Development of Long Endurance Small Unmanned Aerial System

Andreas Quainoo

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MAE Senior Design Project Final Report

Development of Long Endurance Small Unmanned Aerial System

Western Michigan University
Department of Mechanical and Aerospace Engineering

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Shane Russell
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Advisor : Dr. K. Ro
Abstract

The purpose of this Senior Design Project is to design, build, and test an Unmanned Aerial System (UAS). The purpose of this system is to be used as a test platform for airborne electronics. The intended capabilities of this project include a 12-hour flight time, the ability to carry 5lbs of payload, and the ability to recharge the batteries of electronics while in the air. With these parameters in mind, the focus of the design was on endurance, and, by extension, efficiency and durability. After design was completed, the aircraft and its onboard systems was constructed and evaluated.
DISCLAIMER

This project report was written by students at Western Michigan University to fulfill an engineering curriculum requirement. Western Michigan University makes no representation that the material contained in this report is error-free or complete in all respects. Persons or organizations that choose to use this material do so at their own risk.
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Introduction

Background

Unmanned Aircraft Systems (UAS) have been in development for a long time. As they continue to improve and more uses are found for them, the more prevalent they become to society. Recently, there has been interest in developing systems and/or hardware for use in these small aircraft. However, testing such systems for any length of time can prove difficult, be it from not having access to a plane to install said systems into or having access but being limited on what can be tested by the plane. A better platform for such testing was required. The purpose of this project was to design, build, and test such a platform.

Goals and Objectives

There were three specific goals for this project. They were as follows:

- Capability of at least 12 uninterrupted hours of flight
- Ability to carry a minimum of 5 pounds of payload
- Ability to maintain batteries and onboard electronics while in flight.

Design Overview and Development Process

The design process can be broken down into the following steps:

- Benchmarking and Decisions
- Engine Testing and Propeller Selection
- Charging System Design and Sizing
- Airframe and Avionics Design
- Full System Ground Testing
- Final Flight Testing

Each of these steps was critical in the completion of the project. Each of these steps will be examined in detail in the following sections.

Benchmarking and Decisions

Engine Benchmarking

Various engine possibilities were examined. However, due to the cost of glow fuel, the decision was between a 4-stroke gasoline engine and a 2-stroke gasoline engine. While the 2-
stroke engine would have produced more power, the 4-stroke engine was ultimately selected for its efficiency. For more specific results, see the Engine Testing and Propeller Selection section.

Propeller Benchmarking

A variety of propellers (varying in both diameter and pitch) were initially considered. Each propeller was tested on the engine, and specific fuel consumption was determined. The propeller with the best specific fuel consumption (e.g. the most efficient per unit of power) and an acceptable level of thrust was selected.

Charging System Benchmarking

The decision to build a charging system, aided by the decision matrix below, was made comparing one that could be made against one pre-made by Sullivan (The specific model was the Sullivan Genesys). This particular model consists of an alternator and a charging circuit, which is designed to charge a 4.8VDC or, with additional modification and expense, a 13.5VDC battery. The alternator is a ring-type one that surrounds the output shaft of the engine, making a direct connection to the output shaft easy to accomplish. However, power output of this system is limited: 5 watts with one alternator and 10 watts if two are purchased (sullivanproducts.com/GenesysContent.htm, 2008). This power output would be acceptable if this platform were not meant to carry an electronic payload: This product is perfectly capable of recharging batteries used only for ignition and radio transmission. However, with the additional onboard electronics, more power output is desired. Additionally, the cost of such a system is high: ranging from $200 to upwards of $300.
### Decision Matrix

#### Engine Selection

<table>
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<tr>
<th>Engine Type</th>
<th>Cost (30)</th>
<th>Power (20)</th>
<th>Efficiency (30)</th>
<th>Size (20)</th>
<th>Total (100)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glow</td>
<td>14</td>
<td>5</td>
<td>4</td>
<td>6</td>
<td>24</td>
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<tr>
<td>Two stroke</td>
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<td>8</td>
<td>1</td>
<td>7</td>
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<tr>
<td>Four stroke</td>
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<td>7</td>
<td>25</td>
<td>7</td>
<td>51</td>
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</table>

#### Charging System

<table>
<thead>
<tr>
<th>Type</th>
<th>Development time/Testing (30)</th>
<th>Power Output (40)</th>
<th>Cost (30)</th>
<th>Total (100)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pre-made</td>
<td>20</td>
<td>10</td>
<td>10</td>
<td>40</td>
</tr>
<tr>
<td>Build one</td>
<td>10</td>
<td>30</td>
<td>20</td>
<td>60</td>
</tr>
</tbody>
</table>
Engine Testing and Propeller Selection

Load Cell Calibration

One of the load cells used was the ESP-6 beam type load cell (Figure 1) with a capacity of 13.2 lb. to measure torque with a load distance of 0.375 ft. The other two load cells were the MLP series load cells that were used to measure fuel weight and thrust. The fuel weight load cell had a capacity of 10 lb while the thrust load cell had a 25 lb capacity.

The load cells were connected to LabVIEW through the NI-USB 6221 DAQ card. Known weights were applied to the load cell and the voltage recorded. The voltage and weights were tabulated and an equation for the relationship between load applied and voltage measured was obtained using Microsoft Excel. The figures below show the relationship between the voltage in volts (V) and the load in pounds (lb) applied. As shown in the graphs below, the relationship between the load applied and the voltage measured is an almost exact linear relationship with a correlation of regression of at least 0.9995.

![Fig. 1 a) ESP-6 beam type load cell b) MLP series load cell](image)

*Figure 1: ESP-6 Beam Type Load Cell and MLP Series Load Cell*
Figure 2: Load cell calibration curves
LabVIEW Development

The goal was to develop a LabVIEW VI that could measure the voltage of each of the load cells (torque, thrust and fuel weight) and convert it to a load in pounds, measure RPM and control the servo linked to the throttle lever on the engine. The VI should also have the capability of maintain a constant engine RPM by adjusting the throttle. The figure below shows the front panel of the VI.

![Front panel of LabVIEW VI](image)

*Figure 3: Control panel of LabVIEW VI*

The upper left corner had the sampling rate, raw voltage form each of the load cells and a stop button to halt execution of the VI. The upper middle section had a graph with tabs showing the torque, thrust, rpm, fuel weight and RPM signal as a function of time. The upper right section had the processed values, i.e. thrust, torque, power, Specific Fuel Consumption (SFC), average fuel flow rate and current fuel weight. The lower sections from the left had the proportional and differential gains and incoming wind speed, timing, switches for throttle or RPM control, an RPM gauge and the RPM controller. The equations used to calculate these parameters are in Table 1. Figure 4 shows the block diagram. Due to the complexity of the VI, the specific details of the block diagram are illegible. A legible copy is available upon request.
Before the engine could be tested in the wind tunnel, the system had to be proven to work. Static runs were performed and improvements made continually until the system was reliable and the testing procedure was proven. Because of the low fuel consumption of the engine, the most accurate way to measure the fuel flow rate was to measure the change in mass of the fuel with time. Even this proved to be difficult due to the sensitivity of the reading being taken. Initially the fuel tank was mounted behind the engine, but the vibrations from the engine prevented any form of accurate readings to be made since the fuel was sloshing within the tank. It was imperative to develop a dampening system and stand to prevent displacement of the fuel tank and sloshing of fuel in the tank. The first step was to build a separate stand for the fuel tank, but because it was attached to the stand by cables, it was still in the prop wash. A shield was then made to isolate the fuel tank from the prop wake. Once the LabVIEW program and testing apparatus was verified, wind tunnel testing began.
Figure 5: Engine stand development phases
Wind Tunnel Testing

Wind tunnel testing was done at the Applied Aerodynamics Lab, located in the Kalamazoo Airport, using the Advance Design tunnel. An exhaust system was used to duct the fumes from the engine out of the wind tunnel and the lab since the wind tunnel was closed loop. It took some time to set up the stand in the tunnel. The fuel tank was kept out of the tunnel to ensure it was not disturbed.

The drag of the stand in the wind tunnel was measured by reversing the thrust load cell and setting the speed of the wind tunnel at 40 and 50 mph because those were our test speeds. We choose those speeds because that was our estimated cruise speed. During testing we noticed the thrust dropped exponentially with increasing wind speed so 50 mph was the upper limit. The drag was about 1.8 and 3 lb. at 40 and 50 mph respectively. Various propellers (see table below) were tested to at different propeller RPMs. The thrust, power and specific fuel consumption were measured and further analysis was carried out.
Summary of Data

<table>
<thead>
<tr>
<th>Diameter (in)</th>
<th>RPM</th>
<th>Adjusted Thrust (Lb)</th>
<th>Power (HP)</th>
<th>m_bar (g/s)</th>
<th>SFC (/hr)</th>
<th>Fuel for 12 hrs (lb)</th>
<th>Oz for 12hr (oz)</th>
<th>L/D Req</th>
</tr>
</thead>
<tbody>
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<td>1.626</td>
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*Table 2: Summary of performance parameters at 40 mph*
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<th>Diameter (in)</th>
<th>RPM</th>
<th>Adjusted Thrust (Lb)</th>
<th>Power (HP)</th>
<th>m_bar (g/s)</th>
<th>SFC (/hr)</th>
<th>Fuel for 12 hrs (lb)</th>
<th>Oz for 12hr (oz)</th>
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</tr>
</tbody>
</table>

Table 3: Summary of performance parameters at 50 mph

![SFC vs. Engine RPM for various props at 40 mph](image)

Figure 8. SFC vs. engine RPM for various props at 40 mph
Figures 8 and 9 above show the variation of SFC with engine RPM for various propellers. The 16x10, 17x10 and 17x8 have the lowest SFC at 40 and 50 mph. The 17x10 was chosen because it had the best compromise of thrust, SFC and engine RPM.
Charging System Design

Overview

Based on benchmarking and the decision matrix, it was decided that the charging system would be designed and built. Developing a charging system allowed for a higher output, more flexibility in the design, and a lower cost. The basic principle of the charging system operation is to power an electric motor via the engine. Doing this enabled the motor to be used as an alternator. There were two critical factors that governed this design: The minimum speed at which the alternator has to turn in order to produce the desired power output and the method of connecting the alternator to the propeller shaft of the engine. Both of these factors had to be determined via an empirical approach.

The voltage and current output of the alternator was tested at different speeds to build a map of power output. Testing was accomplished by turning the alternator at known speeds with varied loads. This information was graphed, normalized, and extrapolated to determine the optimal speed of the alternator to produce the desired power output. The goal of this design was to have the capacity to produce 20 Watts of useable power from the alternator.

The current produced by the alternator is AC. As all of the on-board electronics require DC power, conversion was required. This is accomplished with a bridge rectifier, where the incoming AC current is converted to rough DC current. Because the voltage produced by the alternator varies with engine speed, a DC-DC converter was employed to regulate the output voltage of the charging system. The alternator was overdriven off of the engine, at a ratio that ensured the required voltage from the alternator was produced when the engine was running at its steady-state cruise speed. From the converter, the current branched out to maintain all electrical systems.

