WING MORPHING DESIGN UTILIZING MACRO FIBER COMPOSITE SMART MATERIALS

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Wing Morphing Design
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Nomenclature

\[
\begin{align*}
\delta_a &= \text{aileron deflection} \\
\alpha &= \text{angle of attack} \\
c &= \text{chord} \\
C_D &= \text{drag coefficient} \\
q &= \text{dynamic pressure} \\
I_x &= \text{inertia in roll} \\
\text{LE} &= \text{leading edge} \\
C_L &= \text{lift coefficient} \\
\text{L/D} &= \text{lift to drag ratio} \\
M &= \text{Mach number} \\
C_m &= \text{pitching moment coefficient} \\
P &= \text{pressure} \\
C_p &= \text{pressure coefficient} \\
\text{Re} &= \text{Reynolds number} \\
C_{\delta a} &= \text{roll moment with respect to aileron deflection} \\
C_{lp} &= \text{roll moment with respect to roll rate} \\
p &= \text{roll rate} \\
p_{ss} &= \text{roll rate (steady-state)} \\
S &= \text{surface area} \\
\text{TE} &= \text{trailing edge} \\
T &= \text{temperature} \\
T &= \text{time} \\
V &= \text{velocity} \\
b &= \text{wingspan}
\end{align*}
\]
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Wing Morphing Design
I. Project Requirements

Demonstrate the abilities of morphing materials technology by modifying a current remote controlled aircraft model using an electric propulsion system and designing and fabricating control surfaces that use Smart Materials micro-fiber-composites (MFCs). The design must be marketable in the R/C community.

The requirements for this project as described above contain basic guidelines for this Senior Design project. First and foremost, we must come up with a way to replace tradition servo controlled surfaces with MFC actuated surfaces in an effective manner. Not only will the control surfaces need to respond as well as, if not better than, the classic servo-motor system, but it will also need to be a fairly simple, potentially cost effective design that can easily be reproduced. The overall goal of this project is to showcase the capabilities of MFCs on an aircraft that could be reproduced on a large scale in today’s R/C market.

II. Smart Materials Application

A. Motivation

Current small radio-controlled aircraft typically use servomechanisms linked to independent control surfaces to provide flight control. A servomechanism consists of a large number of moving parts such as gears, bearings, shafts, motor coils, etc. The complexity of a servo requires a considerable percentage of the volume, power and weight of the entire aircraft. The overall aircraft performance is strongly dependent on these characteristics – volume, power, weight – therefore it is desirable to minimize the power consumption, volume occupied and weight of the control system. Another disadvantage of a servo actuated control system is the drag penalty that results from control linkages, horns, hinges and the discontinuities between the control surface and the stabilizer or wing. A more efficient control system for radio-controlled aircraft is desired.

B. Control System Requirements

To maximize efficiency the new control system should be simple (have a low number of component parts), lightweight, aerodynamically efficient and have low power consumption when compared to traditional servo-actuated control systems.

C. Use of Wing Morphing and ‘Smart Materials’ for a Control System

Wing morphing aircraft change their span-wise wing configurations or chord-wise airfoil cross-sections to optimize performance in various flight regimes. The morphing concept has proven to be successful on a number of
aircraft including the Wright Brothers’ first flyer in 1903 and a modern Northrop Grumman model of an unmanned combat aerial vehicle. The Wright Brothers and early airplanes used a series of pulleys and cables to twist the wing structure (wing warping) to achieve lateral control. More recently, engineers have tried to develop complex mechanical devices housed internally in the wing that change the airfoil or ‘smart materials’. The wing morphing concept has the potential to eliminate the need for a separate control surface, hinges, and control linkages, thereby greatly reducing the complexity of the design. Wing morphing also allows for a hinge-less design that can significantly reduce drag and increase aerodynamic efficiency. These characteristics make wing-morphing a viable solution to the control system requirements stated above.

To achieve morphing, an active or ‘smart material’ can be utilized. ‘Smart’ materials are solid-state devices, therefore they have no moving parts. Such devices can potentially be highly reliable. For example, one class of ‘smart materials’—the Macro-Fiber Composites—have shown “no reduction [in performance] to 100 million electrical cycles”.

D. Types of ‘Smart Materials’

There are several different types of ‘smart materials;’ however, shape-memory alloys and piezoelectrics are common in aerospace applications. Shape-memory alloys operate on the principle of the shape memory effect where changes in temperature as low as 10 degrees Celsius can result in a molecular re-arrangement of the material and thereby change the shape. The molecular arrangement is related to a specific temperature; such materials can effectively ‘remember’ their shape when heated or cooled.

Piezoelectrics are a different type of smart material with the property of transduction, where there is a coupling between the electrical and mechanical properties. The direct piezoelectric effect is the effect exhibited by piezoelectric materials where a mechanical pressure results in an electrical output of the material. The converse piezoelectric effect is the opposite; an electric field is applied that induces a material strain that can be used to actuate some device, such as morphing a wing. The direct piezoelectric effect can be used for sensing or energy harvesting. Piezoelectrics are polycrystalline materials, and the property of transduction is observed after they have been treated with an electric field at an elevated temperature (after the crystals have aligned). Applications of piezoelectrics often involve bonding the active material to a passive material. When there is only one active layer and one passive layer, this configuration is said to be unimorph. When two active layers ‘sandwich’ a passive layer, the configuration is said to be bimorph. The configuration used is dependent on the requirements of the application (i.e. a certain deflection is required). It is important to be aware that systems using piezoelectrics will exhibit hysteresis. Hysteresis is a property of a system where the output depends on the time history of the system. In other words, such systems are path dependant. The same voltage applied will not always result in the same deflection. Figure 1 below illustrates the hysteresis effect for a bimorph actuator – for a set of three different AC frequencies with a voltage varying uniformly with time (increasing than decreasing). The time path traced out by the solid lines shows that different deflections result at the same voltage. The deflection will always depend on the time history of the input signal.

E. Piezoceramics

Figure 1 Bimorph actuator hysteresis.
1. **Monolithic Piezoceramics**

Piezoceramics, which are ceramic based piezoelectrics, exist in several different configurations. The first to be produced and the most basic of these configurations is a monolithic piezoceramic (PZT). These monolithic piezoceramics have the advantages of being simple to use and relatively inexpensive to fabricate, but they are also very brittle and vulnerable to impact with other objects. Monolithic PZT fibers operate in d31 mode.

2. **Active Fiber Composites**

In an effort to combat these disadvantages, the Active Fiber Composite (AFC) was developed by researchers at MIT. Active Fiber Composites consist of round cross-sectional PZT fibers embedded into a soft epoxy matrix as depicted in Figure 2. On each side of the sheet-like configuration of PZT fibers lays a network of interdigitated electrodes, resulting in greatly increased in-plane actuation over standard PZT fibers. The interdigitated design also allows the use of d33 mode. Other advantages of the AFC include improved flexibility and increased robustness to damage. Despite the obvious advantages of the AFC over a standard PZT, it has its own disadvantages, including a high cost of production, a high voltage operating range, and inefficient electric field transfer due to poor surface area connectivity between the flat electrode and round PZT fiber surface areas.

![Figure 2 AFC configuration](image)

3. **Micro Fiber Composites**

The AFC design was further developed at the NASA Langley Research Center, resulting in the Micro Fiber Composite (MFC). The MFC is composed of a laminate of less expensive rectangular cross-sectional PZT fibers in an epoxy matrix and also uses interdigitated electrodes. The MFC includes all of the benefits experienced by the AFC in addition to a lower cost and improved electric field transfer due to increased surface area connectivity introduced by the rectangular PZT fibers.

![Figure 3 MFC configuration](image)

F. **Structural Requirements of a Morphing Wing for a Small R/C Aircraft**

Designing a morphing wing involves a unique set of structural requirements that are not traditionally considered in a wing design. The wing skin (the morphing component of the wing) should be anisotropic such that it has low in-plane stiffness and high out-of-plane stiffness. This will allow the skin to bend but will ensure it is still capable of transferring aerodynamic loads. Aero-elastic effects such as flutter, or limit-cycle oscillations may be a significant issue and need to be carefully analyzed. If a span-wise load bearing member is used, such as a wing spar or stiffener,
it should be compliant or use rigid-links to allow the wing skin to bend. Figure 4a and Figure 4b illustrate an
example of a morphing wing that uses a compliant wing spar. The compliant box as shown in Figure 4a has its top
and bottom surface free to translate and is fixed where the cross symbols are shown. The cross symbols represent the
fixed spar inside the compliant box that can be seen extending from the wing in Figure 4b. An alternate design limits
the active area of the wing to the trailing edge and instead uses a fixed rigid wing spar as shown in Figure 4
c.

![Morphing airfoil with compliant wing spar.](image)

b) Morphing wing with compliant wing spar.

In general the wing should be of relatively low weight and should be easy to manufacture, traditional rectangular shapes are preferred over delta, swept or twisted wings. It is also desirable to have removable outboard wing sections that can be replaced with traditional servo controlled wings and compared on the basis of performance and weight.

III. Configuration and Baseline Aircraft Choice

Based on initial configuration and sizing analyses conducted individually by team members for both electric ducted fan and aerobatic models, the following list of pros and cons for each configuration type was created as shown in Table 1 and Table 2 for ducted fan and aerobatic aircraft respectively.

A. Ducted Fan Aircraft

Table 1 Ducted Fan aircraft.

<table>
<thead>
<tr>
<th>Pros</th>
<th>Cons</th>
</tr>
</thead>
<tbody>
<tr>
<td>• Part of Park Flyer category</td>
<td>• Performance geared for high speed, not high maneuverability – May not show the full potential of an MFC control system.</td>
</tr>
<tr>
<td>• Appeals to a wide market</td>
<td>• Ducted fan system is more expensive than propeller</td>
</tr>
<tr>
<td>• Space for cargo in fuselage</td>
<td></td>
</tr>
<tr>
<td>• Belly lander (no landing gear required)</td>
<td></td>
</tr>
<tr>
<td>• Higher thrust capability – The ducted fan aircraft are capable of carrying more payload weight than equivalent propeller powered aircraft.</td>
<td></td>
</tr>
</tbody>
</table>

B. Aerobatic Aircraft
The propulsion system pros and cons for each R/C aircraft type do not show one as better than the other; the electric ducted fan system could be more expensive because of the fan, while the aerobatic configuration could be more expensive due to the engine thrust requirement. The team decided that the benefit of the aerobatic plane from the perspective of marketing the use of MFCs on a control surface outweighs any advantages in configuration and cost that might have been made with the electric ducted fan configuration. In order to properly size this configuration, information on the electronics payload must be investigated; this will be discussed in the following section.

C. Aircraft Determination

Once the baseline aircraft requirements and category was chosen, hobby websites were searched for aircraft that would meet the project requirements. The list was then narrowed down to 7 aircraft shown in Figure 5.

![Aircraft Selection Possibilities](Figures/selection.jpg)

Table 2 Aerobatic aircraft.

