<td>15.4</td>
<td>900</td>
</tr>
<tr>
<td>Electric Wiring System</td>
<td>1.9</td>
<td>400</td>
<td>4</td>
<td>7.6</td>
<td>1600</td>
</tr>
<tr>
<td>Engine Cowling</td>
<td>20</td>
<td>1300</td>
<td>1</td>
<td>20</td>
<td>1300</td>
</tr>
<tr>
<td>Spinner Cover</td>
<td>1</td>
<td>340</td>
<td>1</td>
<td>1</td>
<td>340</td>
</tr>
<tr>
<td><strong>Section Total:</strong></td>
<td><strong>498.5</strong></td>
<td></td>
<td></td>
<td></td>
<td><strong>18429</strong></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight (lbs)</th>
<th>Cost ($)</th>
<th># of Parts</th>
<th>Total Weight (lbs)</th>
<th>Total Cost ($)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aircraft Seats</td>
<td>20</td>
<td>100</td>
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<td>200</td>
</tr>
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<td>Instrument Panel</td>
<td>19.5</td>
<td>6138</td>
<td>1</td>
<td>19.5</td>
<td>6138</td>
</tr>
<tr>
<td>Flight Controls</td>
<td>2</td>
<td>140</td>
<td>2</td>
<td>4</td>
<td>280</td>
</tr>
<tr>
<td>Canopy</td>
<td>10</td>
<td>1469</td>
<td>1</td>
<td>10</td>
<td>1469</td>
</tr>
<tr>
<td>Restrain Belts</td>
<td>4</td>
<td>80</td>
<td>2</td>
<td>8</td>
<td>160</td>
</tr>
<tr>
<td>Rudder Pedals</td>
<td>2</td>
<td>229</td>
<td>2</td>
<td>4</td>
<td>458</td>
</tr>
<tr>
<td>Flight Control Linkages</td>
<td>25</td>
<td>1900</td>
<td>1</td>
<td>25</td>
<td>1900</td>
</tr>
<tr>
<td>Composite Skin/ Fuselage</td>
<td>45</td>
<td>2800</td>
<td>1</td>
<td>45</td>
<td>2800</td>
</tr>
<tr>
<td>Passengers</td>
<td>200</td>
<td>0</td>
<td>2</td>
<td>400</td>
<td>0</td>
</tr>
<tr>
<td><strong>Section Total:</strong></td>
<td><strong>555.5</strong></td>
<td></td>
<td></td>
<td></td>
<td><strong>13405</strong></td>
</tr>
</tbody>
</table>
Stability and Control

First the neutral point was calculated giving the Flyin’ Lion its aft center of gravity limit. Using Equation 29, the neutral point was found to be located at 0.46 aft of the chord from the leading edge.  

Next the forward limit center of gravity limit was determined. This limit is driven by the down force need to safely control the aircraft when it is flying in ground effect and $C_{L_{max}}$. Using Equation 30, the forward center of gravity was found to be 0.32 of the chord from the leading edge.
With the forward and aft limits of the center of gravity are determined, the interior layout of the Lion was determined. Laying out the major contributors to weight the center of gravity was determined and manipulated. Table 7 outlines the major contributors to weight as well as their positions in the aircraft. After several iterations using Equation 31 the layout in Figure 28 was determined to allow the center of gravity to fit the limits.

![Figure 28: Layout of Cabin and the C.G Locations of Weight Contributor’s](image)