Alternator Selection and Testing

Due to its ready availability, relative low cost, and broad options, the decision was made to use an “out runner” electric motor as an alternator. A sample of this motor is shown in Figure 10. This motor, which is normally used on electric radio controlled airplanes, is a brushless 3-phase motor, normally driven by a frequency modulator controller. However, it was determined that a motor such as this, when mechanically driven, could function as an alternator. The current produced by this motor is, conversely, a 3-phase AC current, which required conditioning before being useful to the charging system.
Before testing was started, research into creating steady DC current from a 3-phase source was done. An AC signal can be converted to a DC signal using a bridge rectifier. A bridge rectifier uses a series of diodes arranged in a bridge pattern to convert the sinusoidal voltage of an AC signal to the steady voltage of a DC signal (Zekavat, 2013). An example of this AC signal is shown plotted in Figure 11. This signal is what the 3-phase alternator produces. As can be seen,
there are three distinct waveforms on this graph. Each of these waveforms represents the voltage of each “phase”. These waveforms were then filtered through a 3-phase bridge rectifier, a schematic of which is shown in Figure 12, where it is converted to a raw DC signal. A graph of this raw DC signal is shown in Figure 13.

Figure 11: 3-Pase Voltage vs time (Graph made using LTSPICE)

Figure 12: Bridge Rectifier Circuit Schematic (Made using LT Spice)
The purpose of the capacitor in the circuit shown in Figure 12 is to smooth out the voltage ripple left over from the bridge rectifier. The capacitor achieves this by charging when the voltage ripple is high and discharging when the voltage ripple is low, thus filling in the “valleys” shown in Figure 13. This produces a smoother DC waveform. However, while the capacitor is very effective at producing a higher quality DC voltage, it is less effective when a load is applied to the output of the rectifier (such as a battery in need of charge). Using LT Spice, it appeared as if the electrical load would draw the charge stored by the capacitor, negating its effectiveness at smoothing the DC ripple. However, as the DC current was being deposited into batteries, rather than delicate electronics, and the batteries did not appear to be affected by this ripple, a capacitor was deemed not strictly necessary. Additionally, the DC-DC voltage converters afforded some level of ripple smoothing.

In order for the charging system to consistently put out voltage, the alternator must turn at a minimum speed. This minimum speed was determined through experimentation. It was initially thought that the kV value of the electric motor could be used in reverse to calculate the minimum RPM. However, after spinning the shaft of the motor, it was found that the kV value was not the same going in reverse (e.g. turning the motor at a given RPM would not produce an easily predictable voltage, as putting a given voltage to the motor produces a known RPM). Furthermore, adding an electrical load to the system causes the voltage to drop. To determine the minimum RPM needed to power the charging system, a series of tests were done. These tests
involved turning the motor at various known speeds and electrical loads and recording the output voltage and amperage. 4 Trials were performed: the first trial with open terminals, and the remaining three with loads varying from 10 ohms to 35 ohms.

Figure 14 below shows the apparatus used to perform these tests. The motor (which from this point forward will be referred to as the alternator) was turned using a drill press and a belt, as shown in Figure 15. To increase the speed of the alternator, a relatively large pulley was fitted to the drill press. To measure shaft speed, the bell of the alternator was covered with white paper and a black strip was marked on this paper. A non-contact tachometer (not pictured) was then used to directly measure RPM.
As can be seen in Figure 14, the 3 leads from the alternator were attached to the 3-phase bridge rectifier. Output from the rectifier was then attached to the load. The load, again as Figure 14 shows, was created using standard 60 Watt incandescent light bulbs. These light bulbs were used because they provided a very low resistance and could allow a large amount of current to pass through. Multiple light bulbs were used to vary the electrical load. This was accomplished by arranging them electrically in series and parallel. However, because the light bulbs heat up a great deal when current is passed through them, their resistance could not accurately be measured. Therefore, two electrical measurements were recorded: Voltage across the output of the bridge rectifier and current passing though the electrical load. Using a basic equation (Power equals Voltage multiplied by Current), power could then be calculated. See Appendix I for these results.

One issue with these trials was that it was not possible to turn the motor at a given RPM for each loading. As the electrical load increases, it becomes increasingly harder to turn the shaft of the motor. To get useful information from these trials, tables of RPM vs voltage and RPM vs power were made for each trial. From these tables, graphs of RPM vs voltage and RPM vs power were built. They can be found in Appendix II. As can be seen, best fit curves were plotted over
each of these graphs, and the equations from each of these curves were used to plot RPM vs voltage and RPM vs power from 1500RPM to 19000RPM in even increments of 500RPM. As the maximum RPM achievable by the testing equipment was roughly 6000RPM, the higher RPM values were extrapolated. See Appendix III for these corrected results. This data was used to correlate these three factors (RPM, voltage, and power), onto one graph. This graph can be found in Figure 16 on the following page. For a given RPM, there is a line, made from 4 points (one from each trial). This line correlates this RPM value to voltage and power output.

Note that the graph shows power outputs over 250W. Power generation this high was not tested: These are purely extrapolated values. Due to the inability to test the motor at higher RPM’s, the power outputs shown on this graph above 50W are for reference only. The required minimum RPM can be determined by looking at the graph in Figure 16. Because 100W is the desired maximum power output, and the input voltage required to charge the accessory battery is 14V, it can be seen that roughly 8800RPM is the minimum needed alternator speed. If the engine turns at 5500RPM in steady-state, the ratio of alternator speed to engine speed is roughly 1.6:1.
Figure 16: Extrapolated Power vs Voltage at a Given Alternator RPM
Charging Circuit Design and System Integration

Figure 17 below is a diagram of the charging system onboard. It shows how the system is currently installed. The 3-phase rectifier and the main DC-DC voltage regulator were selected to withstand at least 100W of power transmission. The diode present at the regulator is there to prevent current flow in the opposite direction, which could potentially cause damage to the circuit. Initially, the plan called for 3 separate batteries, each at their respective voltages for...
ignition, receiver/autopilot, and accessories. While the current design does have 3 batteries, the setup for the accessories battery is different than what was first planned.

Initially, the plans called for using a relatively low capacity battery with a simple chemistry (such as a lead-acid or a Li-Fe battery) that would receive the charge from the main DC-DC voltage regulator and would directly power all accessories. The purpose of the simple battery chemistry was to simplify charging, the idea being that a steady, elevated input voltage would be enough to properly maintain the battery (eliminating the need for a more complex circuit, as would be required for a Li-Po battery). The battery would serve, more or less, as a repository for excess power produced by the alternator, and allow systems to be briefly powered up without the engine running. However, if the alternator were to fail for any reason mid-flight, this low-capacity battery would not be able to power accessories for any appreciable period of time. Additionally, a better battery solution was found: using a portable power supply intended for jump-starting cars.

Specifically, the one initially selected for this project was the Antigravity Batteries Micro-Start XP-1. Figure 18 below shows a picture of this battery pack, along with labels of several of its features. This battery has a 44W-hr capacity and weighs less than 400g. It had 3 useable

![Antigravity Batteries Micro Start XP1: Micro Start Details](Micro Start XP 1, 2014)
outputs: 19V at 3.5A, 5v at 2A, and 12V at 3A (Micro Start XP 1, 2014). These outputs would have allowed different accessories be powered at the same time, potentially without the need for additional voltage regulation. This battery was a Li-Po battery, which normally requires a special balancing charger. However, this pack its own built-in charging circuit; all that was required to maintain voltage was to apply voltage across its charging port. Through experimentation, it was discovered that a minimum load had to be placed on the pack in order for it to activate. Therefore, to keep the pack active (thus guaranteeing its output of voltage), a constant electrical load had to be applied. This was accomplished via the interior lighting, which ran off of the 5V output. This was the primary purpose of these lights, which are shown in the wiring schematic found in Figure 17. Additionally, there was a restriction of the inlet. The maximum charging current was 12V at 2A, meaning the accessory battery pack could only take in 24W of power. Because of this, the DC input was additionally connected to the receiver and ignition battery packs, each of which had their own respective voltages.

This pack was installed on the plane, and was used for initial testing. For a time it worked quite well, and could in fact be recharged via the charging system. However, at one point the battery pack was rapidly discharged during use. This rapid discharge damaged both the batteries and the built-in charging circuit. Upon inspection, the Li-Po battery pack was found to be permanently damaged, and the charging circuit was no longer functional. Therefore, it was concluded that this battery system was not robust enough for either the continued charging and discharging cycles it would see whilst the aircraft was flying, nor the large discharges required of it to power all required devices. Therefore, a replacement pack was not ordered.

Instead, more traditional (for model RC planes) batteries were substituted: Li-Fe (Lithium Iron) battery packs were fitted to the aircraft to serve as accessory batteries. These batteries were chosen due to their increased capacity over the old battery system, as well as their comparatively simple battery chemistry and ability to handle large current discharges; these batteries could be charged with a trickle-type battery charger, which means the output from the charging system could directly be applied to these batteries to maintain them. No special circuitry was required to maintain pack voltages. In the brief taxi testing done with the charging system, it was found that these batteries worked quite well for this application.
Alternator Mount Design

Over the course of development, the alternator mount underwent 3 main design changes. The main challenge was finding a reliable method of connecting the alternator to the engine that could withstand very high speeds. The first design attempted to utilize a pulley and cogged belt. When this did not work as desired, a second design was employed, which involved using a steel drive gear mounted on the engine output shaft and a plastic driven gear mounted on the alternator shaft. This design was replaced with a third iteration, which substituted the plastic driven gear for a steel driven gear. Each of these designs will be discussed in detail.

Design 1: Pulley Connection

The first design involved using a set of cogged pulleys to drive the alternator. Figure 19 shows the pulleys and the cogged belt used to connect them. The cogged belt is made of rubber bonded to a series of Kevlar cords. The larger pulley was attached to the engine and the smaller pulley was attached to the alternator. Due to their sizes, a speed ratio of 1.66:1 was created. The mount utilized an idler pulley and an O-ring belt to maintain tension on the drive pulley. Figure 20 below shows the completed design mounted on the aircraft.

The purpose of the O-ring pulley was twofold: the main purpose was to maintain tension on the cogged drive belt. It was found that a good deal of tension was required to keep the belt from slipping when load was applied to the alternator. The second, and somewhat less critical, purpose of the O-ring was to provide the alternator shaft with an opposing force to counter the tension of the drive pulley. Initially, it was believed that the bearings in the alternator would not survive a substantial side load applied to the shaft. After continuous testing, it was found that this was not the case: Even with a single side load on its shaft, the alternator proved to be consistently reliable.
The issue with the alternator mount came when it was tested with the engine running. The engine imparted a much more significant amount of mechanical shock to the pulley than was expected. After 30 seconds, the Kevlar cords on the cogged belt were delaminated from the rubber, causing the belt to stretch out severely, rendering the charging system inoperable. Additionally, it was very difficult to provide constant tension on the belt via the O-ring belt, as the O-ring would stretch slightly as it heated up. Finally, the mount was constructed with a series of slots milled into multiple pieces. These slots were meant to aid in properly aligning the alternator to the engine. While they served this purpose, they also severely impacted the rigidity
of the mount: It proved to be impossible to keep the arm holding the tensioner pulley from sliding upwards towards the alternator, which also caused the drive belt to slacken and slip.