<table>
<thead>
<tr>
<th>Pros</th>
<th>Cons</th>
</tr>
</thead>
<tbody>
<tr>
<td>• Part of Park Flyer category</td>
<td>• Conventional landing gear (tail dragger gear) is necessary for durability – This is important to fit with the aerobatic park flyer market, but adds complexity to the part manufacture and assembly.</td>
</tr>
<tr>
<td>• Appeals to a wide market</td>
<td>• A more expensive engine may be required to get the necessary thrust from a propeller-powered system.</td>
</tr>
<tr>
<td>• Space for cargo in fuselage</td>
<td></td>
</tr>
<tr>
<td>• Propeller is cheaper than ducted fan</td>
<td></td>
</tr>
<tr>
<td>• Performance geared for high maneuverability – This should show the full potential of MFC actuated control surfaces.</td>
<td></td>
</tr>
</tbody>
</table>

The aircraft specifications were then put into a matrix to compare all of the possibly aircraft. Each aircraft specification was then normalized to the most desired value, giving each particular aircraft a score of up to “1” for each category considered. For example, a cheap aircraft will sell more than an expensive one, so each aircraft cost

---

4 Figures taken from towerhobbies.com
was normalized against the cheapest aircraft of the 7 choices and inverted to allow for the cheapest aircraft to achieve a score of “1”. The result of this normalization process is shown in Table 3. Once these values were calculated, the team ranked each category from most important to least important. Total cost was given the highest score because cheaper aircraft are much more appealing to modeler’s than expensive models. Fuselage volume was ranked as second in importance. Since the objective of the design is to demonstrate piezoceramic control surfaces, ample fuselage volume is required for housing the electrical components used to control the morphing surfaces. If there is limited space in the fuselage, there will not be many options for placement of these components, which may ultimately change the CG of the airframe, possibly causing the model to be statically unstable, resulting in an uncontrollable aircraft. Control surface deflections and roll rates were also ranked high in importance to the team. This is due to the limited actuation of the piezoceramics. If the morphing wings cannot achieve the deflections that standard servos offer, the aircraft may not perform as well as the conventional servo controlled model, resulting in an aircraft with performance characteristics not desired by R/C enthusiasts. The aircraft wingspan was rated as a less important feature of the aircraft. For experimental testing, the Virginia Tech Stability Wind Tunnel will be utilized. The tunnel offers a test section with a square cross section 6 feet on a side, which would accommodate all 7 models in consideration. The wing loading of the aircraft was also deemed less important. Because all of the models in consideration demonstrate high performance flying levels, a possible decrease in maneuverability due to design modifications would still result in a highly maneuverable model.

These weights were then multiplied by the normalized values obtained in Table 3 which gave the final aircraft scores for each aircraft and performance/model characteristic (shown in Table 4). The scores were then summed for each aircraft and the totals examined. Based on the decision matrix format, the aircraft with the highest score would be picked as the model best suited for the design criteria. From the score results, the Great Planes Edge 540 best matched the requirements for the project.
Table 3. Aircraft specifications normalization.

<table>
<thead>
<tr>
<th>Criteria</th>
<th>Desired</th>
<th>Baseline</th>
<th>Edge</th>
<th>SU-31</th>
<th>YAK-54</th>
<th>Escapade</th>
<th>Sequence</th>
<th>Sportster</th>
<th>Akrobat</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total Cost</td>
<td>low</td>
<td>$300.40</td>
<td>$305.40</td>
<td>$305.40</td>
<td>$479.30</td>
<td>$673.30</td>
<td>$362.61</td>
<td>$460.62</td>
<td></td>
</tr>
<tr>
<td>Normalized Cost</td>
<td></td>
<td>1.000</td>
<td>0.984</td>
<td>0.984</td>
<td>0.627</td>
<td>0.446</td>
<td>0.828</td>
<td>0.652</td>
<td></td>
</tr>
<tr>
<td>Roll Rate, degrees/second</td>
<td>high</td>
<td>462</td>
<td>378</td>
<td>361</td>
<td>462</td>
<td>226</td>
<td>227</td>
<td>229</td>
<td>207</td>
</tr>
<tr>
<td>Normalized Roll Rate</td>
<td></td>
<td>0.818</td>
<td>0.781</td>
<td>1.000</td>
<td>0.489</td>
<td>0.491</td>
<td>0.496</td>
<td>0.448</td>
<td></td>
</tr>
<tr>
<td>Control Surface Deflection, degrees</td>
<td>low</td>
<td>10</td>
<td>29</td>
<td>19</td>
<td>22</td>
<td>13</td>
<td>19</td>
<td>10</td>
<td>11.1</td>
</tr>
<tr>
<td>Normalized Control Surface Deflections</td>
<td></td>
<td>0.345</td>
<td>0.526</td>
<td>0.455</td>
<td>0.769</td>
<td>0.526</td>
<td>1.000</td>
<td>0.901</td>
<td></td>
</tr>
<tr>
<td>Wingspan, in</td>
<td>low</td>
<td>41.0</td>
<td>41.0</td>
<td>41.0</td>
<td>41.0</td>
<td>52.5</td>
<td>50.0</td>
<td>48.0</td>
<td>58.0</td>
</tr>
<tr>
<td>Normalized Wing Span</td>
<td></td>
<td>1.000</td>
<td>1.000</td>
<td>1.000</td>
<td>0.781</td>
<td>0.820</td>
<td>0.854</td>
<td>0.707</td>
<td></td>
</tr>
<tr>
<td>Wing Loading, oz/in^2</td>
<td>high</td>
<td>0.1822</td>
<td>0.0907</td>
<td>0.0904</td>
<td>0.0758</td>
<td>0.1822</td>
<td>0.1188</td>
<td>0.1253</td>
<td>0.1706</td>
</tr>
<tr>
<td>Normalized Wing Loading</td>
<td></td>
<td>0.50</td>
<td>0.50</td>
<td>0.42</td>
<td>1.00</td>
<td>0.65</td>
<td>0.69</td>
<td>0.94</td>
<td></td>
</tr>
<tr>
<td>Fuselage Volume, unit-less</td>
<td>high</td>
<td>8.62</td>
<td>5.65</td>
<td>3.53</td>
<td>3.53</td>
<td>3.31</td>
<td>8.62</td>
<td>3.62</td>
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<td>Normalized Fuselage Volume</td>
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<td>0.42</td>
<td>0.70</td>
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Table 4. Aircraft decision matrix.

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<tr>
<th>Characteristic</th>
<th>Reason</th>
<th>Weight</th>
<th>Edge</th>
<th>SU-31</th>
<th>YAK-54</th>
<th>Escapade</th>
<th>Sequence</th>
<th>Sportster</th>
<th>Akrobat</th>
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<tr>
<td>Cost</td>
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<td>8.3</td>
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<tr>
<td>Fuse. Volume</td>
<td>Component fit</td>
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<td>5.9</td>
<td>3.7</td>
<td>3.7</td>
<td>3.5</td>
<td>9.0</td>
<td>3.8</td>
<td>6.3</td>
</tr>
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<td>Deflection</td>
<td>MFC limits</td>
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<td>6.5</td>
<td>6.3</td>
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<td>3.9</td>
<td>3.9</td>
<td>4.0</td>
<td>3.6</td>
</tr>
<tr>
<td>Roll Rates</td>
<td>Marketable</td>
<td>6</td>
<td>2.1</td>
<td>3.2</td>
<td>2.7</td>
<td>4.6</td>
<td>3.2</td>
<td>6.0</td>
<td>5.4</td>
</tr>
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<td>Span</td>
<td>Testing</td>
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<td>2.0</td>
<td>2.0</td>
<td>2.0</td>
<td>1.6</td>
<td>1.6</td>
<td>1.7</td>
<td>1.4</td>
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<tr>
<td>Loading</td>
<td>Maneuverable</td>
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<td>0.5</td>
<td>0.5</td>
<td>0.4</td>
<td>1.0</td>
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<tr>
<td>TOTAL</td>
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<td>22.8</td>
<td>24.4</td>
<td>24.1</td>
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</tbody>
</table>

Wing Morphing Design
D. Baseline Aircraft Weights

Before the baseline model was assembled, each component was weighed and documented in Table 5. The list of weights will be maintained and updated as changes are made to the model to ensure that the gross maximum take-off weight is not exceeded.

Table 5. Edge 540 aircraft component weights.

<table>
<thead>
<tr>
<th>Component Description</th>
<th>Weight, oz</th>
<th>Component Description</th>
<th>Weight, oz</th>
</tr>
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<tr>
<td>Fuselage</td>
<td>6.150</td>
<td>Right Wing</td>
<td>2.011</td>
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<tr>
<td>Canopy</td>
<td>0.451</td>
<td>Left Wing</td>
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</tr>
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<td>Cowl</td>
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<td>Right Aileron</td>
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<tr>
<td>Landing Gear Strut</td>
<td>0.525</td>
<td>Left Aileron</td>
<td>0.670</td>
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<td>Wheel</td>
<td>0.067</td>
<td>Vertical Stabilizer</td>
<td>0.176</td>
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<tr>
<td>Axle</td>
<td>0.041</td>
<td>Rudder</td>
<td>0.353</td>
</tr>
<tr>
<td>Axle Nut</td>
<td>0.011</td>
<td>Horizontal Stabilizer</td>
<td>0.705</td>
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<td>Tail Skid</td>
<td>0.063</td>
<td>Left Elevator</td>
<td>0.247</td>
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<tr>
<td>Short Machine Screw</td>
<td>0.017</td>
<td>Right Elevator</td>
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<td>Small Wood Screw</td>
<td>0.004</td>
<td>Cowl Ring</td>
<td>0.106</td>
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<td>Flat Washer</td>
<td>0.008</td>
<td>Futaba S3114 Servo</td>
<td>0.275</td>
</tr>
<tr>
<td>Rare Earth Magnet</td>
<td>0.010</td>
<td>1250 mAh Lipo Battery</td>
<td>3.951</td>
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<td>CA Hinge</td>
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<td>Futaba Receiver</td>
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<td>0.007</td>
<td>Rimfire Brushless Motor</td>
<td>2.504</td>
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<tr>
<td>Wing Spar</td>
<td>0.635</td>
<td>Electrifly ESC</td>
<td>1.129</td>
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<tr>
<td></td>
<td></td>
<td><strong>Total Weight:</strong></td>
<td><strong>27.27</strong></td>
</tr>
</tbody>
</table>

IV. Electronics Background and Sizing Analysis

This section of the report will provide an overview of the electrical system within our RC aircraft, the function and placement of major electrical components, and finally the cost and sizing specifications of these components.

A. Overview

Before reading this section it is important to understand the basic electronic process taking place within the RC aircraft. Control signals will be sent over radio waves to the receiver located in our plane. These signals will be interpreted as a pulse width modulation (PWM) signal by the receiver and then sent to their perspective areas of the plane (i.e. motor, wing, rudder), whichever the user would like to control at the given time. Because the servo motors have been switched with MFCs, the signal must be converted for use on MFC patches. The PWM signal must be converted to a steady DC voltage, depending on the chosen duty cycle, and then that DC voltage must be amplified to the operational voltage for the Macro Fiber Composites (MFCs). The MFCs used in our design run at a particularly high voltage, low power input, so there are a couple steps that need to be taken to prime the signal for use. A diagram of the layout of electrical components can be seen below in Figure 6. It is important to note that each MFC control surface on the aircraft will need its own set of conversion components, thus making weight and sizing an important consideration in the design process.