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight of Part (lb)</th>
<th>CG location from nose (ft)</th>
<th>Moment (ft*lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Passengers</td>
<td>400</td>
<td>8.2</td>
<td>3280</td>
</tr>
<tr>
<td>Batteries</td>
<td>200</td>
<td>3.5</td>
<td>700</td>
</tr>
<tr>
<td>Batteries</td>
<td>200</td>
<td>12</td>
<td>2400</td>
</tr>
<tr>
<td>Motor</td>
<td>42</td>
<td>2</td>
<td>84</td>
</tr>
<tr>
<td>Controller</td>
<td>30</td>
<td>10</td>
<td>300</td>
</tr>
<tr>
<td>Propeller</td>
<td>15</td>
<td>0.5</td>
<td>7.5</td>
</tr>
<tr>
<td>Seats</td>
<td>40</td>
<td>8.2</td>
<td>328</td>
</tr>
<tr>
<td>Tail</td>
<td>60</td>
<td>20.5</td>
<td>1230</td>
</tr>
<tr>
<td>Fuse+Wing</td>
<td>313</td>
<td>7.5</td>
<td>2347.5</td>
</tr>
<tr>
<td><strong>Total Weight</strong></td>
<td><strong>1300</strong></td>
<td><strong>8.21</strong></td>
<td><strong>10677</strong></td>
</tr>
</tbody>
</table>
During the iterations to find a center of gravity it was noted that the passengers would be best placed on the aircraft center of gravity. This is because the Lion’s only variation in weight comes from passengers. Therefore, the passengers were placed very close to the center of gravity of the Flyin’ Lion. The center of gravity gives the Lion a static margin of 11% as shown in Table 8.

<table>
<thead>
<tr>
<th>Distance from Nose (ft)</th>
<th>Forward limit</th>
<th>Cg location</th>
<th>aft limit</th>
</tr>
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<tr>
<td></td>
<td>8.15</td>
<td>8.21</td>
<td>8.61</td>
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<tr>
<td>% Cord</td>
<td>0.33</td>
<td>0.35</td>
<td>0.46</td>
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</table>

**Table 8: C.G. Limits and Static Margin.**

**Design Philosophy**

While there are many successful Light Sport Aircraft in operation today, the market for an electric-powered LSA is in its infancy. This emerging industry promises a bright future in terms of low cost, fuel-efficient, and easy-to-fly recreational aircraft. An innovative new product such as this poses many technical challenges and comes with investment risks. To overcome these challenges and minimize risks, it is important to understand mistakes made in similar designs and capitalize on design elements that proved successful. This includes adding new technologies that would advance the performance of the final design. All of these considerations dictate the design philosophy.

Some of the design considerations for this aircraft were inspired from the parent aircraft. These include the Yuneec e430 and the Taurus Electro G2, depicted in Figures 7 and 8 respectively, among several other light electric powered aircraft. These airplanes provide a reference point when the answer to a solution is not obvious. For instance, the size and moment arm of the tail required for sufficient balance between stability and control was estimated by studying the schematics of similar airplanes. The idea for a high aspect ratio for the wing came from researching pictures and videos of parent aircraft. Since minimizing drag was a priority, a laminar flow airfoil was used on the wing of this aircraft significantly cut down on profile drag and a high aspect ratio was used to minimize induced drag. Also a high aspect ratio comes at a cost of high root bending moment, but since the structure is composite, these loads can be carried
effectively. The range and endurance depend on the type and size of batteries used, so batteries used in previous electric airplanes were investigated to find suitable matches. Conversely, many ideas for the aircraft were original. Several hours were spent in brainstorming sessions and individual research in order to improve on the design through innovative ideas, including instrument panel integrated with an iPad, and detachable and rechargeable batteries. Some of these ideas are discussed later.

Weights

The initial estimate on the gross takeoff weight of the aircraft was determined to be 1300 pounds. This weight was based on the weights of the batteries needed to obtain the desired performance, the weight of a pilot and flight trainer, and the limitations set forth on light sport aircraft determined by the FAA & FAR. The estimated weight of the batteries needed to power the airplane was 400 pounds. This weight was determined by comparing power and endurance needs with other electric powered vehicles and research conducted on current batteries. Because the aircraft is not powered using fuel, the “fuel” weight is constant throughout our range of operations. The passenger weight of 400 pounds was an estimation based on an above average weight of a person. The weight limitation on the light sport aircraft category, determined by the FAA, is 1,320 pounds gross takeoff weight.

Wing Configuration and Planform

A high wing configuration was chosen because it offered better visibility below the aircraft which is more beneficial for training. Another benefit of a high wing aircraft configuration is that the wing spar will not interfere with the instrument panel or the cabin space. The wing will have a high aspect ratio, a slight taper towards the outboard section, and blended wing-tips. High aspect ratio wings are essential for high glide ratios, increased range, and longer endurance. The downside to high aspect ratio wings is the large root bending moment, but because of the aircraft’s stronger composite construction, the airframe can support these increased loads. The planform of the wing will be mostly rectangular. This is to keep the construction costs of the wing down. One downside of the mostly rectangular planform is that the wing will have a less than elliptic lift distribution which translates to higher induced drag. To
fix this problem, the wing has linear taper, washout, and raked wing-tips to bring the lift distribution closer to elliptic. This also improves the stall characteristics of the wing. The aspect ratio and taper ratio were decreased by making the chord relatively short at 3.5 feet.