Additionally, the drive pulley, which was machined so it could be sandwiched between the hub of the engine output shaft and the base of the propeller, would slip. This slippage occurred between the hub of the output shaft and the face of the pulley, and was found to be caused by the fact that the metal the pulley was made of was too soft to gain purchase on the surface of the hub. There was one final problem with this system: The pulley attached to the alternator was attached with set screws that were tightened down onto its shaft. Under high rpm’s, these set screws would back out, causing the pulley to spin freely on the shaft. Several methods were tried to keep this from happening, but to no avail. For these reasons, a second alternator design was attempted.

**Design 2: Steel Gear on Plastic Gear Connection**

There were several problems with the first design: The shock loading from the engine was quite severe, which caused the belt to bounce. The mount was not rigid enough, and the connection between the driven gear and the alternator was not robust enough. To solve these problems, a new mount, utilizing direct gear drive, was built. A steel gear was used as the drive gear, which was attached to the hub on the output shaft of the engine, and a plastic gear was used as the driven gear. A plastic gear was selected so that there could be a known point of failure in this system: Should the alternator malfunction and stop rotating, the teeth of the plastic gear could shear off, thus ensuring the engine would not be stopped.

One initial concern with using a direct drive gear system was extreme gear wear caused by the shock of the engine. Additionally, there was concern about maintaining a tight enough tolerance to maintain an acceptable level of gear backlash as the engine ran. The first issue with this mount was the concern of gear wear. As it was expected that the plastic gears would wear, they were treated as consumables and several were ordered. However, the larger steel gear (the same 1.6:1 ratio between the alternator and the engine was maintained) could not wear. Therefore, a gear life analysis was performed to attempt to predict the life span of the gear under what was assumed to be worst-case shock and mounting conditions.

To simulate worst-case conditions, it was assumed that the gear would be transmitting a constant 100 Watts of power at 6000rpm, which is greater than the maximum power capable of
being generated by the charging system. The pitch line velocity of this gear was determined using the equation below.

\[ V_p = RPM \times \frac{1}{2} \times d_p \]

*Figure 21: Pitch Line Velocity (Juvinall & Marshek, 2012)*

Were dp is the pitch diameter of the drive gear, which was 2.598in (66mm). It was found that the pitch line velocity was 649.50 feet per minute. Knowing this, the tangential force applied to the tooth face of the gear can be found:

\[ F_t = \frac{Power}{V_p} \]

*Figure 22: Tangential Force Applied to Gear Tooth (Juvinall & Marshek, 2012)*

Ft was found to be 6.183lbf. With this value, the surface fatigue stress equation can be looked at:

\[ \sigma_H = \sqrt{\frac{F_t}{b \times d_p \times I} \times k_v \times k_o \times k_m} \]

*Figure 23: Gear Surface Fatigue Stress (Juvinall & Marshek, 2012)*

All values to solve for the surface fatigue stress can be found in the Fundamentals of Machine Design Textbook (Juvinall & Marshek, 2012). Table 4 below summarizes these terms:

<table>
<thead>
<tr>
<th>Var</th>
<th>Description</th>
<th>Value</th>
<th>Condition (if applicable)</th>
</tr>
</thead>
<tbody>
<tr>
<td>b</td>
<td>Gear Face Width</td>
<td>0.236in</td>
<td></td>
</tr>
<tr>
<td>I</td>
<td>Dimensionless Constant</td>
<td>0.102</td>
<td></td>
</tr>
<tr>
<td>K_v</td>
<td>Velocity Factor</td>
<td>1.541</td>
<td>Assuming gear was hobbed with shaping cutter</td>
</tr>
<tr>
<td>K_o</td>
<td>Overload Correction Factor</td>
<td>2.25</td>
<td>Assuming Most severe shock condition</td>
</tr>
<tr>
<td>K_m</td>
<td>Mounting Correction Factor</td>
<td>1.6</td>
<td>Assuming less rigid mountings, less accurate gears, and full face contact</td>
</tr>
</tbody>
</table>

*Table 4: Gear Surface Fatigue Stress Values*

With these values, the surface fatigue stress is 56542psi. The life factor of the gears can be determined using the equation below.

\[ S_H = S_{fe} \times C_{Li} \times C_R \]

*Table 5: Alternate Gear Surface Fatigue Stress (Juvinall & Marshek, 2012)*
The values used to solve for the life factor (CLi) can be found in the table below. Values are again found using the Fundamentals of Machine Design Textbook:

<table>
<thead>
<tr>
<th>Var</th>
<th>Description</th>
<th>Value</th>
<th>Condition (if applicable)</th>
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</thead>
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<tr>
<td>SH</td>
<td>Surface Fatigue Stress</td>
<td>56542psi</td>
<td></td>
</tr>
<tr>
<td>Sfe</td>
<td>Surface Fatigue Strength</td>
<td>5600psi^0.5</td>
<td>Assuming gear Brinell hardness (Bhn) of 165</td>
</tr>
<tr>
<td>CLi</td>
<td>Fatigue Life Factor</td>
<td>Unknown</td>
<td></td>
</tr>
<tr>
<td>CR</td>
<td>Reliability Factor</td>
<td>0.80</td>
<td>Assuming gear system to be 99.9% reliable</td>
</tr>
</tbody>
</table>

*Table 6: Alternate Gear Surface Fatigue Stress Values*

Using the above equation with the values in the above table, the Fatigue Life Factor was found to be 1.26. Using Figure 15.27 in the Fundamentals of Machine Design Textbook, which is a plot of Fatigue Life Factor vs Surface Fatigue life (in cycles), it was determined that the gear, under worst-case conditions, would last 550,000 cycles, which, at 6000rpm, correlates to 1.5 hours. With this information, it was deemed acceptable to run the gear on the engine and periodically monitor it for surface wear. Additionally, to remove unnecessary weight, the drive gear was shortened and a web was cut.

The alternator mount itself was simplified: to restrict unwanted movement while the system was in operation, the only adjustment permitted was the ability to slide the alternator up and down, which was required to set the backlash between the two gears. Figure 24 shows a picture of this mount. As can be seen, the driven gear is held onto the alternator shaft by 4 bolts. Initially, this gear was held in place by a hardened steel pin which was driven through the gear and the shaft. However, due to the severe shock imparted on the gear, this pin would break repeatedly and was found to be unreliable. Therefore, a more robust 4 bolt setup was used.

This setup proved to be far more reliable than the initial design, which again used a belt. The main issue was with gear alignment and backlash. The gear alignment problem was twofold: To begin there was great difficulty in maintaining perfect concentricity of each outer gear surface and its respectable shaft. This caused the alternator gear to “bounce” out of the drive gear at high rpm’s, which accelerated wear on the driven gear. This problem of concentricity stems from the fact that it was impossible to machine the inner bores of the gears to the exact diameters of their respective shafts: there was a very small amount of clearance, which was required to fit the gears onto the shafts. This problem was rectified by utilizing aluminum foil, which has a thickness of less than 0.003in (0.08mm).
The aluminum foil was cut into a strip the same width as the hub of the respective gear, and was wrapped around the shaft at the point where the gear would reside. The gear was then slipped onto the shaft and pressed onto the aluminum foil, which, because it was wrapped evenly and tightly around the shaft, centered the gear to the shaft. This method took several trials to perfect, but worked very well in reducing the concentric run-out of both the drive and the driven gears. This in turn meant that the clearance between the gears could be set tighter, reducing backlash and therefore impact. Also, higher engine rpm’s could be achieved before the alternator starting to “bounce”.

The second issue, the parallelness of the two shafts, was never fully rectified before the first and only test flight took place. It was reduced by taking care to align the gears up as best as possible, but it was never fully eliminated. This problem essentially caused the steel drive gear to rapidly wear down the plastic drive gear: 5 minutes would pass before the teeth on the plastic gear would strip off completely. Therefore, a third mount design was employed.
Design 3: Steel Gear on Steel Gear Connection

The third and final design iteration for the alternator mount is essentially the same as the second iteration, but substituting a steel gear for the plastic gear. The purpose of this was to decrease rapid gear wear with a harder material. Figure 25 shows the final execution of this gear. As can be seen, a web was cut into the gear and the same 4 bolt connection from before was again utilized for this mount.

![Figure 25: Alternator Mount Third Design](image)

Final Testing on Aircraft

Ultimately, flight testing with the charging system functional was never accomplished. A great deal of ground testing was done before flight took place, which exposed several additional issues with the final design. To begin, there was a great deal of gear whine, which was to be expected with the straight-cut gears used, especially when they were running without any kind of enclosure at high speed. Additionally, and more concerning, the alternator would bounce around a great deal when the engine was brought above 5000rpm. At higher rpm’s, the alternator gear would bounce out of the drive gear completely. Upon later inspection, a great deal of gear wear could be seen. An example of this gear wear is shown in Figure 26 below.
Figure 26: Alternator Gear Wear

Remember that the worst-case estimate for the drive gear wear was 1.5 hours. The physical results suggest that, had prolonged running been permitted, the gear teeth would have worn and fractured long before then. Therefore, the shock and meshing misalignment experienced were far greater than that which could be accounted for in the gear life analysis. It is believed that this rapid wear was caused primarily from gear misalignment.

While it may be possible to overcome the misalignment issue and make the alternator mount work, the design as a whole has been shown to be impractical. Also, in hindsight, it was perhaps not good practice to power the charging system off of the same engine that was
powering the propeller. A better method of powering the charging system is recommended for the future.

**Charging System Conclusions and Recommendations**

In conclusion, the charging system did not prove to be practical. The reason why was primarily due to the reliability of power being delivered to the alternator: The connection between the main engine and the alternator was the weak link in this system. The precision required for the gearing to function smoothly at high rpm was not possible; Between the flex present in the firewall and alternator mount, and the lack of better concentricity, the gearing experienced an excessive amount of shock, enough to cause visible damage to the steel teeth and raise concerns of reliability.