![Figure 6 Electronic Configuration](image)

Figure 6 Electronic Configuration. Shows general electrical layout of MFC unique components within aircraft. It is important to note that the exact configuration will most likely be modified during the fabrication and testing phases.
B. Function of Components

This section will cover the function and purpose of the various electrical components as they are used in our aircraft.

1. PIC Microcontroller

After leaving the receiver the signal is first converted from a PWM to DC signal using a lowpass filter (covered in further detail below). The signal is then passed into a PIC microcontroller where it will be split into two separate signals, one for each MFC patch. The coding and purpose of the microcontroller can be found in section ___.

![Figure 7 PIC Microcontroller](image)

Figure 7 PIC Microcontroller. Shows a standard 16 bit PIC microcontroller that will be used to increase the accuracy and capabilities of the circuit.9

2. RC-DC Converter

The first step in the process is to convert the PWM signal to a steady DC voltage. The design group looked at a couple different ways of accomplishing this. The first option was to buy an “off the shelf” RD-DC converter. However, these tend to be relatively expensive and their capabilities far outmatch their use in this project. Because these filters consist of a relatively simple design, the decision was made to fabricate the circuit. After some research it became clear that a relatively simple loss pass filter would convert those sharp PWM frequencies into a steady DC signal. Depending on the desired response time and accuracy of the controller weight a pass or active lowpass filter can be used. For RC control requirements both types should achieve the desired accuracy needed for radio controlled flight. The different methods of converting the signal can be found below in Figure 8 a & b.

![Figure 8 Low pass filter schematic](image)

a) Passive Low Pass Filter. Shows a typical passive low pass filter, the design is simple however may lacks the necessary accuracy for our requirements.
b) Active Low Pass Filter. Shows a typical active low pass filter, the design includes the use of an operational amplifier, adding feedback control into the system.

Figure 8 Low pass filter schematic

3. Buffer Amplifier

The next step in the process is to increase the current flowing to the dc-dc converters by adding a buffer amplifier. The main concern here is that the DC-DC converter must be supplied with more current than is presently in the system. To solve this problem a single-transistor circuit can be used to provide the converted DC signal with current from an external source (the aircraft battery). A diagram of a basic common collector amplifier can be found below in Figure 9.
Figure 9 Common Collector Circuit. Shows a simple common collector amplifier. In this diagram $V_{BB}$ is the internally supplied voltage, $V_{CE}$ is the externally supplied voltage, and $R_L$ is the load resistance. Notice that the current across the load is shared by both the collector and the source.\textsuperscript{10}

4. DC-DC Converter

Finally, before the signal is ready for use it must reach the range of specified operating voltages for the MFC patches on the wing. Because the MFCs have a very high operating voltage (500 – 1500 Volts DC), the signal must be converter once more using a DC-DC converter. A DC-DC converter is a relatively simple device the amplifies a DC signal depending on a desired voltage and power output. The DC-DC converters that must be used for this application tend to be specific to MFC usage because of the low power consumption of the patches. The group has tested the lab converters and found that they still work and supply the specified proportional voltage gain. Several tests were run to compare the voltage input/output of several AM Power Systems DC/DC converters, the results of which can be found in Figure A2. After analyzing these results we found the AM 2505 to be best suited for our needs. The AM Power Systems DC-DC Converters used in our aircraft can be seen below in Figure 10.

Figure 10 DC/DC Converter. Shows AM Power Systems standard PCB mount DC-DC converter.\textsuperscript{11}

C. Performance Enhancing Solutions

To achieve the greatest resolution we chose to first convert the PWM signal from the receiver to a DC signal into the microcontroller. We used a common operational amplifier to increase the range of the signal using a resistor combination that can be found in Table A2. This allowed for the microcontroller to achieve the greatest accuracy possible while staying inside the limits specified in the PIC30F2010 data sheet. Another factor we had to take into consideration was weighing the importance of a clean, level signal against the need for a quick circuit response time. To do this we had to manipulate values of the capacitors found within the RC converters in the circuit. Smaller capacitance would lead to a quick response however the signal would be as clean as would be achieve with larger capacitors. A table with the results of several different capacitors and their response times can be found in Table A1. Another major concern is that these DC-DC converters do not allow a bi polar (both negative and positive) output, thus not allowing the wing to morph in two directions. To solve this problem a configuration has been designed which places two DC-DC converters in parallel and then the current is able to flow in either direction across the patch. The PIC microcontroller, mentioned above, will determine the strength of the signal sent to each converter. The amplitude of the converters will affect which way the current flows through the MFC patches, thus which way the patch will morph (either tension or compression). A diagram of parallel dc-dc converters can be seen below in Figure 11. Before the signal reaches in patches it will also pass through a voltage divider. This divider is a circuit consisting of several resistors and diodes that will cut the voltage down to the proper ratios as to achieve maximum deflection from the wings. Note that the voltage divider is not shown below for simplicity purposes.
D. Circuit Simulation

To ensure the accuracy of the bimorph/bipolar design mentioned above, the team used Multisim to virtually fabricate and simulate the performance characteristics of the system. The virtual model is important because it allowed the group to try several different designs with the labor constraints of “real life” construction. This virtual model is also important because it allows the user to calculate the resistance that must be placed across the wing. Choosing this resistance value is important because the MFC material has a high capacitance, so a change will be held in the wing until released. These “bleed” resistors allow for the energy to be dissipated from the wing rapidly, allowing a fast response time.

E. Fabrication

1. Initial Breadboard Design and Fabrication

After using Multisim to design and test the layout for the circuit, we then constructed a model on a standard 4” x 8” breadboard. The breadboard turned out to be extremely useful because we were able to try several different configurations noting the advantages and disadvantages of each. Another benefit of the breadboard was that we were able to interchange the values of several components to determine which would optimize our performance. Programming the microcontroller involved several cases of trial and error and the breadboard allowed us to swap in several different controllers without having to remove solder. However, the breadboard itself had several flaws that kept us from using it during actual flight. The first of which is that the breadboard is far too heavy for our situation. We would need four separate boards each weighing nearly 73.3 grams. This addition would be far greater than the maximum allowable weight as given by the aircraft specifications. Another major flaw with the breadboard is that components could be easily jostled loose from the board during turbulence in flight. We eventually concluded that while the breadboard was sufficient in determining design and tweaking component performance, we would need another method to convert our signal within the plane during flight. A picture of the breadboard can be seen below in Figure 12.
Figure 12 Initial Breadboard Design. Assembling the components on a breadboard allow us several advantages in the early fabrication phase. Keep in mind this circuit does not included the microcontroller or the voltage divider circuit.

2. PCB design and Fabrication

Once we achieved the desired performance from the circuit shown above on the breadboard we concluded the best solution for our needs would be to design and printed circuit board (PCB) that would fit our needs. To do this we utilized Express PCB™ fabrication software, which could be downloaded online and was free as long as the user ordered the board from ExpressPCB.com. A layout of the PCB using this design software can be found in Figure A1. Once the final circuit was designed four boards were ordered at a total cost of $176.00 ($44.00 per board). Each board had a silkscreen layer (to label where each component should be located) and a top and bottom layer for integrated wiring. Then all that was left to do was solder the separate components to the PCBs and then they would be ready for flight. The advantages on the PCBs were that they weighed 1.9 ounces each and all the components were soldered securely in place. A picture of an individual PCB can be seen below in Figure 13.

Figure 13 Final PCB. This figure shows the PCB as ordered from Express PCB.com. It is also important to note that there is further wiring underneath the board which can be seen in the schematic in Figure A1.

F. Ideas for Future Improvement

During flight our pilot commented that the control surfaces were slow to respond compared to other traditional RC aircraft. One reason for this is that the MFCs do not have enough current to respond as quickly as they do during lab test (in which current draw is essentially unlimited). To combat this problem several more DC/DC converters could be placed in parallel to allow more current to flow to the wings. Another component which slows the reaction time of the circuit is the low pass filter placed before the microcontroller. We added this filter to simplify the signal coming into the microcontroller; however it is not necessary to the circuit and would not adversely affect circuit performance.

Another, more radical, change would be adding an autopilot type system to the aircraft. To do this we could add gyroscopes and accelerometers to the control surface to measure the position and movement of the plane. This data would be interpreted by a central processor and that processor would respond with the correction due to the changing user input. The pilot would input angular rates and directions to the system and the system would respond by deflecting a certain surface until the desired rate was reached. Similar systems have already been implemented and seem to work very well.

G. Weight/Sizing

Because all three of the above components will be on the same board, their weights and sizes were able to be estimated as a single box. The board, with all components added, will weigh roughly 1.9 ounces and be $4.5 \times 2.1 \times 2$ inches squared. It is important to keep in mind that every control surface on the plane will require 1 of these boards each.
V. Microcontroller

DsPIC30f2010 Microcontrollers were used to interpret the signal from the aircraft receiver and control the output received by the MFC patches. The microcontrollers are programmed to receive a PWM signal from the receiver after it’s been passed through a low pass filter and been converted to a DC voltage ranging between 0 and 5V. Upon deflection of one of the transmitter control sticks, this voltage fluctuates towards a maximum value or minimum value depending on the direction of the stick deflection. The microcontroller then performs an analog to digital (A/D) conversion of this voltage.

The microchip then produces two PWM signals proportionate to the stick input. A 0% duty cycle of these signals feeds 0V into the circuit and represents no deflection of the MFC patch, while a 100% duty cycle feeds 5V into the circuit and represents maximum deflection of the MFC patch. These signals each feed into a different patch of the bimorph configuration. The bimorphs are set up so that each patch’s maximum range of deflection is in the opposite direction of the other, allowing for maximum deflection of the patches in both directions. This sets the criteria that the PWM signals must be inverse of each other. For example, if it is desired to deflect the bimorph configuration in an upward direction, you would want the patch with maximum deflection in that direction to deflect fully, while the other patch deflects in an inverse direction.

The initial voltage, PWM output relation was piecewise and is displayed in Figure 14 below. This relation proved to work, but induced a pause when passing from positive and negative deflection.

![Output PWM Signals vs. Received Voltage](image1)

Figure 14 Original piecewise linear PWM output relation.

In an effort to eliminate this lag, a fully linear output relation, displayed in Figure 15 below, was developed. This new relation removed the pause and allowed for smooth movement between positive and negative deflection.

![Output PWM Signals vs. Received Voltage](image2)

Figure 15 Modified fully linear PWM output relation.
As can be seen from the figures, the program also allows for minimum and maximum PWM output values to be defined. The minimum values are defined to continuously supply a minimum voltage to the DC/DC converters required for them to stay on. The maximum value is defined as a safety measure to ensure the patches do not receive voltages high enough to cause depolarization.

VI. Baseline Airfoil Analysis

XFOIL is a respected program created by Mark Drela at MIT to design and analyze 2D airfoil aerodynamic characteristics.\textsuperscript{12} It requires an input airfoil shape in x and y coordinates normalized to a chord length of one. It also requires the flight conditions of the aircraft in the form of the non-dimensional Reynolds number and Mach number. This program is used to produce results in for the baseline airfoil analysis.