Figure 29: NLF-0115

The airfoil section being used for the Lion is the NLF-0115, shown in Figure 29. This airfoil was selected based on the size of the drag bucket, low profile drag, and it offered a $C_l$ range which matched our mission specifications. Natural Laminar Flow airfoils can maintain laminar flow in some cases up to 50% of the chord length, and reduce the dependence of flow separation with surface roughness. These airfoils were also designed for low speed, general aviation aircraft like the Flyin’ Lion.

**Fuselage and Cabin Design**

Considering the Lion is a trainer aircraft, a two person side-to-side configuration was determined to be appropriate. Having the student and the instructor sit side-by-side would make communication easier and more effective. It would also make demonstrating a maneuver or flight procedure an easier task for the instructor. The side-by-side configuration and battery placement influenced the design of the fuselage.

Although there are innumerable starting points when it comes to fuselage design, one could be to determine the cross-sectional shape. This is an obvious starting point because the
cross-sectional shape is primarily influenced by the required payload of the aircraft. The idea is to design the shape to accommodate passengers, crew, service equipment, flight electronics, and luggage. Minimizing empty space and making most of the available space is important in aircraft design because large parasite drag can translate into an increase in fuel consumption.

The two passengers in a side-by-side configuration made the design of the fuselage fairly simple. An aerodynamic shape was drawn around the electric motor, passengers, and batteries. The wetted area was reduced towards the aft region of the aircraft until reaching the tail region. This estimation should be accurate because there is no need for a space to retract landing gear or an area to store fuel.

Figure 30 shows the cockpit layout of the Lion.
Tail Design

A T-tail configuration was selected for the empennage of the aircraft. This design was decided primarily because of the stability the aircraft would gain from the benefits of a high tail. The high tail configuration reduces the downwash parameter of the tail, which is the amount of air directed downward on the tail from the main wing. A smaller downwash parameter results in a further aft neutral point. This is essential for our aircraft because the position of the batteries can be changed in a finalized design and the fact that aircraft can be flown with one or two passengers.

A T-tail design was also somewhat of an adoption of many glider designs. To give the Lion high glide characteristics and low drag the best place to look was at gliders. The fact that many gliders utilized T-tails was an assurance that it was a good design decision. The airfoil that was selected for both the horizontal and vertical stabilizers was the Eppler 521 depicted in Figure 31, which was deemed to be the optimum choice for the mission of the Lion.

![Figure 31: Horizontal & Vertical tail Airfoil](image)

The airfoil section for the horizontal tail was decided to be a symmetric airfoil at a positive tail incidence angle. The positive tail incidence angle of the airfoil will produce the down-force on the tail needed at the wing-body’s zero angle of attack. This will also help with the stability of the airplane. For the elevator sizing, the lion has been fitted with an all-moving elevator. During the stability and control calculations the elevator effectiveness factor became extremely
important in the calculation of the forward center of gravity. In order for the lion to be able to be controllable at max $C_L$ and in ground affect, the effectiveness of the elevator had to be 1.

**Batteries and Electric Motor System**

The primary objective of the battery system was to supply 2 hours mission time for the designated flight envelope and design speeds. The largest power draw during flight will be during takeoff and climb. The minimal power setting will be during descent. In order to choose a proper battery system, various characteristics and trends in the development of battery technology were considered:

1. **Engine Output**: 50-65 kW ideal engine power for the light sport aircraft of similar size. This figure is expected to rise significantly with improving battery technology over the next few years.

2. **Type of battery system** - The most common battery types are Lithium Ion, Lithium Polymer, or Lithium – “Metal Air Batteries.”

3. **Battery system weight** - High energy density (Whr/kg delivered) batteries are preferred. Three to four year projections suggest that energy density will be increased, resulting in an increase in power output of 10%.