However, due to the long endurance requirement of the aircraft, a charging system is needed to maintain battery pack voltages. Therefore, a recommendation for powering the alternator will be made: Use a second engine to power the alternator and only the alternator. Additionally, size the kV value of the alternator such that this second engine can be directly coupled to it, so that overdriving is not required. To do this, a lower kV value motor would be required. Alternatively, a higher-revving engine could be employed. Either method would eliminate the need for any sort of gearing or pulley system, which would simplify the system and greatly improve the charging systems overall reliability. The downside of this recommendation is the additional requirement for fuel and the additional weight needed onboard, but it may be possible to mitigate these issues. The use of a very small engine (such as a 0.049 cubic centimeter engine) that has a very high maximum rpm could minimize the weight of the charging system and the added fuel consumption. For example, engines of the size mentioned that run on a mixture of nitro methane and oil are manufactured, are relatively inexpensive, and are readily available. Testing would have to be done to determine fuel consumption, but it may be possible to power one such engine for the required time period with less than one pound of fuel. Using a system such as this would give the charging system a much better chance of working. The rest of the charging system (the bridge rectifier and the use of DC-DC voltage converters) worked very well and could possibly be used again.
Aircraft Design

Configuration

Considerations

- Unobstructed forward view from nose of aircraft to allow for easy camera placement and addition future systems.
- Significant payload area
  - Flat floor to allow for easy system installation
    - Gimbal, Ground Scanning Radar, Thermal Camera’s etc.
- Statically Stable (Not a flying wing or tailless design)
- Traditional configuration to avoid unneeded complications
  - Ex: High wing, empennage, single engine, tricycle landing gear, etc.
- Uncomplicated Geometry
- Utilize existing Cessna 337 wing and tail section
  - Donated from previous design project

To stay within our existing considerations, we initially only needed to design a fuselage on which we would mount our main wing, empennage, and landing gear. This completed the majority of our aircraft structure, until the addition of our lower wing. The lower wing increased our wing area and reduced our max takeoff weight. It also made the stall speed a more reasonable value of 43 mph, down from 50 mph without our lower wing. It also allowed us to move our center of lift back helping increase our static margin allowing for a more stable aircraft.

Final Aircraft Configuration

- Biplane w/ Mid-Wing Struts
  - Upper Wing 80”
    - Outboard Ailerons, Inboard Flaps
  - Lower Wing 63”
  - Material: Balsa, Bass Wood, Light Birch Plywood
- Tricycle Landing Gear
  - Nose Wheel Steering
    - Material: Aluminum and Spring Steel
- Carbon Main Gear
- Inflated 3” Tires
- Dual Boom Vertical Stabilizers w/ Rudders
  - Material: Balsa, Bass Wood, Light Birch Plywood, Depron
- Blown Horizontal Stabilizer
  - Full Span Elevator
  - Material: Balsa, Bass Wood
- Pusher Propulsion Selection
  - Saito FG-21 4 Cycle Gas Engine
  - Zinger 15” x 8” Pusher Prop
- Streamlined fuselage
  - Rear Engine Component Bay
  - Mid Body Fuel Tank Bay
    - 180 oz Capacity (1.4 Gallons)
  - Mid Body Elevated Avionics Shelf
  - Forward Payload Bay
  - Nose Located Electronics Bay
  - Material: 1/8” to 3/8” Laminated Birch Plywood, Depron for Hatch

*Figure 27: Concept Design*  
*Figure 28: Final Airframe*
<table>
<thead>
<tr>
<th>Aircraft Configuration Specifications</th>
<th>Wing Span (Ft)</th>
<th>Wing Area (Ft²)</th>
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</thead>
<tbody>
<tr>
<td>Aspect Ratio</td>
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<td>9.38</td>
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<tr>
<td>Taper Ratio</td>
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<td>0.9126</td>
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<tr>
<td>Oswald Efficiency Factor</td>
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<td>~Cdo</td>
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<tr>
<td>Wetted Surface Area (Ft²)</td>
<td>0.0026</td>
<td>0.0103</td>
</tr>
<tr>
<td>Lift / Drag Max</td>
<td>17.92</td>
<td>2.3</td>
</tr>
<tr>
<td>Power Available (Hp)</td>
<td>2.3</td>
<td>0.75</td>
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<td>Propellant Efficiency</td>
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<td>Mtow Wing Loading (Lb/Ft²)</td>
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<td>2.41</td>
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<tr>
<td>Empty Wing Loading (Lb/Ft²)</td>
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<td>0.7</td>
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<tr>
<td>CI Max</td>
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</tr>
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</table>

Table 7: LEUAS Specifications

Figure 29: Various Pictures
Performance

Predicting the performance of any prototype aircraft is mandatory for the safe development and design of any air fairing vehicle. To achieve this critical goal, we applied our knowledge obtained in our Aircraft Performance class; A MATLAB code was written to evaluate the prototype aircraft. With only sixteen initial inputs, we were able to evaluate an aircraft, with reasonable accuracy, throughout an endurance mission profile. Refer to Appendix IV for the detailed MATLAB code.

16 Initial Conditions

1. Aircraft Weight 9. Wetted Surface Area
2. Altitude MSL 10. Current Temperature
4. Desired Cruising Speed 12. Root Cord
5. Static Thrust 13. Tip Cord
7. Propeller Efficiency 15. Min Fuel Volume (Gal)
8. Cl Max 16. Max Fuel Flow Rate

Figure 30: Initial Performance Conditions

<table>
<thead>
<tr>
<th>Take Off Weight</th>
<th>Empty</th>
<th>Flight Tested</th>
<th>Max</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight (Lbs)</td>
<td>22.6</td>
<td>24.6</td>
<td>30</td>
</tr>
<tr>
<td>Stall Speed (Mph)</td>
<td>37.57</td>
<td>39.2</td>
<td>43.94</td>
</tr>
<tr>
<td>Take Off Distance (Feet)</td>
<td>127.7</td>
<td>151.4</td>
<td>225.2</td>
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<tr>
<td>Endurance Cruise Speed (Mph)</td>
<td>39.24</td>
<td>40.91</td>
<td>45.18</td>
</tr>
<tr>
<td>Max Rate Of Climb (Ft/s)</td>
<td>24.79</td>
<td>22.31</td>
<td>17.2</td>
</tr>
<tr>
<td>Climbing Airspeed (Mph)</td>
<td>51.6</td>
<td>53.84</td>
<td>59.46</td>
</tr>
<tr>
<td>Comb Angle (Deg)</td>
<td>14.79</td>
<td>12.51</td>
<td>8.65</td>
</tr>
<tr>
<td>Endurance Range (Miles)</td>
<td>0</td>
<td>160</td>
<td>630</td>
</tr>
<tr>
<td>Fuel Volume (Gal)</td>
<td>0</td>
<td>.3</td>
<td>1.1</td>
</tr>
<tr>
<td>Time Aloft (Hours)</td>
<td>0</td>
<td>4</td>
<td>14</td>
</tr>
</tbody>
</table>

Table 8: LEUAS Flight Performance
Takeoff Distance and Flight Performance predictions are shown in Figure 31. The graph in Figure 32 represents flight conditions the day flight testing occurred. Refer to the legend below for clarification on flight performance.

**Figure 31: Flight Testing Takeoff Distance Graph**

**Figure 32: Flight Testing Performance Graph**
Takeoff Distance and Fight Performance predictions in Figure 33 and Figure 34 represent a max takeoff weight endurance flight for fourteen hours. Refer to the above legend for clarification on results.
**Stability and Control**

With any quality aircraft design it is critical that the stability, or tendency of an aircraft to return to a previous orientation when disturbed by an applied force, is considered. This allows the pilot to operate the aircraft in the traditional manner, without the intervention of any computer stabilization interface. Computer stabilization or control is necessary when a statically instable aircraft is designed. This is possible through the use of proportional integral differential (PID) closed loop control systems: Taking pilot inputs and making the correct control surface deflections enables the aircraft to maneuver as the pilot intended. The natural response of the aircraft wouldn’t be as expected, making for likely uncontrollable flight conditions.

Evaluation of the LEAUS was done through a Stability and Control MATLAB code developed for this project. To evaluate the aircrafts, fifteen physical parameters were needed for proper calculation to obtain a realistic predicted static margin. Ideally, we want the value to be greater than five percent to ensure stability, with the perceived error in our calculation to be on the cautious side for our design. Refer to Appendix V for more information.

**Fifteen Required Inputs**

**English Units**

1. Upper Wing Third Cord Position
2. Lower Wing Third Cord Position
3. Upper Wing Surface Area
4. Lower Wing Surface Area
5. Sweep Angle
6. Horizontal Stabilizer Surface Area
7. Distance Between Horizontal Stabilizer and Estimated Aerodynamic Center
8. Distance Between Wing Body AC and Horizontal Stabilizer Third Cord
9. Upper Wing Wingspan
10. Average Cord Upper Wing
11. Average Cord Lower Wing
12. Empty Aircraft CG Position
13. Position of Fuel Tank CG
14. Empty Aircraft Weight
15. Fuel Weight

*Table 9: Fifteen Required Stability and Control Inputs*
### Static Margin Calculation

<table>
<thead>
<tr>
<th>Fuel Weight (Lbs)</th>
<th>0</th>
<th>1</th>
<th>4</th>
<th>7</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aircraft Weight (Lbs)</td>
<td>22.6</td>
<td>23.6</td>
<td>26.6</td>
<td>29.6</td>
</tr>
<tr>
<td>Flight Time (Hrs)</td>
<td>0</td>
<td>2</td>
<td>8</td>
<td>14</td>
</tr>
<tr>
<td>Static Margin (%)</td>
<td>16.41</td>
<td>17.06</td>
<td>18.74</td>
<td>20.07</td>
</tr>
</tbody>
</table>

*Table 10: Static Margin Results*

As desired, the empty weight static margin is greater than five percent at 16.4% and increases to 20% with a full fuel load. With any additional payload, the max takeoff weight would be exceeded, requiring the use of potentially excessive trim. So to stay within the max takeoff weight, less fuel would be loaded. As you have likely observed, there is no payload weight factor for our static margin calculations. We were most concerned with being statically stable with low fuel levels; we achieved this by adding payload significantly forward of the fuel tank. This meant the plane would become increasingly stable at low fuel levels, so it isn’t a concern of instability. It is hypothesized that increased nose up trim would be required due to the further forward CG. Potential pitch control limitations may have been discovered during flight testing at higher payloads, but unfortunately we were not able to evaluate those conditions.
Progress Photo Documentation & Captions

Figure 36: Bare Engine Dyno Stand

Titebond Original Wood Glue used for lamination and other major fuselage construction. A bit heavier option but wanted to ensure structural integrity.

Figure 37: Laminating Fuselage Profiles (2 x 1/8” Birch Plywood)

Figure 39: Laminating Various Fuselage Sections (Birch Plywood)

Figure 38: Laminating Upper Wing Mount (2 x 3/16” Birch Plywood)
Right before Titebond was applied and structure compressed with clamps. Clamped over major bulkheads and floor boards.

Before the fuselage was constructed it was critical to installation the nose gear. Advice: Fully evaluate each step of construction to make sure you’re not complicating a future step.

The engine was aligned on the center of the firewall for ease of initial installation. This provided enough room for the charging system to be installed as well.
A mounting structure was built to mate the lower wing to the fuselage with the addition of a plywood plate (Figure 45) and reinforcement structure. Minor modification to the fuselage was required. Primarily sanding away of material (Figure 46) on the side of the aircraft allow for wing to recess properly into fuselage.