A. XFOIL Validation of NACA 63-009

To use XFOIL, a validation of its accuracy was conducted by comparing wind tunnel results from Ref. 13 for the baseline airfoil section, NACA 63-009, currently used in the EDGE model. The experiment by the NACA was conducted at a Reynolds number of 5.8 million and Mach number of 0.167 using a 5 ft chord model that spanned the width of the 7 ft wind tunnel.\textsuperscript{8} The experimental results were plotted against the results produced by XFOIL for the same airfoil and flight conditions in Figure 16. The critical amplification ratio, $m$, used in XFOIL was set at four.

Stall was predicted by XFOIL to occur at $C_l = 1.326$ and $\alpha = 18^\circ$, while it was shown from testing to occur at $C_l = 1.06$ and $\alpha = 3.9^\circ$. This shows that XFOIL was valid to within 20% of the true values. Since XFOIL was originally designed for lower Reynolds number flow conditions, it may be that our results at lower Reynolds numbers are more accurate. However, until we can test this for ourselves, we must assume that XFOIL is only accurate at stall conditions to within 20%. Otherwise, the data seems to be in good agreement, and it should withstand analysis involving relative comparisons between airfoil shapes or flight conditions.

B. Flight Speed Analysis

The Reynolds numbers and Mach numbers relevant to our current baseline aircraft were determined by assuming a standard temperature and pressure in Blacksburg; the elevation in Blacksburg is 633 meters, and the temperature and pressure were calculated as $T = 284 K$, and $P = 93.2$ kPa. Flight velocities were initially assumed to be
between 40 and 60 miles an hour, so the Reynolds and Mach numbers corresponding to 40, 50, and 60 miles per hour were calculated. The corresponding Reynolds and Mach number pairs were 200,000 and 0.116, 260,000 and 0.148, and 310,000 and 0.174.

Figure 17 shows the results of analyzing the baseline 63-009 airfoil at the three Reynolds and Mach number combinations described above. They show that there is very little difference in lift and drag coefficients between these Reynolds numbers over a range of angle of attack. The differences are most notable at stall, but they are still very slight deviations from one another. From this information, it was determined that any further comparisons only need to be made between airfoil shapes, without too much concern for variations in the Reynolds and Mach number in flight. All further analysis is conducted at $Re = 310,000$ and $M = 0.174$.

C. Deflection Analysis

The following analysis of the baseline was conducted at $Re = 310,000$ and $M = 0.174$. Figure 18 shows information for the NACA 63-009 airfoil at 0, 11, and 20 degrees deflection; these deflections were chosen as they represent the maximum deflection for low and high rates on the current baseline aircraft. The pressure distribution looks as should be expected for a symmetric airfoil at 0, 11, and 20 degrees positive deflection.
Figure 19 shows the lift and moment coefficients plotted against angle of attack, as well as the drag polar plot for a range of angle of attack from -5 degrees to 15 degrees. The curves show a trend as expected for increasing deflection for a symmetric airfoil. The maximum lift coefficient and corresponding angle of attack for 0 degrees deflection, 11 degrees deflection, and 20 degrees deflection respectively are .86 and nine degrees, 1.17 and 5.5 degrees, and 1.32 and 5 degrees.

Figure 18 NACA 63-009 pressure distribution in three deflection configurations
The section lift coefficient plotted against angle of attack from -5 to 15 degrees.

Drag polar: section lift coefficient plotted against section drag coefficient from -5 to 15 degrees angle of attack.

The section moment coefficient plotted against angle of attack from -5 to 15 degrees.

Figure 19 NACA 63-009 aerodynamic data for varying deflection configurations.
VII. Variable Camber Airfoil Design

The utilization of a variable camber wing introduces complex airfoil design considerations. The morphing requirement calls for the airfoil to be highly flexible. Due to the limited actuation abilities of the chosen MFC smart material, as mentioned above, it is desired that the airfoil be thin as well, to maximize the deflection of the actuator. The aerobatic requirements of the aircraft call for the wing to be stiff in both torsion and bending as well as aerodynamically favorable. These structural properties favor a thicker and smoother airfoil. These conflicting design requirements cause need for a unique morphing wing design capable of being both flexible and sturdy.

In an effort to fulfill both of these requirements, an example airfoil configuration, found in Figure 20, is analyzed. The design consists of a thick and aerodynamically favorable leading edge which smoothly blends into a thin and flexible morphing trailing edge. With this design, both the morphing and structural requirements are fulfilled.

To determine the best shape for the airfoil, a program was developed in which key parameters could be defined and coordinates for the airfoil output. These parameters include properties such as the leading edge NACA profile, the length of the thick leading edge, the length of the morphing trailing edge, the thickness of the trailing edge, and the input voltage for smart material actuation.

To create the airfoil, the program first outlines the NACA profile, splices it at a user defined fraction of the cord, and then resizes the remaining section to the desired leading edge length. Next, the trailing edge is outlined and sized and deflected based upon the user input, length, thickness, and actuation voltage respectively. The program then uses a spline interpolation to create the smooth transition between the leading and trailing edge, completing the airfoil. This airfoil creation process is displayed in Figure 21. The coordinates for these airfoils are then output into a .dat file in proper input format to be read by XFOIL.

![Figure 20 Sample airfoil with morphing trailing section.](image)

![Figure 21 Airfoil program creation process.](image)
For the initial airfoil design, symmetrical NACA profiles with thicknesses between 4-15% were investigated. Using a fixed trailing edge length of 4.4” and thickness of 0.032”, the airfoils were analyzed at the root, mid-span, and tip of the wing. The leading edge length was defined as 0.78 chord of the remaining chord not taken up by the trailing edge, with the other 0.22 chord used to transition between the leading and trailing edge. -1400V, 0V, and 1400V actuations were investigated for each thickness and location, representing maximum negative, zero, and maximum positive deflection respectively. These maximum and minimum voltage values correspond to an approximate 5% camber. A sample initial airfoil design for a NACA 0011 at the root chord and 1400 V actuation is shown in Figure 22.

![Figure 22 Initial Airfoil Design.](image)

VIII. Variable Camber Airfoil Parametric Study

A parametric study was conducted for symmetric airfoils with thicknesses ranging from 4-15%. The goal of the study was to determine any problems with the airfoil and variable camber trailing edge configuration and to match the aerodynamic characteristics as they currently are for the baseline airfoil. This is necessary to assure the aircraft is able to fly with the same quality whether using a morphing wing or tradition wing. General comparisons were made, and detailed analysis of the chosen airfoil was conducted.

A. Pressure Coefficient Comparison

The following section shows results for the pressure distribution over the morphing airfoil for zero deflection and maximum deflection corresponding to 1400 V. Results are shown for the pressure distributions at zero angle of attack (i.e. $C_l = 0$). The analysis was conducted at $Re = 310,000$ and $M = 0.174$ corresponding to a flight speed of sixty miles per hour.

Figure 28 shows the pressure coefficient distributions for the series of morphing airfoils tested. The pressure coefficient plot as compared with the airfoil plot seems to show an appropriate distribution over the airfoil, and they should provide proper results. It can be seen that more lift can be generated with higher the airfoil thicknesses. This will be important to achieve considering the weight demands of the aircraft.
B. Aerodynamic Characteristics Comparison

Figure 24 shows the lift and moment coefficients plotted against angle of attack, as well as the drag polar plot. The results are shown for angle of attack ranging from -5 to 15 degrees for airfoils ranging in thickness from 4-15%. Each plot shows trends relating to airfoil thickness. It should be expected that drag coefficient values are larger for larger thicknesses due to larger separation towards the trailing edge. Also, stall becomes more drastic as thickness increases; however, the maximum lift coefficient also increases.
a.) Zero deflection case: section lift coefficient vs. angle of attack.

b.) Zero deflection case: drag polar.

c.) Zero deflection case: section moment coefficient vs. angle of attack.

Figure 24 Morphing airfoil aerodynamic data for zero deflection case for $Re = 310,000$. $Re = 310,000$. 
Table 6 below shows further detail into the study. The table shows that a tradition lift curve slope cannot exist for thicknesses less than nine percent with zero degrees deflection; therefore, only thicknesses of nine percent and greater will have reliable stall characteristics. A thickness upwards of 12% will be best structurally due to the spar tube thickness limitations. Since it is shown that thicker airfoils generate more lift and stall later, the NACA 0015 adjusted airfoil will be best for our purposes.

**Table 6 Numerical aerodynamic results.** Italicized results indicate baseline airfoil data.

a.) **Zero deflection numerical aerodynamic results.**

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<th>NACA</th>
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<td>-5.0</td>
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<tr>
<td>0013</td>
<td>1.65</td>
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<td>0.011</td>
<td>-5.0</td>
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<tr>
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<td>0.012</td>
<td>-4.5</td>
</tr>
<tr>
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<td>1.72</td>
<td>6.5</td>
<td>0.012</td>
<td>-5.0</td>
</tr>
<tr>
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<td>1.17</td>
<td>5.5</td>
<td>0.0105</td>
<td>-1.0</td>
</tr>
</tbody>
</table>

b.) **Maximum deflection numerical aerodynamic results.**

<table>
<thead>
<tr>
<th>NACA</th>
<th>Clmax</th>
<th>αmax(°)</th>
<th>Cdmin</th>
<th>αmin(°)</th>
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<td>0.009</td>
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<td>0.009</td>
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<td>0.011</td>
<td>-5.0</td>
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<td>1.60</td>
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<td>0.011</td>
<td>-5.0</td>
</tr>
<tr>
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<td>1.65</td>
<td>6.0</td>
<td>0.011</td>
<td>-5.0</td>
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<tr>
<td>0014</td>
<td>1.69</td>
<td>6.5</td>
<td>0.012</td>
<td>-4.5</td>
</tr>
<tr>
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<td>1.72</td>
<td>6.5</td>
<td>0.012</td>
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<td>1.32</td>
<td>5</td>
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<td>-1.5</td>
</tr>
</tbody>
</table>

C. **Airfoil Design Analysis**

The analysis of the NACA 0015 adjusted morphing airfoil was conducted with a 320 panels instead of 160 panels for higher fidelity modeling of our chosen design. A turbulence value of \( \nu = 9 \) was used to simulate the laminar flow expected over the wings.

1. **Pressure Coefficient**

The following section shows results for the pressure distribution over the morphing airfoil for 0, 500, -500, 1500, and -1500 Volts deflection. Results are shown for pressure distributions at zero degrees angle of attack. The analysis was conducted at Re = 304.000 and M = 0.129, corresponding to a flight speed of sixty miles per hour. Figure 25 shows the pressure coefficient distributions for the NACA 0015 morphing airfoil.
2. Aerodynamic Characteristics

Figure 26 shows the lift and moment coefficients plotted against angle of attack and the drag polar plot for the root chord and tip chord airfoil cross sections. Data was also collected from the zero chord and mid chord cross sections for stability analysis purposes. The results are shown for angle of attack ranging from -6 to 12 degrees in one degree increments for the NACA 0015 adjusted morphing airfoil.
A. Development of the Aerodynamic Model

The stability and control of the baseline and morphing Edge 540 remote controlled aircraft was analyzed using the Athena Vortex Lattice (AVL) code developed by Mark Drela and Harold Youngren at the Massachusetts Institute of Technology. The AVL program is used to analyze rigid body aerodynamics and flight dynamics of aircraft.
Aircraft. Aerodynamic properties are determined using an extended vortex lattice model for lifting surfaces. Flight dynamics are determined by linearization of the aerodynamic model for a specified trim condition. The aerodynamic model of the lifting surfaces in AVL can be defined in several ways. An airfoil coordinate file may be specified from which the mean camber line is computed. The airfoil is therefore approximated by a series of thin flat plates and potential flow vortices. This is an inviscid solution. A more refined aerodynamic model can be specified by inputting parameters the lift curve (CL vs. alpha), this is the method used in the present analysis.