4. **Type of Motor** – Out-runner Brushless DC Electric Motors were considered for convenience of propeller placement, motor cooling, higher efficiency, decreased size, and decreased weight (weight to power ratio). The disadvantage is that it costs more.

5. **Battery Management System (Controller)** - Required to control the energy flow and distribution to and from the batteries to ensure that the system runs efficiently and does not completely drain the battery system.

The battery system will be the power source for the entire aircraft and will be important to achieve the desired design specifications. Also, all navigation, communication, and aircraft indicators will be powered from the battery system. These secondary systems will create a power draw on the batteries. To accommodate these additional power draws, the battery system will
need to have the maximum excess power. The excess power will allow for additional take-off and climb demands as well as the proper operation of the secondary systems.

The type of battery system selected determines the type of controller system necessary, the cooling requirements, the max power output, as well as the cost of the overall system. The system for our LSA will be a balanced integration of the optimal characteristics using the projected trends available in a few years. The primary battery system inserted in aircraft was the Lithium Ion battery due to the high energy density, projected to be around 240 kWh/kg. This allows for the highest power-to-weight ratio. The weight of the battery system is expected to be around 20-25% of the entire aircraft weight, since the aircraft is primarily composite materials.

Another component of the battery system is the battery management system controller. The controller can be expected to add an additional 20-40 pounds, but is necessary to ensure the proper distribution of power from the battery system. An efficient controller, though it may have higher costs, will allow for a longer flight time and help to create prolonged battery life. The controller directly monitors the out-runner brushless DC electric motor that was selected for the design. The RPM’s are directly related to the controller as well as the entire aircraft electrical system.

The Yuneec Power Drive 60 out-runner brushless DC electric motor was selected for the design because of the amount of benefits compared to the disadvantages. The Power Drive uses a 60kW motor to provide enough power and thrust to meet the aircraft mission specifications. The location of the stator as the main mounting hub, allowed for easy propeller set up. Also, the location allows for easy set-up of air ducts to help cool the motor. The efficiency of the brushless vs. brush engine was notably higher and was worth the higher cost. As future trends progress, improvements in the electromagnetics of the brushless system will help reduce the costs. Future trends will also continue to increase the power-to-weight ratio and decrease the size.

The most important factor of the entire batter system was to have the best power-to-weight ratio. This will allow for the largest excess power in the system. The advancement in the battery and management systems helps to further increase the power output of the system. Each component of the battery system was selected based on the weight contribution to the entire aircraft. A benefit of the electric system is that the weight will remain constant during flight.
Innovations in the Design

The idea to incorporate an iPad into the instrument panel may sound like a gimmick or a novelty at first, but the power of third party software developers should not be underestimated. There are iPad applications, otherwise known as ‘apps’, targeted towards aviation enthusiasts. Some of these apps such as ‘Foreflight: Intelligent Apps for Pilots’ supplement already existing Avionics. Some of the features in this particular app include realtime, terrain-based maps, radar, satellite imagery, winds, ceilings, flight rules, temperatures, forecasts and flight planning. This could be an invaluable tool for an instructor allowing him/her to teach a student about safe flying techniques. Another advantage of using an iPad on board is that it is a light and compact electronic that comes with its own batteries. For particularly bright conditions, the iPad can be fit with an anti-glare screen. Even with modest use, iPad 2 has a battery life of up to 10 hours, sufficient for an entire day’s training sessions.

It is no secret that carbon composite structures are increasingly being used in aerospace applications these days and for good reason. The high strength-to-weight ratio and excellent rigidity characteristics make it an appealing material to use on airframes. Although it is more expensive than aluminum, carbon fiber-reinforced polymer materials offset some of the weight issues that arise with the use of relatively heavy batteries in the case of an electric LSA. Another advantage is that the high strength and low weight allows the aircraft to have better performance characteristics. It is important for this trainer aircraft to be able to perform flight maneuvers safely and effectively.