Note the two vertical 1/8" balsa strips (Figure 44) to aid in alignment of the wing to the fuselage. As well as the ¼ x 20 steel bolts through the top of the wing. The top was reinforced with 1/8” plywood to avoid any crushing or deformation when loads were applied to the wing.
Figure 51: Completed Modification to Fuselage (Post Crash)

Figure 50: Bottom of Lower Wing Mount

Figure 49: Initial Wiring Harness Configuration

Figure 48: Setting Lower Wing Incidence Angle
Figure 52: Having to Disassemble the Battery to Use the Drill

Figure 53: 3-Phase AC to DC Rectifier
Installing the lower wing and wing struts drastically improved over all wing rigidity. This solved our excessive wing deflection problem (Figure 54)

Figure 54: Excessive Upper Wing Deflection

Figure 55: Wing Struts Installed
Window Install Process
1. Cut Clear Monokote to cover desired section
2. Cut out window from overlaying Monokote by following existing structure
3. Using low heat, apply clear covering
4. Finish by applying overlaying colored covering
Use Caution when using heat gun as it easily melts through clear covering.

Propeller Trimming Method
1. Draw a line tip to tip
2. Draw parallel line to edge of propeller desired distance away
3. Hand cut
4. Round edges and make each tip close to identical
5. Balance
6. Lightly sand heavy end
7. Repeat 5 & 6 until acceptable
Flight Testing

To properly evaluate any aircraft, it is necessary to use it as intended, not just to simulate. Real-world, high-risk flight testing is required: Where all oversights, assumptions, solutions, and investments are put on the line and proven to be true or false. That is all, no more or less, as long as it is operated within design limitations and flown appropriately to what it was designed to do. Then it will be a successful flight test at no fault to the pilot. Fortunately we are only operating radio controlled aircraft, and barring direct impact, harm to the pilot is appropriately low.

On November 9th, 2014 we ventured to Muncie, Indiana to fly at the Academy of Model Aeronautics (AMA) headquarters and primary airfield of international caliber. Conditions were ideal for that time of year, moderate winds 5-10 mph, sunny, and relatively warm at 40-45°F. We arrived by 10AM and began setting up our ground station. At this point we gave the charging system one last opportunity to work. After several tensioning methods, we observed aggressive gear and decided to call it and uninstall our charging system motor. This saved a bit of weight and simplified our aircraft. We began to top off all of our batteries in preparation for initial taxi testing and our first flight. After about an hour all of our batteries were topped off and we had the wings installed and struts secured. Next step would be warm up the engine and monitor battery pack voltages to make sure they remained stable under load.

Here are the first indications that we should have stopped testing that we missed. Under deflections of all control surfaces the servo battery back voltage dropped significantly and continued to drop as deflections cycled. This indicated a large load was being applied to the battery, which it was not able to provide the power required. A proper solution would have been to use a higher quality servo battery pack with a higher C (discharge rate) rating, as the battery
pack we were using was intended for ground vehicles which generally use significantly fewer servos than we were (low C rating). Also, with the indicated large load, the battery pack would have likely not operated the plane for long if the following event didn’t occur. **Takeaway:** Use a battery pack with appropriate discharge rating and capacity for your application. Repurposing a battery pack as we did was risky and **not advised.** Simply, we did not have time, nor did we know to investigate this system. It cost us.

This next lesson was taught quickly and proved to be catastrophic: once the engine was tuned in for the day and was warmed up, we began to taxi around; adjusting nose wheel alignment and elevator trim so it could be slightly up to help with rotation and the presumed nose heavy condition. Then, once the pilot was comfortable with its ground handling characteristics, he performed a high speed taxi run up.

*Pilots Note:* With triple rates there were ample options to adjust for taxing speed; high rates were to be used only at very low speeds as the plane was very responsive. An inverse relation held true for medium and low rates as speed increased use lower rates. It would be advised that less nose wheel deflection would make for an easier aircraft to control on the ground as it was very sensitive. This is possible by placing the nose wheel linkage at the base of the servo horn and the tip of the nose wheel control horn. Refer to Figure 58 for clarification on nose wheel control horn.

![Figure 61: Takeoff](image)

Finally it was time to fly LEUAS. We were all anxious to see what would happen! As the pilot lined LEUAS up on centerline and slowed it to a stop at the beginning of our measured takeoff markings. The wind picked up as the sun slipped behind the incoming clouds and there was nothing left to do but fly. The throttle was advanced and our aircraft briskly accelerated down the runway. The engine came to life, revving out to 8000 rpm. Reaching 40mph, the main gear became light and it rotated in only 150 feet with minimal elevator input. This indicated an
appropriately balanced aircraft. Immediately it was apparent that significant left aileron input was required to counteract the torque of the engine. This was easily corrected for and LEUAS gently climbed as it was brought around on the downwind leg of its flight. Several stall conditions were encountered, as it was impossible to monitor airspeed and keep the aircraft in view at the same time. Each time the nose came back down and normal flight resumed. This is a reassuring sign of a stable aircraft. Realizing that the trim required was more significant than expected, assistance was requested by a fellow pilot whom we met that day.

*Pilots Note: The next few moments are very important and are an indication of low voltage/power available to the control system.* As we began the base leg of our flight pattern, the plane began to diverge from controlled flight. Best described as mushy or unresponsive control sticks, it seems you have to deflect your sticks more and more to obtain the same response ultimately being unable to change its flight path at all. This is a very bad situation and you typically only have seconds left before you become an observer instead of a pilot as you were moments before... A very discomforting experience to any pilot but this is why we learn from mistakes, ideally others mistakes but firsthand experience is always the best unfortunately.

From that moment on our fate was sealed, as we heard the engine go silent and the airspeed drop. The nose dropped and the LEUAS banked right as it made its final decent. Shortly thereafter we lost sight of it behind a hill. Given the high speed, we knew it was all over. The only thing to do was go pick up the wreckage.
Recoverable Flight Test Data

Figure 65: Flight Test Data Graph

<table>
<thead>
<tr>
<th>Flight</th>
<th>Airspeed (Mph)</th>
<th>Ground Speed (Mph)</th>
<th>Altitude (Feet)</th>
<th>Rate of Climb (Ft/S)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max</td>
<td>52.863</td>
<td>55.660</td>
<td>252.166</td>
<td>31.300</td>
</tr>
<tr>
<td>Average</td>
<td>42.357</td>
<td>38.049</td>
<td>116.786</td>
<td>5.052</td>
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<tr>
<td>Min</td>
<td>33.030</td>
<td>30.745</td>
<td>20.566</td>
<td>-12.768</td>
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</table>

Table 11: Flight Data Table

<table>
<thead>
<tr>
<th>Takeoff</th>
<th>Time (Seconds)</th>
<th>Distance (Feet)</th>
<th>Approximate Linear Acceleration (Ft / S^2)</th>
<th>Static Thrust (Lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>8</td>
<td>150</td>
<td>4.6875</td>
<td>10</td>
</tr>
</tbody>
</table>

Figure 64: Taxi & Flight Path.
Red X marks the believed position of the aircraft as the pilot began to lose control. Beginning of battery pack failure. Loss of connection occurred where the flight path ends abruptly.
Recommendations

- Reduce structure weight from 53% of empty aircraft weight to a more refined value, perhaps 35 – 40% would be a reasonable weight. The lower the better while not risking structural failure or fatigue under critical conditions.
- Replace NiMh battery packs with LiPo’s to save weight, gain charging capacity, and allow for less voltage drop when loads are applied. Significant voltage drop was observed with freshly charged 5 cell NiMh batteries indicating insufficient C (discharge rating) for the battery packs.
- Properly analyze entire electrical system to ensure no wire will be at risk for supplying too much current and failing. Take a weight penalty to ensure proper operation and reliability of aircraft systems
  - Test worst case scenario by fixing all control surfaces so that they are rigid and monitor servo battery pack voltage and current. While control sticks are deflected to corner’s to determine near worst case power requirements to the servos. There is hesitation in saying worst case scenario, as an absolute dynamic loading may be more significant than this static test. Record peak current and size supply and supplemental wires properly with a safety factor.
- Based on our experience, we do not suggest running a charging system off of the main engine because of performance loss and the potential for the main propeller to be stopped if the alternator or gearing system were to fail.
  - We do suggest, perhaps as a separate senior project, that a gas powered or glow fuel powered generator system be developed to produce 40 watts of power to maintain/charge the onboard aircraft systems. A supply system over a charging system is recommended. Because of the desire to use LiPo batteries, the need for cell balancing is necessary, which will complicate the charging and discharging system. If you and your team are able to overcome this then by all means use a charging system with LiPo’s. However, a repeat of the electrical failure we experienced must be avoided.
    - In designing a charging system we recommend a direct coupling of the driveshaft of the charging engine to the chosen electric motor. In reference of electric motors we suggest a Kv (RPM/Volt) rating of 300 or
lower to produce sufficient voltage at said operating RPM of the engine. Alternative, an engine with a higher rpm band may also be employed with a motor having a higher Kv value. Testing of each motor is recommended because a motor’s rated Kv changes when it is used as a power generator instead of as an electric motor. Once these critical components have been evaluated, a properly sized 3 phase AC to DC rectifier can be selected.

- We would recommend installing properly sized diodes to only allow current to flow out of the batteries, into the voltage regulators inputs, out of the voltage regulators outputs, and onto each desired system. This eliminates back feeding from one system into another, which could result in charging a LiPo battery without cell balancing. This would also isolate each system, meaning if one were to fail or short out, continued operation may still be possible.

- Install a redundant/backup power supply system so no one failure can lead to loss of control or connection with the aircraft.
  - Properly size your wire!! Invest in quality wire!

- We do not recommend a gear or belt drive due to the difficulty in making these components robust and precise enough to work reliably at the high speeds that are experienced. If one feels motivated, recognize this challenge is not for the faint of heart and will require a significant amount of design and custom machine work.

- When installing a pressure port into your muffler, use a steel bolt with a hole drilled though it. The brass ones available will break easily.
  - Ensure you have positive tank pressure to overcome a lower fuel level than carburetor height. Simply, when you remove your fuel line from the carburetor, fuel should flow from it as the tank depressurizes. Do not just rely on the fuel pump to pull fuel from the tank if your engine has one.

- Don’t spend too much time selecting and evaluating and engine. Once you have reached the best inflight thrust production (dynamic thrust), stay with that prop and engine.
  - A good process to select an engine would be select the thrust to weight ratio you want, size propeller you want to use and what rpm to achieve the static thrust.
required, and select an engine that can meet those prop sizing and rpm requirements. Then provide enough fuel for it to fly your desired duration. This is a simpler process than working backwards from fuel consumption desired and complicating engine selection and using a propeller that wasn’t meant for that engine size as we have.

- Engines are designed to operate throughout their RPM range and if you don’t utilize the majority of the advertised RPM range they will not run properly. Tuning the carburetor will be complicated and excessively finicky: you will likely not be able to achieve maximum power available because you will not be able to utilize the top end of the power band.
  - If you wish to do more research into engines and propeller selection we recommend doing it as a separate senior design project.