B. Configuration Geometry

The AVL model used in this analysis consists of a main wing, horizontal tail and a vertical tail. Both the baseline and morphing configurations are studied. The main wing is defined using four cross sections. Analysis has shown that increasing the number of cross sections defining the main wing from two to four had virtually no impact on the results (< 0.1%). At each station the lift curve slope is adjusted to the value found from the Xfoil analysis and changes linearly from one station to the next. The wing-fuselage is approximated by a 5 in. section compromising a varying camber shape. The shape is an interpolation from the left wing root camber, to a midpoint section defined by a flat plate, to the opposite right wing root camber. This configuration is chosen in order to allow the left and right wing sections to be actuated in an opposite sense. In other words, this prevents a discontinuity in the camber line variation at the midpoint. The horizontal tail is modeled using three sections. The vertical tail is modeled using two sections. The locations of the leading edge of each lifting surface and their respective chord lengths were taken from the CAD model and were used to define the aircraft geometry. The reference point for the aircraft was assumed to be the front of the engine cowl. The center of gravity of the aircraft ready to fly was found to be 9.6 in. aft of the reference point. The final AVL geometry is shown in Figure 27.

C. Wing Aerodynamics

The following figures show that the wing aerodynamics of both the baseline and morphing configuration. The drag polar (Figure 28) indicates that the performance of both wings is very similar. The morphing configuration results in a slightly higher lift curve slope (Figure 29). The similarity of the results is to be expected since both airfoils are relatively thin and have approximately the same leading edge radius. The wing sections of the AVL model were defined solely on the basis of the mean camber line and lift curve slope. The vortex lattice analysis does not handle viscous effects – the expected drag reduction from a smooth morphed control surface over a protruding servo-aileron is not represented here (Figure 30).

![Figure 27 AVL model of the Edge 540](image)

![Figure 28 Drag polar for morphing and baseline configurations.](image)
D. Pitch Static Stability for Symmetric-Section Wing

Figure 31 shows the variation of the pitching moment of the aircraft with angle of attack for various elevator deflections or voltages. The wing is assumed to be in its un-deflected configuration at 0V. In both the morphing and baseline case, the analysis indicates that the elevator has strong authority and only small deflections will be necessary to trim the aircraft. The abundant pitch control is required for acrobatic flight.
In the event that the lift coefficient of the un-actuated wing section is inadequate, the trimmed flight condition may require the wing to be operating at a non-zero voltage, to produce more lift. In this case, the elevator must still be able to trim the aircraft. This problem is approached by considering the extreme case where the wing is fully actuated (+1500V) to provide maximum lift, and the elevator is shown to be capable of trimming the aircraft.

![Figure 32 Pitching moment for fully actuating morphing wing](image)

**E. Steady State Roll Rate**

The roll rate was found experimentally by studying a video of another Edge 540 (an existing aircraft used to film promotional video for manufacturer) in flight, performing steady state roll maneuvers. It is assumed the aircraft in the video was using the highest possible aileron deflections also known as "3D roll rates." Because this was a promotional video, and the aircraft rolls appeared very quick it is reasonable to assume 3D rates were used. A whole number of rolls, n, were counted and the elapsed time, t, were recorded with a stopwatch. The roll rate, $p$, was found (in deg/s) using the equation $p = \frac{360n}{t}$. The results from this testing are shown in Table 7 below. The average steady state roll rate was found to be about 377 deg/s.

**Table 7 Experimental results of steady state roll rates**

<table>
<thead>
<tr>
<th>Number of 360 Flips</th>
<th>Elapsed Time (seconds)</th>
<th>Roll Rate (deg/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Trial 1</td>
<td>7</td>
<td>3.44</td>
</tr>
<tr>
<td>Trial 2</td>
<td>11</td>
<td>5.90</td>
</tr>
<tr>
<td>Average</td>
<td></td>
<td>377.2</td>
</tr>
</tbody>
</table>

Analytical expressions for the roll rate are fairly simple and are discussed below. The transient roll response to a step aileron input assuming one degree of freedom in roll, $p$, is governed by Equation 1:

$$p = -\frac{2V C_{l_{\delta a}}}{b C_{l_{\rho}}} \delta a (1 - e^{L_{\rho} t})$$

(1)

where:

$$L_{\rho} = \frac{g b^4}{2V^2 \delta a}$$

(2)

where $b$ is the wingspan, $V$ is the airspeed, $C_{l_{\delta a}}$ is the roll moment with respect to the aileron deflection, $C_{l_{\rho}}$ is the roll moment with respect to the roll rate, and $\delta a$ is the aileron deflection. Note that the exponential term is the transient component, and it is proportional to the inertia of the aircraft in roll. The range of aileron deflections is known and is summarized in Table 8 below for each "rate setting" (controlled by the pilot).

**Table 8 Baseline aileron deflection for various settings (travel / deflection)**

<table>
<thead>
<tr>
<th>Low Rate</th>
<th>High Rate</th>
<th>3D Rate</th>
</tr>
</thead>
<tbody>
<tr>
<td>16 mm / 11 deg</td>
<td>29 mm / 20 deg</td>
<td>44 mm / 33 deg</td>
</tr>
</tbody>
</table>
The wingspan of the Edge 540 is $b = 41$ in. and the cruise speed is approximately $V = 47$ ft/s. The stability derivatives are estimated using the Athena Vortex Lattice (AVL) code and the analytical steady state roll rate is computed using Equation 3 below which does not include the transient component.

$$\rho_{ss} = -\frac{1}{b} \frac{C_{l_{ss}}}{C_{p_{\alpha}}} \Delta \alpha$$

(3)

It was found that the morphing aircraft had a slightly higher maximum steady state roll rate, however roll performance appears to be essentially unchanged (Figure 33). Again, note that the anticipated benefits of the smooth flow resulting from morphing wings over servo-actuated wings will not show up in these inviscid solutions, since they do not account for flow separation and viscous effects. The graph below is in terms of percent actuation where the baseline is assumed to have a maximum actuation of 20 deg. aileron and the morphing wing has a maximum actuation of +1500V.

![Figure 33 Theoretical steady state roll rates](image)

A. Edge 540 - Iteration 1

Prior to developing a CAD model, individual aircraft components were photographed to assist with dimensioning the CAD model. The photographs were then imported into Matlab and a digitizing code was used to accurately measure the dimensions of the various aircraft components. Once the image is imported into the Matlab figure, the user defines the axes. This was done by selecting two points along the horizontal part of the L-square in the image to define the x-axis and two points along the vertical part of the L-square to define the y-axis. This becomes the basis for referencing the location of the points selected by the user. Points can be selected at various locations on the image which can then be exported to a separate Matlab figure or to a .dat file. The resulting figure or .dat file can be used to determine the desired lengths of the aircraft components which can then be used to dimension the CAD model. Measurements obtained through this method are only as accurate as the ruler used to reference the components, thus all measurements obtained from the digitized photographs were approximated to the nearest 1/16th of an inch. Figure 34 below shows an example of a photograph used in conjunction with Matlab.
In addition to the results of the digitization process, a technical drawing obtained from Great Planes was utilized to aide with dimensioning when creating the 3 dimensional CAD model. The technical drawing includes cross-sectional views throughout the fuselage, a feature not obtainable from the digitization process. Figure 35 represents the fuselage portion of the technical drawing provided by Great Planes.

From the measurements obtained from the technical drawing and digitized photographs, a 3 dimensional CAD model was constructed. Due to the high detail of the technical drawing and complex structure of the aircraft, the resulting model was simplified, keeping only the critical features of the model. Aerodynamic features were modeled as closely as possible to the original drawing, while the interior and structural detail shown in the technical drawing was eliminated.

The CAD model will be used to show the various changes that will be implemented on the aircraft to accommodate the piezoceramic lifting surfaces and control components. Figure 36 represents the first iteration CAD model in its standard configuration with no modifications.

Black box representations of the various control components that will be used to control the model with either the servos or piezoceramics were also created in SolidWorks. The servo representations were then placed in the model assembly to show their physical locations on the airframe. The piezoceramic control models were then placed in the model’s fuselage to simulate the would-be location for the components when the aircraft is flying with
the morphing configuration. An RC/DC converter is composed of all of the electronic components necessary to control one morphing surface. While the final placement design for these components is still in the design phase, the current model dimensions are 5.25" x 1.75" x 0.75". The CAD model for the MFC Power Electronics is shown in Figure 37 and the full aircraft with the piezoceramic and servo control component locations is shown in Figure 38.

![Figure 37. MFC power electronics.](image)

![Figure 38. Aircraft control component locations.](image)

The CAD model was also used to estimate the moments of inertia of the full aircraft. This was done by assigning each component of the CAD model a weight based on the weights obtained in Table 5, then allowing the built-in calculator of SolidWorks to obtain the values for the moments. This information was then used to run aerodynamic simulations discussed later. The recorded moments of inertia are located in Table 9 and the location of the axes about which the moments are recorded is shown in Figure 38.

### Table 9. CAD model moments of inertia about CG.

<table>
<thead>
<tr>
<th>Moment</th>
<th>Positive Direction</th>
<th>Direction</th>
<th>Value, lbs-in²</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ixx</td>
<td>Out nose of aircraft</td>
<td>Roll</td>
<td>59.9</td>
</tr>
<tr>
<td>Iyy</td>
<td>Out starboard wing</td>
<td>Pitch</td>
<td>120.1</td>
</tr>
<tr>
<td>Izz</td>
<td>Out bottom of fuselage</td>
<td>Yaw</td>
<td>173.6</td>
</tr>
</tbody>
</table>
B. Edge 540 – Iteration 2

To better represent the baseline aircraft, the CAD model of the fuselage was redesigned. Changes to the original model include: realignment of bulkheads, detailed firewall/engine cowl section and a detailed representation of the avionics compartment of the fuselage. The bulkheads were realigned to match the actual aircraft, maintaining the full-length continuous belly of the fuselage (Figure 39). The baseline servo actuated aircraft is shown in Figure 40.

![Figure 39 Original fuselage (top) and updated fuselage (bottom).](image1)

![Figure 40 Full baseline aircraft.](image2)

C. Edge 540 – Iteration 3

After completion of the baseline CAD model, the morphing lifting surfaces were designed and modeled. The optimum airfoil shape determined from the aerodynamic analysis codes was imported into SolidWorks to form the basic wing structure. Leading and trailing edge stringers as well as the main wing spar were added to the wing model. The bimorph MFC assembly was modeled by creating a model of the MFC patch and stainless steel shim substrate, then combining the two parts into a single model based on the design configuration.

To allow for removal and replacement of MFC patches in the event of a malfunction, a bolt clamp mechanism was added to the trailing edge of the rigid wing structure. If a patch was to fail, the clamp would be loosened, the faulty patch removed and replaced, and the clamp would be retightened. The primary job of the clamp mechanism was to eliminate the chance of destroying critical structural elements of the wing during patch replacements. The final wing model is shown in Figure 41.