Focus of the Design

The focus of this project was always directed to coming up with a better design while keeping mission requirements and cost in mind. For instance, this aircraft does not have to perform acrobatics or combat; therefore it does not require any of those performance characteristics. However, it does need to accommodate two people, fly using an efficient electric engine, and do so while staying within the legal parameters of what is considered a “light sport aircraft.” Balancing these requirements and restrictions and going through several iterations, an
interim design was determined. The proposed aircraft has mostly simple features with innovative ideas and excellent performance characteristics, making it ideal for training purposes. It is simple so beginners can operate it without a huge learning curve. Most importantly, it is electric-powered so it stays ahead of the competition that is inevitable in the future of aviation.
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Battery Configuration

Lithium Ion Batteries

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Eco Friendly Electric LSA
http://www.ecofriend.com/entry/2010-zero-emission-electric-aircrafts/

Aircraft Performance Specs and Runway Take Off and Landing Info
http://www.free-online-private-pilot-ground-school.com/aircraft_performance.html

T-Tail Figure

NACA 0009 Horizontal Tail Airfoil

Piper J-3 Cub Picture
http://altairva-fs.com/fleet/ava_fleet_piperj3cub.htm
Yuneec e430 Picture
http://goodcleantech.pcmag.com/aviation/279285-yuneec-e430-electric-light-sport-aircraft\n
Pipistrel Taurus Electro G2 Picture

Electric Power Formula’s for UAV (Used for modified Range and Power Output Equations)
http://contentdm.lib.byu.edu/ETD/image/etd1223.pdf

Sample Mission Picture:
http://history.nasa.gov/SP-4404/ch8-3.htm

Component Pricing Main Website

"Airframe Parts from Aircraft Spruce." *Pilot Supplies, Avionics, and Homebuilt Aircraft Parts*


FAA General Aircraft Costings


Carbon Fiber Estimation Costs

"Carbon Fiber Manufacturing - 4'x8'x.050 carbon fiber flat sheet (Powered by CubeCart)."


Yuneec Motor and E Charger


Engine Costs


Pipistrel Virus Kit Pricing Site


General Manufacturing Costs


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<th>Referenced Text</th>
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# Appendix I: Aircraft Specifications

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<tr>
<td>Powerplant @ 2,400 RPM Yuneec</td>
<td>81 hp / 60 kW</td>
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<td>Power Drive 60</td>
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<td>(Thrust/Weight) T.O.</td>
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<td>Range</td>
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Appendix II: Panel Method for Windows

In order to properly obtain a span efficiency for the wing, the use of panel methods for windows was used. The method uses coefficients such as the non-dimensional airfoil coordinates, leading edge positions, aspect ratio, chord length, and angle of attack for the semi-span of the wing. The hybrid coefficient result was obtained by using the wake influence and relaxation over 24 different filaments a span efficiency value of 0.9606. Refer to Figures A1 and A2 for a visual representation of Panel Method for Windows.

Figure A1: Top View Panel Method for Windows
Figure A2: Back View Panel Method for Windows

Appendix III: X-Foil

![Graph](image-url)
Appendix IV: Aileron Roll Rate Calculations

% This Code will be used to determine the Aileron Size of the Flyin' Lion

% Constants

PI = 3.14159;
d2r = PI/180;   % Degrees 2 radians
r2d = 180/PI;   % Radians 2 degrees
chord = 3.5;    % Feet
V = 169;        % Ft/sec (Velocity)
b = 46;         % Span in Feet
rho = .002048;  % Density @ 5000 ft ( Slugs/ ft^3)
a = 1.25;       % Per Radians (Lift Slope)
tau = .6;       % Flap efficiency factor
y1 = 20;        % Outboard edge of the aileron from mid-span (Ft)
y0 = 15;        % Inboard edge of aileron from mid-span (Ft)
del_a = 20;     % Aileron Deflection (degrees)
semi_span = 23; % Ft
S = 156;        % Area of wing (Ft^2)

% Calculations

L_del_a = 0.5*rho*V^2*a*tau*del_a*d2r*(0.5*y1^2-0.5*y0^2);
\[ C_{l\_del\_a} = \frac{L_{del\_a}}{(S*b*0.5*rho*V^2*del\_a*d2r)}; \]
\[ Clp = \frac{(2*a*chord*(semi\_span^3))}{(3*S*b*V)}; \]
% Roll Rate Calculation
\[ p = \frac{C_{l\_del\_a}*del\_a*d2r*V}{(semi\_span*Clp)}; \]
\[ p = p*r2d \]