- We recommend a more refined empty weight static margin to allow for lower required trim and control forces allowing for potentially higher payloads. Basically, balance the aircraft more accurately without using ballast. Ballast is dead weight and using it is not good aerospace engineering practice. It’s obvious: add a few more systems and make it more complicated.
Conclusion

Expenses

<table>
<thead>
<tr>
<th>Expense Breakdown</th>
<th>Amount</th>
</tr>
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<tbody>
<tr>
<td>Engine &amp; Components</td>
<td>$ 576.66</td>
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<tr>
<td>Fuel System</td>
<td>$ 120.58</td>
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<tr>
<td>Props</td>
<td>$ 143.73</td>
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<td>Flight Computer</td>
<td>$ 481.49</td>
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<td>RC Electronics</td>
<td>$ 510.15</td>
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<tr>
<td>Aircraft Structure</td>
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<td>Landing Gear</td>
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<tr>
<td>Charging System</td>
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<tr>
<td>Fuel</td>
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<tr>
<td>Engine Dyno</td>
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<tr>
<td>Camera System</td>
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<tr>
<td>Lighting</td>
<td>$ 23.65</td>
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<tr>
<td><strong>Total</strong></td>
<td><strong>$ 3,069.64</strong></td>
</tr>
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</table>

*Table 12: Expense Breakdown*

**Expense Distribution**

*Figure 66: Expense Distribution*
Weight Analysis

LEUAS Weight Analysis

Figure 67: Weight Analysis

Rebuild

Eight months ago we set course to work on this project and devoted a massive proportion of what time we could spare to it. Spending weekends, evenings, nights, mornings, and portions of our breaks to accomplish as much as we have. It was a fantastic learning experience and we solved more problems than we ever anticipated. For another group to not follow in our footsteps would be a real loss as there is a great chance to improve on what we have done. By halving the structural weight, scaling down the

<table>
<thead>
<tr>
<th>Rebuild Expense</th>
<th>Prediction</th>
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<td>Engine Repair</td>
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<td>Fuel System</td>
<td>Use Available</td>
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<td>Props</td>
<td>$25.00</td>
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<td>Flight Computer</td>
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<tr>
<td>RC Electronics</td>
<td>$100.00</td>
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<tr>
<td>Aircraft Structure</td>
<td>$300.00</td>
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<tr>
<td>Landing Gear</td>
<td>$40.00</td>
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<td>Charging System</td>
<td>Don’t Develop</td>
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<tr>
<td>Fuel</td>
<td>$6.00</td>
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<tr>
<td>Engine Dyno</td>
<td>Not Relevant</td>
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<tr>
<td>Camera System</td>
<td>$115.00</td>
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<td>Lighting</td>
<td>$25.00</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>$1,192.49</strong></td>
</tr>
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</table>
endurance requirement (three to six hours), using high capacity LiPo battery packs for the systems, and repairing the engine (new carburetor, intake tube, and spark plug all quite easy to replace), you would have a fully capable unmanned aircraft easily capable of carrying significant payloads for an estimated twelve hundred dollars. Build again! Make waves, be a trend starter, inspire others, shoot for the moon!! Make this a project that you will be proud to talk about.

**Closing Remarks**

In closing it only took one mistake to cause our aircraft to terminate itself midflight and to cut our project short. To be frank it was quite amateur of us to miss such a perceivably obvious problem as we did, but consider we were operating at a very fast pace and were contending with dwindling flying weather. It’s unlikely that any other team would have been able to pull off what we have, even if it ended up as a shattered wreckage of an aircraft. The LEUAS did fly and was controllable, proving that what we learned in the classroom does directly apply to an actual aircraft build. From the crash, we have learned to pay more attention to onboard power requirements and the need to test for potential faults in them more thoroughly. Therefore, from this standpoint, this project can be viewed as somewhat successful. The first steps towards a platform capable of sustained autonomous flight have been made. Now all that is needed are future groups willing to continue to push development and push what is possible. To take steps into the dark allows you to discover the light, enticing others to push beyond what is known.
Works Cited


http://shop.antigravitybatteries.com/micro-start-xp-1/


Appendices

Appendix I: Raw Alternator Testing Data

The table below shows testing results from the alternator tested at various speeds. See Alternator Selection and Testing for more specific details. The resistance values given were measured across the terminals of the load created by the light bulbs. As these values were recorded with the bulbs cold, they are estimates only and were not used in calculating the power output.

<table>
<thead>
<tr>
<th>RPM</th>
<th>Voltage (V)</th>
<th>Amperage (A)</th>
<th>Power Output</th>
<th>kV value</th>
</tr>
</thead>
<tbody>
<tr>
<td>1500</td>
<td>2.30</td>
<td>0.00</td>
<td>0.00</td>
<td>652.17</td>
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<td>2550</td>
<td>4.35</td>
<td>0.00</td>
<td>0.00</td>
<td>586.21</td>
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<td>3900</td>
<td>7.00</td>
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<td>0.00</td>
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<td>4860</td>
<td>8.95</td>
<td>0.00</td>
<td>0.00</td>
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<tr>
<td>6120</td>
<td>11.50</td>
<td>0.00</td>
<td>0.00</td>
<td>532.17</td>
</tr>
</tbody>
</table>

Trial 2: 2 bulbs in series (R=35ohms, approx.)

<table>
<thead>
<tr>
<th>RPM</th>
<th>Voltage (V)</th>
<th>Amperage (A)</th>
<th>Power Output (W)</th>
<th>kV value</th>
</tr>
</thead>
<tbody>
<tr>
<td>1500</td>
<td>1.46</td>
<td>0.04</td>
<td>0.06</td>
<td>1027.40</td>
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<tr>
<td>2520</td>
<td>3.44</td>
<td>0.08</td>
<td>0.28</td>
<td>732.56</td>
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<tr>
<td>3800</td>
<td>5.95</td>
<td>0.10</td>
<td>0.60</td>
<td>638.66</td>
</tr>
<tr>
<td>4650</td>
<td>7.65</td>
<td>0.12</td>
<td>0.92</td>
<td>607.84</td>
</tr>
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</table>

Trial 3: 1 bulb (R=25ohms, approx.)

<table>
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<tr>
<th>RPM</th>
<th>Voltage (V)</th>
<th>Amperage (A)</th>
<th>Power Output (W)</th>
<th>kV value</th>
</tr>
</thead>
<tbody>
<tr>
<td>1500</td>
<td>1.44</td>
<td>0.04</td>
<td>0.06</td>
<td>1041.67</td>
</tr>
<tr>
<td>1800</td>
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<td>0.10</td>
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<td>2400</td>
<td>3.08</td>
<td>0.08</td>
<td>0.25</td>
<td>779.22</td>
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<td>2460</td>
<td>3.31</td>
<td>0.08</td>
<td>0.26</td>
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<tr>
<td>3000</td>
<td>4.34</td>
<td>0.09</td>
<td>0.39</td>
<td>691.24</td>
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<tr>
<td>3300</td>
<td>5.08</td>
<td>0.10</td>
<td>0.51</td>
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<td>3750</td>
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<tr>
<td>3800</td>
<td>6.00</td>
<td>0.10</td>
<td>0.60</td>
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<tr>
<td>4500</td>
<td>7.10</td>
<td>0.11</td>
<td>0.78</td>
<td>633.80</td>
</tr>
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</table>
### Trial 4: 2 bulbs parallel (R=20ohms, approx.)

<table>
<thead>
<tr>
<th>RPM</th>
<th>Voltage (V)</th>
<th>Amperage (A)</th>
<th>Power Output (W)</th>
<th>kV value</th>
</tr>
</thead>
<tbody>
<tr>
<td>1230</td>
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### Trial 5: 3 bulbs parallel (R=10 ohms approx.)

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Appendix II: Graphed Alternator Testing Data with Best Fit Curves

**Trial 1 RPM vs Open Terminal Voltage**

\[ y = 0.002x - 0.7185 \]

**Trial 2 RPM vs Voltage**

\[ y = 0.002x - 1.4981 \]

**Trial 2 RPM vs Power**

\[ y = 4 \times 10^{-8}x^2 + 5 \times 10^{-5}x - 0.088 \]

\[ R^2 = 0.9986 \]
Trial 3 RPM vs Voltage

\[ y = 0.0019x - 1.4611 \]

Trial 3 RPM vs Power

\[ y = 0.0002x - 0.331 \]

Trial 4 RPM vs Voltage

\[ y = 0.0017x - 1.4992 \]
Trial 4 RPM vs Power

\[ y = -1E-10x^3 + 3E-06x^2 - 0.0061x + 2.8694 \]

Trial 5 RPM vs Voltage

\[ y = 0.002x - 1.7583 \]

Trial 5 RPM vs Power

\[ y = -7E-12x^3 + 2E-07x^2 + 9E-05x - 0.2937 \]
### Appendix III: Corrected Alternator Testing Data

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### Appendix IV: Performance MATLAB Code

```matlab
clc
clear all
close all
format compact

Takeoff_Weight_Lbs = 30
Altitude_Ft = 932;
FlightTimeHR = 1;
Estimated_Cruise_Speed_MPH = 50;
Static_Thrust_Lb = 10;
HP = 2.3;
Etapr = .75;
Clmax = .7;
```
\[ T_W = \frac{\text{Static Thrust Lb}}{\text{Takeoff Weight Lbs}} \]

Swet = 37.5;

Temperature = 50;

b = 6.6; % Wing Span (Feet)

cr = 13/12; % Root Cord Length

ct = 7/12; % Tip Cord Length

TR = ct/cr; % Taper Ratio

c = (ct+cr)/2; % Average Chord Length

S_Top = b*c; % Wing area

S_Lower = 560;

S = S_Top+S_Lower/144;

S_in = S*144;

W_S = \frac{\text{Takeoff Weight Lbs}}{S};

AR = \frac{(b^2)}{S};

e = 1.78*(1-.045*AR^.68)-.64;

K = \frac{1}{(\pi*e*AR)};

Full_Fuel_Gal = 1.4;

Min_Fuel_Level_Gal = .05;

Maximum_Fuel_Consumption_Rate_CC_Min = 5;

rho = .00237 - (\text{Altitude Ft}+800)*(4.933E-8);

miu = (4.6e-10)*\text{Temperature}; % Kinematic Viscosity

Vest = \text{Estimated Cruise Speed MPH} * 1.46;

RE = (\text{rho*Vest*c})/\text{miu}; % Operating Reynolds Number

cf = .42/(\log(.056*\text{RE})^2);

SwetS = Swet/S; % Wetted Area to Wing area ratio

Cdo = SwetS*cf;

LoDmax = \sqrt{\frac{1}{(4*Cdo*K)}}

Endurance_Airspeed_Mph = \sqrt{\left(\frac{2}{\text{rho}}\right)\sqrt{\frac{K}{(3*Cdo)}}*W_S}/1.46

Vstall_Mph = \sqrt{\left(\frac{2}{\text{rho}}\right)*W_S*(1/\text{Clmax})}/1.46
Power = 746*HP;

%Flight Simulation

TimeLanding = 60*FlightTimeHR;