![Figure 41 Final wing model.](image3)
To assist with fabrication of the wing, a wing rib jig was modeled and laser cut. The jig would ensure that both wings are built symmetrically to avoid aircraft trim issues during flight. The jig model is shown in Figure 42.

The horizontal stabilizer was designed to resemble the baseline stabilizer as closely as possible. The planform area of the rigid stabilizer portion was modeled with dimensions taken from the baseline stabilizer. To accommodate the MFC bimorph, the stabilizer was split into three layers: two outer layers and a single middle layer. A channel 0.75 inches wide was removed from the middle layer so the MFC bimorph could be joined to the stabilizer. Each of the three layers and an MFC bimorph layer were modeled and assembled in SolidWorks prior to fabrication. The same three-layer technique was used to model the vertical stabilizer, both of which are shown in their proper configuration in Figure 43.
Once each component was modeled, a full aircraft assembly was created to show the full morphing aircraft configuration. This model is shown in Figure 44.

D. **Edge 540 – Iteration 4**

The tail was redesigned due to the large moment created from the additional weight of the MFC surfaces. This large moment caused the aircraft to be statically unstable, a condition that makes the aircraft difficult to control in flight without the assistance of a computer control system. To reduce the weight of the tail, six of the ten MFC patches were removed from the vertical stabilizer and four of the twelve patches were removed from the horizontal stabilizer and four patches along the tips of the stabilizer were replaced with smaller patches. The redesigned MFC patch layout is shown in Figure 45.
The stainless steel substrate in between the patches on the wings and tail was removed and replaced by a latex membrane. The two-layer stainless steel rudder extension was also replaced with a two layer fiberglass substrate to further reduce the weight.

The wing bolt clamp structure was removed and replaced by a permanent glue clamp mechanism. The bolt clamp mechanism does not provide a transfer of shear forces between the MFC patches and the rigid wing structure which can cause the connection to “give” allowing the morphing portion of the wing to rotate under load, lowering the effective camber. This problem is corrected with the glue clamp mechanism because the shear forces are now transferred into the rigid wing structure. Figure 46 represents the updated morphing wing.

To make balancing the aircraft for flight simpler, the nose of the aircraft was extended. This allowed for more space between the motor, electronic speed control, battery and electronic converters, promoting better cooling due to the increased airflow over the components. The redesigned firewall is shown in Figure 47 and full aircraft showing the placement of electronics is shown in Figure 48.
A. Bonding Procedure

For Macro-Fiber Composites (MFCs) to be properly used in control surface actuation, a particular bonding process is necessary. Before beginning the bonding process, MFCs were checked to make sure they worked and that they could achieve their full voltage potential. The MFCs were connected via an amplifier to a Labview program that generated a sinusoidal voltage input to the patch as shown in Figure 49. The DC voltage was incremented in 10 Volt increments to 750 Volts and then the AC voltage was added for a 1500 Volt peak-to-peak actuation.
The MFC patches must be bonded to a thin, pliant substrate in a bimorph configuration to allow for maximum deflection. A .001 inch thick stainless steel substrate was used based on experimental data conducted in the CIMSS Lab (REF). It is important that the patches are aligned correctly with respect to each other so that the entire surface deflects evenly. A procedure was developed to ensure these qualities were maintained in each control surface.

1. Clean the work area.
2. Clean the Sheet Metal Base (stainless steel substrate) with rubbing alcohol and a scraper.
3. Cover the Sheet Metal Base with the Vacuum Bag using masking tape.
4. Kapton tape the substrate to the reverse side of the sheet metal base as shown in Figure 50.

5. Place the MFC’s on the substrate and measure and mark their locations.
6. Tape the MFC’s so they can flip down onto the substrate.
7. Clean the steel substrate and MFC’s using alcohol.
8. Mix Epoxy in the mixing cup.
9. Cover the MFC with epoxy (flipped-up side) using the stir stick.
10. Add more epoxy on the inactive areas.
11. Flip MFC down onto substrate as shown in Figure 51.
12. Tape free end of the MFC.
13. Check Alignment.
14. Apply the Peel-Ply followed by the Felt.
15. Place the assembly into the vacuum bag.

16. Seal the remaining edge of the vacuum bag.
17. Apply the vacuum.
18. Squeeze out excess epoxy from each patch using the tool shown in Figure 52.

B. Wings – Iteration 1
Construction of the wings began with laser cutting the ribs and wing building jig from 3/32” light-ply wood. The ribs were glued to the outer portion of the I-beam wing structure, composed of two 1/8” x 1/8” balsa sticks, using medium cure cyanoacrylate (CA) glue. The inner I-beam supports were cut from 1/16” balsa sheet and were fixed to the I-beam sticks with medium cure CA. Figure 53 shows the completion of the wing fabrication thus far.
Strips of 1/16” sheet balsa were cut 2” wide and glued with medium cure CA to the top and bottom of the wing flush with the leading edge of the ribs. A balsa block was added to the leading edge of the ribs to form the leading edge of the wing. The additions of these two features are shown in Figure 54.

The trailing edge balsa sheeting and basswood clamp mechanism were installed along the trailing edge of the wing structure. This process is shown 50% complete in Figure 55.
Excess material was trimmed from the root and tip of the wings and 1/16” x 1/4” balsa spacers were added to each rib connecting the leading edge sheeting with the trailing edge sheeting. These spacers act as a contact point for the plastic film covering which was applied later. Figure 56 shows the wing fabrication thus far.

Final fabrication of the wings included forming the leading edge of the wings and installing the wing spar tube and wing lock mechanism. The wing lock mechanism was laser cut from 3/32” plywood and the spar tube was made from a fiberglass/epoxy matrix. Both components were glued to the wing spars using 6 minute epoxy to ensure a strong bond and transfer of aerodynamics loads to the fuselage. The final Wing skeleton is shown in Figure 57.
The MFC patch assembly was installed by sliding the bimorph into the slot along the trailing edge of the wing structure. Through holes were drilled along the top of the trailing edge to accommodate the screw clamp mechanism. Holes were also cut through the trailing edge sheeting to allow access to the solder points on the MFC patches. Both the top and bottom MFC patches were soldered in parallel with each other. Figure 58 shows the addition of the MFC bimorph structure to the wing structure.

C. Tail – Iteration 1

Prior to assembly of the stabilizers, each of the layers for the horizontal and vertical stabilizers was laser cut from 1/16” balsa sheets. A layer of fiberglass was added to the outward facing side of the outer stabilizer layers using a vacuum bag technique to produce a strong matrix with low weight. The purpose of the fiberglass layer is to create a shell structure to distribute aerodynamic forces along the outside of the stabilizers. Figure 59 shows the fiberglassed stabilizer outer layers and middle layers. The vertical and horizontal stabilizer was constructed using the technique outlined below.
Assembly began by placing one outer stabilizer layer fiberglass layer down on a building board. A 1/32” thick balsa sheet spacer was glued along the length of the trailing edge of the stabilizer using ScotchWeld DP460 epoxy. The elevator MFC bimorph structure was glued on top of the spacer using the ScotchWeld epoxy. This assembly is shown in Figure 60.

The MFC control wires were then routed through the middle layer of the stabilizer assembly and tack glued in place using thin cure CA. The middle stabilizer layer and top balsa spacer were then epoxied in place using ScotchWeld (Figure 61). A final layer of ScotchWeld epoxy was applied to the top of the middle stabilizer layer and the second outer stabilizer layer was set in place (Figure 62).
The entire stabilizer assembly was weighted down using steel and allowed to cure for 24 hours (Figure 63).

D. Wings – Iteration 2

As described in the section “Edge 540 – Iteration 3” the wing bolt clamp mechanism was replaced with a permanent ScotchWeld DP460 epoxy clamp. This was done by removing the trailing edge sheeting from both sides of the wing and constructing new replacement sheeting and basswood clamps. The outer surface of the sheets was fiberglassed to provide additional strength. A layer of ScotchWeld was spread on the inside of the trailing edge sheet at the rib contact points and put in place on the wing. The MFC bimorph was reinstalled in the trailing edge of the wing structure using a layer of ScotchWeld on both the top and bottom surface of the bimorph, followed by the second trailing edge sheet assembly. The full wing assembly was then weighted and allowed to cure for 24 hours.

Once the wing was cured, the stainless steel substrate between the MFC patches was removed and replaced with latex patches. This was done to eliminate stress on the bimorph structure during bending as well as reduce the overall weight of the wings. Finally, the rigid structure of the wing was covered using TopFlight Monokote, and the step gap between the MFC bimorph structure and rigid wing trailing edge was smoothed using silicone caulk. The final wing is shown in Figure 64.
E. Tail – Iteration 2

Due to the large moment created by the heavy horizontal and vertical stabilizers, the MFC patch layout was redesigned and fabricated. The stabilizer fabrication process follows the same steps and described in “Tail – Iteration 1” with the addition of weight reducing measures. Once the stabilizers were assembled and cured, the stainless steel substrate between the MFC patches was removed and replaced with a latex membrane. As with the wings, this was done to reduce the stress on the bimorph structure as well as reduce the overall weight of the stabilizers. The stainless steel rudder extension was also removed and replaced by a two layer fiberglass extension to reduce the weight of the vertical stabilizer. The completed stabilizers are shown in Figure 65.

![Completed morphing stabilizers](image)

Figure 65 Completed morphing stabilizers

F. Fuselage - Iteration 2

To assist with balancing the aircraft and to promote better cooling of the electronic components during flight, the engine firewall was extended. As described in the CAD model section, the extension was created using SolidWorks and laser cut from 3/32” plywood. The baseline firewall was removed and the new firewall was installed using 6 minute epoxy to ensure a strong bond. The new firewall is shown on the fuselage in Figure 66.

![Fuselage firewall extension](image)

Figure 66 Fuselage firewall extension
G. Center of Gravity and Weight Evaluation

After completion of the Phase I design, a center of gravity analysis was conducted by comparing major components of the baseline and morphing aircraft. It was found that the Phase I morphing aircraft was unstable. In order to exactly match the center of gravity from the baseline, the Goal Seek function in Excel solved for the necessary fuselage center of gravity. The results are shown in Table 10 below.

Table 10. Center of gravity calculation and comparison between the Baseline and Phase I Morphing aircrafts is shown. Green represents the desired stable CG, red denotes unstable center of gravity, and orange represents Phase I Morphing components.