Time = 1:1:TimeLanding;

for i = 1:length(Time)

Fuel_Used(i) = Maximum_Fuel_Consumption_Rate_CC_Min * Time(i);

Aircraft_Weight(i) = Takeoff_Weight_Lbs-(.00179*Fuel_Used(i));

W_S(i) = (Takeoff_Weight_Lbs-(.00179*Fuel_Used(i)))/S; % .00179 is lb/cc of fuel with a density of 6.8lb/gal

Endurance_Airspeed_Mph(i) = sqrt((2/rho)*sqrt(K/(3*Cdo))*W_S(i))/1.46;

Vstall_Mph(i) = sqrt((2/rho)*W_S(i)*(1/Clmax))/1.46;

Takeoff_Roll(i) = (1.21*W_S(i))/(32.2*rho*Clmax*T_W);

Rate_of_Climb(i) = .5*((Etapr*Power)/Aircraft_Weight(i))-sqrt((2/rho*sqrt(K/(3*Cdo))*W_S(i)))*(1.155/LoDmax);

Climbing_Cruise_Airspeed(i) = sqrt((2/rho*sqrt(K/(Cdo))*W_S(i)))/1.46;

Climb_Angle(i) = .333*asind(((Etapr*Power)/(Climbing_Cruise_Airspeed(i)*1.46*Aircraft_Weight(i))-.5*rho*((Climbing_Cruise_Airspeed(i)*1.46)^2)*(1/W_S(i))*Cdo-W_S(i)*((2*K)/(rho*((Climbing_Cruise_Airspeed(i)*1.46)^2))));

end

disp(' ')

Vstall_Mph(1)

Takeoff_Roll(1)

Endurance_Airspeed_Mph(1)

Rate_of_Climb(1)

Climbing_Cruise_Airspeed(1)

Climb_Angle(1)

Aproximate_Fuel_Required_Gallons = Fuel_Used(TimeLanding)/3785

figure(2)

plot(Time,Takeoff_Roll)

grid on

xlabel('Flight Time (Min)')
ylabel('Feet')
title('Takeoff Distance vs Flight Time')

figure(1)

plot(Time,Endurance_Airspeed_Mph,Time,Vstall_Mph,Time,Aircraft_Weight,Time,Ra
te_of_Climb,Time,Climbing_Cruise_Airspeed,Time,Climb_Angle)

grid on

xlabel('Flight Time (Min)')
ylabel('Airspeed (Mph)')
title('Airspeed vs Flight Time')
legend('Endurance Cruise Speed','Stall Speed','Aircraft Weight','Rate of Climb (Ft/s)','Climbing Airspeed','Climb Angle','location','best')

Appendix V: Stability & Control MATLAB Code

clear all
close all
format compact

% Cameron Segard
% 9-18-2014
% LEUAS Lower Wing Positioning

% All measurements are in relation to a specific datum point on the aircraft
% inline with the leading edge of the top wing

% Units: Inches & Pounds

% Wing Locations
XLWThrddCordd = -6.5;

% Wing & Tail Specifications Units: Inches^2
Tail_Surface_Area = 160;
AUpperWing = 880;
ALowerWing = 528;
Swpl4 = 0;

hh = 4;  % Vertical distance of horizontal tail from AC
ltbar = 32;  % Wing AC to horizontal tail AC distance
b = 80;  % Total wing span of both wings
A = b^2/(AUpperWing + ALowerWing);
l = ((7/13)*AUpperWing+(6.75/10)*ALowerWing)/(AUpperWing + ALowerWing);
% Taper ratio average
CTW = 11;
CLW = 8.375;
Kh = (1-(hh/b))/(3*sqrt(2*ltbar/b));
Kl = (10-3*l)/7;
Ka = (l/A)-(1/(1+A^1.17));
\[ \text{de}_{\text{da}} = 4.44 \times (\text{Ka} \times \text{Kl} \times \text{Kh} \times \cos(Swp14)^{.5})^{1.19}; \]

% Longitudinal Positions

\[ \text{XTWThrdCordd} = -3.63; \]
\[ \text{Xcgd} = -3.5; \]
\[ \text{XFuel} = -2; \]

% Weights

\[ \text{Wplane} = 22.6; \text{Weight as of 11-8-2014} \]
\[ \text{Fuel Weight} = 7; \]

% Calculations

\[ C = (\text{CTW} + \text{CLW})/2 \]
\[ \text{Total Wing Area} = \text{AUpperWing} + \text{ALowerWing} \]
\[ \text{Final Weight} = \text{Wplane} + \text{Fuel Weight} \]
\[ \text{Xcg} = (\text{XFuel} \times \text{Fuel Weight} + \text{Xcgd} \times \text{Wplane})/\text{Final Weight} \]
\[ \text{Xacw} = ((\text{AUpperWing} \times \text{XTWThrdCordd} + \text{ALowerWing} \times \text{XLWThrdCordd})/\text{Total Wing Area}) \]
\[ \text{h}_{\text{wAC}} = \text{norm}((\text{Xacw} - ((C \times .33) + \text{Xacw}))/C) \]
\[ \text{h}_{\text{CG}} = \text{norm}((\text{Xcg} - ((C \times .33) + \text{Xacw}))/C) \]
\[ \text{Static Margin}_{\text{wb}} = 100 \times (\text{h}_{\text{wAC}} - \text{h}_{\text{CG}}) \]
\[ \text{Vhbar} = (\text{lbar} \times \text{Tail Surface Area})/(C \times \text{Total Wing Area}); \]
\[ \text{hn} = \text{h}_{\text{wAC}} + (\text{Tail Surface Area}/\text{Total Wing Area}) \times \text{Vhbar} \times (1 - \text{de}_{\text{da}}) \]
\[ \text{Static Margin} = 100 \times (\text{hn} - \text{h}_{\text{CG}}) \]
Appendix VI: Assessment Program Outcomes

Assessment of Program Outcome #9

ME 4790/ME 4800

The MAE faculty members have identified “A knowledge of contemporary issues” as one of the student outcomes for both mechanical and aeronautical engineering programs.

Contemporary issues are any issues that you hear on the news related to new and old products and their safety, new innovations, technologies, standards, and regulations in general. As you develop your proposal for your senior design project, we ask you to start answering the following questions. These questions would guide you in the development of ideas you need to include in your proposal and final project reports. You are required to submit the completed form with your final proposal in ME 4790 and again with your final report in ME 4800. In your proposal and report, please include page references in response to each question below.

Evaluation of program outcome “A Knowledge of contemporary issues”

1. Why is this project needed now?

   *In recent years, much effort in both industry and at universities has been directed into the development of Unmanned Autonomous Systems (UAS). This project is needed now so that future systems developed at WMU can be tested in flight.*

2. Describe any new technologies and recent innovations utilized to complete this project

   *Recent technologies used to complete this project include the utilization of off the shelf autopilot systems.*

3. If this project is done for a company-how will it expand their potential markets?

   *This project is not done for a company, although there is a growing market for unmanned aircraft.*
4. How did you address any safety and/or legal issues pertaining to this project (e.g., OSHA, EPA, Human Factors, etc.)

   One crucial element into human safety involved the testing of the engine. To mitigate potential for injury, much of the testing was automated. Additionally, care was taken when working on or around the engine: Group members were aware of the dangers.

   An important legal aspect to this project involved Federal Aviation Administration (FAA) restrictions. As this plane is technically considered a hobby aircraft, it had to fly below a 400 foot envelope. Additionally, flying the plane outside the operator’s line-of-sight was not permitted. These guidelines were adhered to when the system was flown.

5. Are there any new standards or regulations on the horizon that could impact the development of this project?

   As was stated under the previous question, FAA regulations must be followed to ensure the UAS is not flown too high or too far away from the operator.


   Currently there is not a potential for a patent in this design.
Assessment of Program Outcome #12  
**ME 4790/ME 4800**

The MAE faculty members have identified “An understanding of the impact of the engineering solutions in a global, environmental and societal context”

As one of the program outcomes for both mechanical and aeronautical engineering programs. As you develop your proposal for your senior design project, we ask you to start answering the following questions. These questions would guide you in the development of ideas you need to include in your proposal and final project reports. You are required to submit the completed form with your final proposal in ME 4790 and again with your final report in ME 4800. In your proposal and report, please include page references in response to each question below.

Evaluation of program outcome “An understanding of the impact of the engineering solutions in a global, environmental and societal context”

1. Is the project useful outside of the United States? Describe why it is or is not—provide details.
   
   The project is useful outside of the United States. UAS technology is being developed in other countries. What this technology can provide (for example, surveillance) has value in other countries and other areas outside of the United States.

2. Does your project comply with US and/or international standards or regulations? Which standards are applicable?
   
   This project does comply with US standards, namely FAA regulations requiring the UAS to fly below 400 feet and within sight of the operator. International regulations will not be examined, as it is beyond the scope of this project.

3. Is this project restricted in its application to specific markets or communities? To which markets or communities?
   
   This project could be restricted in certain communities. These areas include zones in which operation of a UAS are not permitted, enforced by the FAA, local government, or otherwise.
4. If the answer to any of the following is positive, explain how and, where relevant, what were your actions to address the issues?

To address the issues outlined in question 3, flight location was critical: Local ordinances were observed and all necessary guidelines followed. If applicable, permission to fly the UAS over a particular area will be obtained before flight takes place.

Design is focused on serving human needs. Design also can either negatively or positively influence quality of life. Address the impact of your project on the following areas.

**Air quality?**

Initially, this project involved a great deal of gasoline engine testing, which was accomplished on a testing bench. To ensure air quality was affected, these tests were carried out in well-ventilated areas. When this wasn’t possible (for example, in wind tunnel testing), care was taken to channel the flow of exhaust to a safe area. Air quality ceased to be an issue when the UAS was flown, as this operation was carried out in a wide open area, where exhaust fumes could not be trapped, preventing harm to both the operator and bystanders.

**Water quality?**

This project does not involve nor affect water quality.

**Food?**

This project does not involve nor affect food.

**Noise Level?**

The gasoline engine that was used in this project can have an impact on noise level. Initially, the engine was operated on a test stand, occasionally with bystanders around. Bystanders were either be required to wear necessary hearing and eye protection or were barricaded from the engine. Once the engine was installed on the UAS and was in flight, noise level was not a concern, as flight only took place where the elevated noise level was acceptable.
Does this project impact:

**Human Health?**

*Provided those involved with engine testing and flight take necessary safety precautions, this project does not impact human health.*

**Wildlife?**

*This project does not impact wildlife.*

**Vegetation?**

*This project does not impact vegetation.*

Does this project improve:

**Human Interaction?**

*This project does not improve human interaction*

**Well being?**

*This project does not improve well being*

**Safety?**

*This project does not improve safety.*

**Others?**

*Not Applicable*
Assessment of Program Outcome #13

ME 4790

The MAE faculty have identified “A recognition of the need for, and ability to engage in life-long learning” as one of the program outcomes for both mechanical and aeronautical engineering programs. As you develop your proposal for your senior design project, we ask you to start answering the following questions. These questions will guide you in the development of ideas you need to include in your proposal and final project reports, as well as help you identify areas in which you need improved proficiency. You are required to submit the completed form with your final proposal in ME 4790 and again with your final report in ME 4800 (addressing slightly different points of view). In your proposal and final project report, please include page references in response to each question below. This item will be included in the Team Assets section of the proposal. The format of the response to the questions in the report is of your own choosing but must address the below listed questions. Questions 2, 3 and 4 will also be directly addressed in the final Appendix of the report in the format shown below.