<table>
<thead>
<tr>
<th>Component Description</th>
<th>Baseline Weight (oz)</th>
<th>Baseline CG (in from tip)</th>
<th>Baseline Moment (oz-in)</th>
<th>Morphing Weight (oz)</th>
<th>Morphing CG (in from tip)</th>
<th>Morphing Moment (oz-in)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>21.2</td>
<td>8.9</td>
<td>188.2</td>
<td>31.5</td>
<td>5.6</td>
<td>175.9</td>
</tr>
<tr>
<td>Starboard Wing</td>
<td>2.6</td>
<td>11.5</td>
<td>30.0</td>
<td>7.6</td>
<td>12.0</td>
<td>91.0</td>
</tr>
<tr>
<td>Port Wing</td>
<td>2.8</td>
<td>11.5</td>
<td>32.0</td>
<td>7.1</td>
<td>11.5</td>
<td>81.5</td>
</tr>
<tr>
<td>Vertical Tail</td>
<td>0.5</td>
<td>32.5</td>
<td>17.2</td>
<td>2.9</td>
<td>33.5</td>
<td>98.1</td>
</tr>
<tr>
<td>Horizontal Tail</td>
<td>1.2</td>
<td>31.5</td>
<td>38.9</td>
<td>4.0</td>
<td>31.5</td>
<td>126.7</td>
</tr>
<tr>
<td><strong>Sum Weight</strong></td>
<td><strong>28.4</strong></td>
<td><strong>10.8</strong></td>
<td><strong>306.3</strong></td>
<td><strong>53.1</strong></td>
<td><strong>10.8</strong></td>
<td><strong>573.2</strong></td>
</tr>
</tbody>
</table>

The required fuselage center of gravity for the Phase I Morphing aircraft to be stable was too far forward. It was impossible for us to balance the aircraft without adding a significant amount of weight due to the substantial weights of the vertical and horizontal tail. It was determined that the only way to fly a stable aircraft was to redesign the tail and add a fuselage extension to move the converters further forward on the aircraft. Table 11 shows the additions to the aircraft, and the desired fuselage center of gravity for the Phase II Morphing aircraft.

Table 11. Center of gravity calculation and comparison between the Baseline and Phase II Morphing aircrafts is shown. Green represents the desired stable CG, orange represents Phase I Morphing components, and purple represents the Phase II Morphing components.

<table>
<thead>
<tr>
<th>Component Description</th>
<th>Baseline Weight (oz)</th>
<th>Baseline CG (in from tip)</th>
<th>Baseline Moment (oz-in)</th>
<th>Morphing Weight (oz)</th>
<th>Morphing CG (in from tip)</th>
<th>Morphing Moment (oz-in)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>21.2</td>
<td>8.9</td>
<td>188.2</td>
<td>17.0</td>
<td>11.5</td>
<td>155.5</td>
</tr>
<tr>
<td>Starboard Wing</td>
<td>2.6</td>
<td>11.5</td>
<td>30.0</td>
<td>7.5</td>
<td>12.0</td>
<td>91.0</td>
</tr>
<tr>
<td>Port Wing</td>
<td>2.8</td>
<td>11.5</td>
<td>32.0</td>
<td>7.1</td>
<td>11.5</td>
<td>81.5</td>
</tr>
<tr>
<td>Vertical Tail</td>
<td>0.5</td>
<td>32.5</td>
<td>17.2</td>
<td>1.8</td>
<td>33.5</td>
<td>60.3</td>
</tr>
<tr>
<td>Horizontal Tail</td>
<td>1.2</td>
<td>31.5</td>
<td>38.9</td>
<td>2.9</td>
<td>31.5</td>
<td>91.4</td>
</tr>
<tr>
<td>Fuselage Extension</td>
<td></td>
<td></td>
<td></td>
<td>2.8</td>
<td>2.4</td>
<td>6.8</td>
</tr>
<tr>
<td>Battery &amp; 3 Converters</td>
<td></td>
<td></td>
<td></td>
<td>10.8</td>
<td>1.2</td>
<td>13.27</td>
</tr>
<tr>
<td><strong>Sum Weight</strong></td>
<td><strong>28.4</strong></td>
<td><strong>10.8</strong></td>
<td><strong>306.3</strong></td>
<td><strong>50.1</strong></td>
<td><strong>10.8</strong></td>
<td><strong>539.4</strong></td>
</tr>
</tbody>
</table>

The chart in Table 12 below shows the comparisons between the weight and center of gravity for the baseline, first iteration aircraft, and second iteration aircraft. The first iteration morphing aircraft was unstable and too heavy due to additional MFC weight in the tail. The second iteration eliminated extra tail weight and added a nose extension to balance the aircraft. Both morphing aircraft are still significantly heavier than the baseline aircraft.

Table 12. Final weight comparison chart.

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Weight (oz)</th>
<th>CG (in from tip)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Baseline</td>
<td>28.4</td>
<td>10.8</td>
</tr>
<tr>
<td>1st Iteration</td>
<td>53.1</td>
<td>12.7</td>
</tr>
<tr>
<td>2nd Iteration</td>
<td>50.1</td>
<td>10.8</td>
</tr>
</tbody>
</table>
XIII. Laser Scan Tests

A. Geometry Scans of Lifting Surfaces

In order to capture the geometry of the fabricated lifting surfaces, measurements were taken about each surface using a laser and a two-axis linear stage. The laser was mounted on an aluminum L-bracket as shown in Figure 67. The wing was covered with masking surface to guarantee a consistent reading from the laser (Figure 68). A breadboard based circuit was used as the converter for the actuation of the wings (Figure 69). The wing was mounted to the moving base of the linear stage using the wooden rig previously used for the bending-torsion tests.

These ‘scans’ were automated using a code developed in LabView. The scan was taken for all four surfaces at voltages of +1500V, 0V, -1500V. The measurements were time-intensive, running between 4-6 hours per scan depending on the surface. The data from these scans was processed enough to verify good results but has not been further refined as it did not influence the design nor the construction of the final morphing aircraft. The data from these scans is nonetheless attached on the accompanying DVD and may be used, in the future, to compare the actual airfoil section to the design section. This may be considered a measure of the effectiveness of the fabrication technique employed by the current design team. An image of the raw data is presented in Figure 70 to give the reader an idea of the resolution and resulting data from these scans. It is not intended to present results.

XIV. Wind Tunnel Test

A. Purpose

The test was conducted in the Virginia Tech Stability Wind Tunnel on April 15, 2010. The goal of this test was to determine roll rates for the baseline aircraft, the Edge 540, and the morphing wing aircraft. It is important for us
to have accurate roll rate information to determine the benefits morphing control surfaces can provide for an R/C aircraft.

B. Model
The test rig consists of a full Great Planes Edge 540 EP 41” fuselage and tail with interchangeable baseline and morphing wings. The model was mounted to a rigid tube running the length of the fuselage along its roll axis, and exiting the aft section of the fuselage as shown in Figure 71. The tube extended back to the tunnel sting mount, through bearings and fixed with collars.

C. Procedure
The rig was mounted in the wind tunnel, and roll rates were measured for each set of speeds and aileron deflections. Multiple tunnel speeds were chosen in the range of 20-50 mph, depending on visual stressing to the model. Tunnel speeds and Reynolds numbers are shown in the run schedule in Table 13 below.

<table>
<thead>
<tr>
<th>Run Schedule</th>
<th>V = 0 mph</th>
<th>V = 30 mph</th>
<th>V = 40 mph</th>
<th>V = 50 mph</th>
<th>V = 60 mph</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Re = 0</td>
<td>Re = 200,000</td>
<td>Re = 270,000</td>
<td>Re = 340,000</td>
<td>Re = 410,000</td>
</tr>
<tr>
<td>Run 1 - Baseline</td>
<td>11 Degrees</td>
<td>11 Degrees</td>
<td>11 Degrees</td>
<td>11 Degrees</td>
<td>11 Degrees</td>
</tr>
<tr>
<td></td>
<td>20 Degrees</td>
<td>20 Degrees</td>
<td>20 Degrees</td>
<td>20 Degrees</td>
<td>20 Degrees</td>
</tr>
<tr>
<td>Run 2 - Morphing</td>
<td>500 V</td>
<td>500 V</td>
<td>500 V</td>
<td>500 V</td>
<td>500 V</td>
</tr>
<tr>
<td></td>
<td>1000 V</td>
<td>1000 V</td>
<td>1000 V</td>
<td>1000 V</td>
<td>1000 V</td>
</tr>
<tr>
<td></td>
<td>1500 V</td>
<td>1500 V</td>
<td>1500 V</td>
<td>1500 V</td>
<td>1500 V</td>
</tr>
<tr>
<td></td>
<td>-500 V</td>
<td>-500 V</td>
<td>-500 V</td>
<td>-500 V</td>
<td>-500 V</td>
</tr>
<tr>
<td></td>
<td>-1000 V</td>
<td>-1000 V</td>
<td>-1000 V</td>
<td>-1000 V</td>
<td>-1000 V</td>
</tr>
<tr>
<td></td>
<td>-1500 V</td>
<td>-1500 V</td>
<td>-1500 V</td>
<td>-1500 V</td>
<td>-1500 V</td>
</tr>
</tbody>
</table>

Aileron deflections for the baseline aircraft correspond to the maximum deflection on the low and high rate settings. Aileron deflections for the morphing wing are achieved with piezoelectric materials (Macro-Fiber composites), so these were measured in Volts. Trim settings were used to set each of these voltage points. A procedure detailed below was used to properly load and unload the ailerons to eliminate effects of hysteresis (shown...
in Figure 72). A video camera and multiple stopwatches were used to determine the roll rates of the aircraft at each of these speeds and deflections.

Figure 72. Hysteresis loop with labeled points.

Positive Voltage Procedure
2. Load to 100% voltage/trim (pt. A).
3. Wait 20 seconds.
4. Measure roll rate at 100% voltage/trim (pt. A).
6. Wait 20 seconds.
8. Load to -100% voltage/trim to cancel out hysteresis (pt. D).
9. Repeat steps 2-8 for all positive voltage cases.

Negative Voltage Procedure
2. Load to -100% voltage/trim (pt. D).
3. Wait 20 seconds.
6. Wait 20 seconds.
7. Measure roll rate at pt. E.
8. Load to 100% voltage/trim to cancel out hysteresis (pt. A).
9. Repeat steps 2-8 for all negative voltage cases.

D. Wind Tunnel Testing Results

The baseline aircraft configuration uses servo actuated ailerons that operate in three settings: low rates, high rates and 3D rates, this corresponds to a maximum aileron deflection of ±11°, ±20° and ±30° respectively. The morphing aircraft configuration uses Macro-Fiber Composites (MFC’s) that were set to output a maximum of ±500V, ±1000V, ±1500V for low, high and 3D rates respectively. The present wind tunnel test was used to compare the steady state roll rate of the baseline configuration (Figure 73) low and high rates and the morphing configuration (Figure 74) across its operating range.
Results from Figure 75 show that the baseline configuration at high rates deflection (±20°) was able to achieve about a 40% higher roll rate than that of the morphing configuration at its maximum voltage (+1500 V). Although the morphing wings provide an aerodynamic benefit by virtue of their smooth and continuous surface, the MFCs were not able to deflect as much as the servos, and therefore had less control authority.

For both configurations the roll rate varied depending on the direction of roll (i.e. a positive or negative deflection or voltage). This directional bias for both cases can be attributed to differences in fabrication between wings and to the slight asymmetric mounting of the axle with respect to the thrust line. The wind tunnel rig rigidly restrains the aircraft and this causes vibration and bending of the mounting axle during the roll. During the test the aircraft nose ‘wobbles’ or oscillated in small circles, and occasionally a ‘sticky point’ in the roll was observed where the roll momentarily slowed down. These effects are negligible at small airspeeds however become evident at higher speeds and may explain, in part, the decrease in roll rate at 60 mph for some cases. It is worth noting that the direction bias was small for the baseline but significant for the morphing configuration. This may be due to the fact that the
morphing wings were fabricated in-house, but may also be a result from some hysteresis that was not fully negated as assumed during the test procedure.

Figure 75 Variation of roll rate with airspeed. Degrees refer to aileron deflection of baseline configuration. Voltages refer to wing MFC actuation of morphing configuration

XV. Flight Testing

A series of flight tests were conducted that culminated in a final full morphing configuration flight. Craig Sossi was the primary pilot for all of the flights, in general an engine cowl was not used in flight for cooling purposes. The following is a summary of the main objectives, flight notes and lessons learned from each flight.

Flight Test No. 1 – December 08, 2010

The first flight test took place on the baseball field located on the corner of Church St. and Clay St. in Blacksburg, Virginia. The flight objective was to check out the flying characteristics of the out-of-the-box the aircraft. It was found that the aircraft was highly aerobatic, and with minor trim adjustments was easy to fly. Total flight time was approximately 8 minutes.

Flight Test No. 2 – April 02, 2010

Upon fabrication of the morphing surfaces and accompanying electronics, the team had a good sense of the weight of the morphing aircraft configuration. The flight was used to simulate the morphing configuration flying weight by using lead as ballast. Lead shot was packaged and taped on each wing and tail independently, as well as in the place of the converters. This was done to simulate not only the total weight increase, but to approximate the change in inertia as well. Takeoff from the asphalt runway was relatively long. The aircraft was trimmed in flight and although it behaved sluggishly, it flew well. Several racetrack laps were completed. Low rates and 3/4 throttle was used throughout the flight. Final approach was difficult due to headwind. The aircraft landing path diverged onto the grass and this brought the aircraft to an immediate stop – resulting in a nose over that broke the engine firewall. This damage was minimal and was repaired on site with five minute epoxy. Larger, rubber wheels with treads were recommended to help with landing and takeoff. Total flight time was approximately 4 minutes 50 sec.

Flight Test No. 3 – April 02, 2010

The flight test objective was to investigate the acrobatic maneuverability of the ballasted aircraft. The configuration was not changed from Flight Test No.2 . It was found that the aircraft flew straight and level with hands-off controls. Successful maneuvers completed include inverted flight and rolls (although slow). A snap roll and loop were attempted however the motor did not provide enough power to complete these maneuvers. While flying downwind after a missed approach a loss of power was encountered, both motor and servo control was lost. A distant crash landing resulted in damage to the structure in front of the foremost bulkhead, to the Rimfire 0.10 electric motor and to the onboard LiPo battery. The electronic speed controller (ESC) has a voltage cutoff device that reduces power to the motor in order to save the expensive LiPo batteries. It is suspected that a long flight reduced the voltage below this cutoff and resulted in the power loss. Total flight time was 5 minutes 5 sec.
Flight Test No. 4 – April 11, 2010

Bench testing found that the ESC cutoff at approximately 9.7 Volts and around 4-5 minutes of run-time. A new, more reliable electric motor was installed (EFLITE 10). The objective of this flight test was to investigate the flying characteristics of the airframe with the new motor. After takeoff the aircraft was severely out of trim in pitch and roll. Once corrected the aircraft was well behaved. Several racetrack laps were completed. A slightly hard landing resulted in a crack in the side of the fuselage between the starboard gear and the fore bulkhead. This portion was repaired by making a clean cutout from this damaged region and replacing it by gluing (with thick CA) a similar piece from the backup fuselage. Total flight time was 1 minute 35 seconds.

Flight Test No.5 – April 11, 2010

This flight test was the first to include morphing surfaces. The aircraft configuration consisted of MFC actuated wings and a servo actuated tail. The objective of the flight was to demonstrate the ability of the morphing wing to sustain flight and the actuation to provide roll control. It was found that MFCs were sufficient to control aircraft in roll. However the response time from pilot transmitted signals was delayed and the tendency for the MFCs not to center proved problematic. The effect of flying the aircraft by line of sight, with the lag in the MFCs, resulted in overcompensation in roll controls. Due to the limited flight time (from batteries) and the lack of confidence in the aircraft, an emergency landing was attempted. The landing position was misjudged and resulted in a nose strike and wing tip strike into the ground. The damage included a broken wing spar, minor cracks in the fuselage around the landing gear and a broken propeller. The damage to the fuselage was repaired with epoxy and CA. A new wing spar and propeller were installed.

Flight Test No.6 – April 29, 2010

This was the first all-morphing surface flight. Several takeoffs were unsuccessful due to the lag in rudder control. The torque from the motor pulls the aircraft to one side of the runway and keeping the aircraft centered was a challenge. On the third takeoff attempt the lag in the rudder upon liftoff was corrected with some roll control, however once again the lag and hysteresis of the tail and wings turned this into an uncontrollable roll and eventual dive into the ground. The impact destroyed the fuselage, however the morphing control surfaces remained intact. Total flight time was approximately 10 seconds.

Flight Test No.7 – April 29, 2010

A spare fuselage was assembled on the field from the wreckage of Flight Test No.6. It was decided to fly without the morphing rudder to avoid the problems encountered at takeoff with the previous flight test. For the present flight test, the aircraft flew successfully however there was an oscillatory roll action due to the lag and hysteresis of the wings throughout the flight. In addition there were high winds and periodic drops in altitude resulting from the constant side to side rolls. The pilots attention was focused on keeping the aircraft level and there were no moments of steady flight that would allow the pilot to climb to a comfortable altitude. Eventually one of the altitude drops resulted in the aircraft crashing into a tree. The foremost part of the fuselage was significantly destroyed. The gears in the servo of the rudder were stripped, however the morphing elevator and wing remained intact. Total flight time was approximately 1 minute and 30 seconds.

Conclusions

The goal of the flight test program was to assess the ability of MFC morphing surfaces to control a remote controlled aircraft. The flight tests demonstrated that the aircraft was able to accommodate the additional weight of the morphing system and the morphing surfaces provide enough control authority for flight. However, flight tests also showed that the morphing aircraft was sluggish and resembled a ‘slow flyer’ rather than an aerobatic aircraft like the baseline. Although the morphing surfaces provided sufficient control, the dynamic stability of the aircraft became a major concern because of the hysteresis and lag effects of the MFCs. Nonetheless, a fully morphing flight did occur and several partial morphing flights were very promising. The flight tests demonstrated the potential of morphing wing flight and provided insight into the major challenges associated with this technology.

XVI. Recommendations

A. Fabrication Timeline

It is recommended that the future team take a more ‘hands-on’ approach. In our experience, design decision for the final morphing aircraft were mostly driven by practical concerns. The airfoil and stability and control analysis was used to verify the design is feasible however the analysis did not drive the geometry of the aircraft. Instead, things beyond our control such as MFC length or spar diameter decided the sections used. It is critical to become aware of these practical concerns as early as possible. Ideally a first iteration of the aircraft should be constructed and flown in the first semester.
B. Weights and Balance
It is recommended one individual be responsible for documenting the weights and balance of the aircraft throughout the design and fabrication process. It is particularly important to use actual scales as often as possible rather than rely on estimates. It is also recommended that weight reduction become a high-priority in any fabrication decision. The tradeoff between structures, aerodynamics and additional mass must be considered in the context of an R/C aircraft, normally with a high power loading. In some cases, the drag reduction by say, adding a fairing, or adding a silicon boundary transition may not be worth the additional mass.

C. Dynamic Stability and Control Analysis
One of the major lessons we learned from this project was that the lag exhibited by the MFCs powered by the onboard converters, has a significant adverse effect on the flying qualities of the aircraft, making it ‘hard to fly.’ Apart from ensuring a seasoned pilot is at the sticks on your first flight attempt it is recommended a thorough study of the dynamic stability and control of the aircraft be conducted. In particular it would be worthwhile to model the actuator lag and hysteresis of the control surfaces and to modify the electronics to provide more responsive actuation. It would be ideal to implement a control system that converges to the commanded signal or to simulate these dynamics in a R/C simulator to give the pilot extensive training on the ground.

Appendix. Electronics

Table A1 Capacitance values and the corresponding system response time.

<table>
<thead>
<tr>
<th>Capacitance (µF)</th>
<th>Low Pass Filter</th>
<th>Settling Time (seconds)</th>
<th>Range (V)</th>
</tr>
</thead>
<tbody>
<tr>
<td>220</td>
<td></td>
<td>2.00</td>
<td>1300</td>
</tr>
<tr>
<td>100</td>
<td></td>
<td>0.75</td>
<td>1200</td>
</tr>
<tr>
<td>47</td>
<td></td>
<td>0.50</td>
<td>1100</td>
</tr>
<tr>
<td>22</td>
<td></td>
<td>0.13</td>
<td>1000</td>
</tr>
<tr>
<td>10</td>
<td></td>
<td>0.10</td>
<td>500</td>
</tr>
</tbody>
</table>

Table A2 List of all components and their values located on the PCB.

<table>
<thead>
<tr>
<th>Component</th>
<th>Value</th>
<th>Quantity</th>
<th>Component</th>
<th>Value</th>
<th>Quantity</th>
</tr>
</thead>
<tbody>
<tr>
<td>R1</td>
<td>3.3 kΩ</td>
<td>4</td>
<td>R12</td>
<td>1 kΩ</td>
<td>4</td>
</tr>
<tr>
<td>R2</td>
<td>JUMPED</td>
<td>4</td>
<td>R13</td>
<td>33 MΩ</td>
<td>4</td>
</tr>
<tr>
<td>R3</td>
<td>OPEN</td>
<td>4</td>
<td>R14</td>
<td>33 MΩ</td>
<td>4</td>
</tr>
<tr>
<td>R4</td>
<td>100 Ω</td>
<td>4</td>
<td>R15</td>
<td>100 MΩ</td>
<td>4</td>
</tr>
<tr>
<td>R5</td>
<td>500 Ω</td>
<td>4</td>
<td>R16</td>
<td>5 MΩ</td>
<td>4</td>
</tr>
<tr>
<td>R6</td>
<td>500 Ω</td>
<td>4</td>
<td>R17</td>
<td>5 MΩ</td>
<td>4</td>
</tr>
<tr>
<td>R7</td>
<td>100 MΩ</td>
<td>4</td>
<td>C1</td>
<td>150 µF</td>
<td>4</td>
</tr>
<tr>
<td>R8</td>
<td>56 kΩ</td>
<td>4</td>
<td>C2</td>
<td>47 µF</td>
<td>4</td>
</tr>
<tr>
<td>R9</td>
<td>56 kΩ</td>
<td>4</td>
<td>C3</td>
<td>220 µF</td>
<td>4</td>
</tr>
<tr>
<td>R10</td>
<td>50 kΩ</td>
<td>4</td>
<td>C4</td>
<td>OPEN</td>
<td>4</td>
</tr>
<tr>
<td>R11</td>
<td>10 kΩ</td>
<td>4</td>
<td>C5</td>
<td>150 µF</td>
<td>4</td>
</tr>
</tbody>
</table>
Figure A1 PCB layout. Red lines indicate top wiring. Green lines indicate bottom wiring. Yellow is silkscreen.

a.) AM 1505
b.) AM 2003

c.) AM 2505

Figure A2 DC/DC Converter voltage input/output comparison.
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10Common Collective Filter. Georgia State University. http://www.hyperphysics.phy-astr.gsu.edu


20“Hysteresis,” 27 April 2010 <http://image.tutorvista.com/content/magnetism-matter/hysteresis-loop.jpeg>