Your responses will be used in the Evaluation of program outcome “A recognition of the need for, and ability to engage in life-long learning”

A well-organized team brings necessary backgrounds and talents together that are needed to successfully execute the design process. Each team member plays an important role on the design team. Individual members must be prepared to gain any additional skills necessary, and improve existing skills during project execution.

For each team member:
1. In detail identify the skills you bring to your design project that would be considered assets to the project team
2. Delineate the skills necessary to successfully execute your responsibilities on the project.
3. Define skills you will personally need to strengthen to achieve the task at hand.
4. Explain how you plan to gain the skill level necessary to successfully execute your responsibilities to the design team.
Shane Russell:

1. Skills I bring to this design project that would be considered assets to the project team include courses taken in thermodynamics and engine design, courses taken in relation to materials and materials mechanics, familiarity with basic electronic principles, experience in CAD modelling, and experience in machining and metalworking.

2. Skills necessary for me to successfully execute my responsibilities on the project included obtaining a more in-depth understanding of electrical generation and battery charging, methods of model aircraft construction, skills necessary to interpret experimental data from engine test results, and the ability to machine necessary pulleys, brackets, and other required components.

3. Skills I personally needed to strengthen to achieve the task at hand include knowledge of electrical generation, knowledge of battery charging, and establishing necessary testing procedures.

4. I plan on achieving the necessary skill levels of the skills outline in question 3 by researching said skills and, if necessary, asking advice from someone knowledgeable in the applicable field.

Andreas Quainoo:

1. I am proficient in several programming languages which I believe will be key to developing a LabVIEW program that meets our needs.

2. Fundamental parameters of aircraft performance and how they impact the project

3. Understand the fundamental parameters and how they can be optimized to meet our mission objective. Build on the team work skills and improve communication between other members

4. Speak with faculty and perform testing of the engine to understand how the decision we make affect its performance
Cameron Segard:

1. I bring a significant amount of leadership and project planning experience. Allowing me to work with each portion of the project to maximize our potential for success by providing connections and access to finances/facilities for manufacturing and testing. I also have a strong background in 3D modeling, aircraft design, RC systems and building techniques, RC piloting, trouble shooting, wood working, time management and problem management.

2. To carry our my portion of the project I will need to utilize my skills as an aircraft designer, build such aircraft with my wood working and 3D modeling skills, and pilot said aircraft successfully through manual flight testing and autopilot operations. While maintaining gas engine operations and reliability through its continued use, constant pre-flight inspections and preparations to catastrophic failures jeopardizing the projects.

3. I will need to refine my understanding of how electronic system vibration affects its operations, work on independent project development and not step in to work on others portions of the project. Develop my knowledge of large capacity fuel systems, separation of several electrical systems to eliminate noise from each other.

4. By performing online research, working with teammates, and hands on testing I will acquire the skills necessary to achieve all tasks at hand.
## Appendix VII: Gantt chart

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Appendix VIII: Resumes
Shane Russell
sirussell49@yahoo.com

Education
Western Michigan University
College of Engineering and Applied Sciences
1903 W. Michigan Ave
Kalamazoo, MI 49008
Pursuing Bachelor's in Mechanical Engineering.
Graduation: Fall 2014
September 2012 - Present

Glen Oaks Community College
62249 Shimmel Road
Centreville, MI 49032
Associates Degree in General Studies. Additional courses taken involving CAD, Machine Tool, and welding
August 2010 - May 2012

Kalamazoo Valley Community College
6767 W O Avenue
Kalamazoo, MI 49009
Part time student. Courses taken to transfer towards engineering degree
June 2011 - May 2012

Work Experience
Armstrong International Steam and Condensate Group
816 Maple Street
Three Rivers, MI 49093
Engineering Intern. Experience Includes:
- Design, testing, and improvement of Controlled Disk Steam Traps
- Casting design for valves and fittings
- Product testing and documentation
- Other product design
- Developing of program to verify ASME Section VIII code calculations
July 2013 - Present

Armstrong International Hot Water Group
221 Armstrong Blvd.
Three Rivers, MI 49093
Temporary position in CAD Department. Experience Included:
- Updating existing drawings and assembly files
- Creating drawings for sales department and customers
- Creating production drawings and models
- Assisting engineers in product design
November 2011 - July 2013

Glen Oaks Community College
62249 Shimmel Road
Centreville, MI 49032
Tutor. Subjects included Math, Chemistry, Computers, and English
September 2011 - April 2012

Skills
- Experience with CAD software: Very proficient with SolidWorks, familiar with Inventor and AutoCAD
- Experience with Microsoft Visual Basic for Applications (VBA) for Excel
- Familiar with MATLAB and LabVIEW
- Experience with basic machining and metalworking techniques (e.g. milling, lathe work, and welding).
- Customer service experience

Extras
- Member of Tau Beta Pi Engineering Honors Society
- Boy Scouts of America Eagle Scout Rank
- National Honors Society Member
ANDREAS Q. QUAINOO

4296 Hidden Hills Dr. Apt 201 Kalamazoo, MI 49006 http://www.linkedin.com/in/andreasquainoo/
(412) 294-8316 andreas.quainoo@gmail.com

SUMMARY
Self-motivated, responsible and adaptable individual seeking a full-time position after graduation. Interested in working on a multidisciplinary team in a mechanical/aerospace related field with emphasis on current and emerging technologies that will further corporate goals.

EDUCATION
Bachelor of Science in Aerospace Engineering Graduation: December 2014
Western Michigan University Kalamazoo, MI
GPA: 3.95

PROJECTS
Box Beam Wing
• Designed a wing structure and performed FEA analysis using Abaqus to create the lightest possible structure that was able to withstand the applied loads in shear, bending, torsion, and buckling.

Three-Speed Transmission
• Designed a three-speed manual transmission and performed analysis using ROMAX that was as light as possible but could withstand the loads during one life cycle without failing at the bearings or shaft.

WORK EXPERIENCE

Teaching Assistant (Two Classes each semester) September 2013 – Present
College of Engineering and Applied Sciences – Mechanical and Aerospace Department
• Guide students in areas where they may have misunderstood the concept to ensure they learn from their mistakes and grade weekly quizzes and homework assignments of 30 to 60 students in assigned classes.
• Create and update grade sheets for students in the assigned classes to track their progress throughout the semester.

Peer Mentor and Co-Chair of the Training and Advisory Committee September 2012 – Present
Mentoring for Success (Division of Multicultural Affairs)
• Provide peer mentorship, academic review and tutoring to three to seven assigned students.
• Perform candidate interviews to ensure prospective peer mentors are the best fit for the job.
• Train new hires to teach them the knowledge and skills they will need to be successful in their new role.
• Administer, moderate and solve technical issues on the online forum consisting of more than fifty users to facilitate communication among employees and participants of our target and non-target population.

On-Board Diagnostics (OBD II) Calibration Intern May – August 2014
Chrysler Group LLC (Chelsea, MI)
• Analyzed the influence of different OBD demo prep procedures on tailpipe emissions for four engines using the Federal and Unified Test Procedures for several monitors in order to develop an efficient OBD prep procedure.
• Validated the effect of different drive modes on tailpipe emissions to reduce the number of drive modes that needed to be tested.

Vehicle Integration and Validation (VIV) Intern May – August 2013
Chrysler Group LLC HQ (Auburn Hills, MI)
• Assisted in the development and validation of two vehicle platforms to ensure launch readiness.
• Maintained and updated the corporate scorecard to assess the status of current vehicle compliance with a proposed standard that the competition has begun using.
• Planned and acquired data using USB Data Acquisition System (UDAS) on six vehicles during a thermal validation trip in order to verify that they exceeded the minimum federal requirements.

SOFTWARE & LANGUAGE PROFICIENCY

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<tr>
<th>Software</th>
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<tr>
<td>AutoCAD</td>
<td>Dreamweaver</td>
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<td>Abaqus</td>
<td>Java, C++, C#</td>
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LEADERSHIP, INVOLVEMENT AND AWARDS
Outstanding Student in Aerospace Engineering Spring 2014
Cataloger of Tau Beta Pi April 2013 – April 2014
Undergraduate Research Award Fall 2013 – Spring 2014
Secretary of Pi Mu Epsilon (Math Honor Society) September 2012 – December 2012
Cameron Segard
8995 Alanada Dr SE, Caledonia, MI 49316 Home: 616-891-3528 Cell: 616-481-9911
E-Mail: cameron.j.segard@wmich.edu
Character Traits: Strong Leadership, Creative, Team Player, Reliable, Honest, Detail Oriented,
Effective Communication Skills, Personable, Dedicated

Achievements
- WMU Dean's List: Fall 2012, Spring 2011
- Eagle Scout: February, 2009

Education
Western Michigan University- 1903 W. Michigan Ave, Kalamazoo, MI 49008: (269)-387-1000
Aerospace Engineering: Senior Status, GPA: 3.34, Expected Graduation: Fall 2014

Experience
Western Michigan University AIAA Pegasus Chapter: 2011-Present
- 2014 WMU AIAA Chapter President – Advises all AIAA chapter programs, coordinates volunteering events, and public speakers to provide educational opportunities to fellow students and our local community.
- 2013 DBF Manager - Greatly improved efficiency from past years with improved member utilization. Created an effective team structure with multiple positions with team leaders.
- 2012 DBF Construction Leader – Lead construction and installation efforts on all components of the aircraft

WMU Student Activities & Leadership Programs - LEAD Corp Intern: April 2012 to May 2014
- Leaders Unplugged Co-Facilitator—Co-Facilitated a five day backpacking trip to expose students to an alternate leadership experience on the trail without electronics.
- Emerging Leader Mentor - Encourage positive development of character and study habits for incoming freshmen and transfer students during their first year on campus.

Job Experience
Cessna Aircraft Company 2013 Summer Intern: 2013 May - July
- Citation Sovereign Control Systems Design Engineer - Projects include sustainment and certification in support of Del Devore under Joe Philips. Projects include: Control Column Redesign, Mach Trim Link, Spoiler Pushrod, and Rigging Specifications for all control systems with Tashi Sherpa.

Harvey Lexus, July, 2007 – August, 2012
- Customer Service Representative – Ensured the highest level of service possible by applying my extensive automotive knowledge, conversing ability, and attention to detail to every customer.

Memberships
- American Institute of Aeronautics and Astronautics (AIAA) 2008 – Present
- Academy of Model Aeronautics (AMA) 2006 – Present