Abstract

The basis of this multi-disciplinary project is to reverse engineer, integrate, automate and flight-test an unmanned miniature Flying Wing Air vehicle. This project was done in close collaboration with industrial aircraft manufacturers, Cradence Services, principally centered about their latest miniature drone, the Golden Eagle, together with my colleague Mr. Low Jun Horng whose efforts were mainly in integration, flight controls and automating the craft for GPS waypoint flight.

This study establishes a reverse engineering routine primarily for the aerodynamic data generation for an unconventional miniature reflexed Flying Wing airfoil, for which there was insufficient contractors’ aerodynamic data and stability derivatives provided. The thesis then goes on to describe in detail also a material research and selection procedures and the reverse-prototyping of the test platforms for which there was also insufficient contactsors specification. With these accomplished, we then focused on further analysis and modification to the original power plant to enable the platform to carry an additional payload of 250g, which encompasses an autonomous navigation system, and a real time operating camera. Some of the techniques adopted were 3D Laser profile scanning, Computational Fluid Dynamics studies, weight and balance matching, CG and Inertia tensor estimation and a series of coordinated glide and flight tests. Various tests were done through the course of the project to validate and proof the integrity of theoretical results derived. Results of the calculations were found to be consistent and useful in characterizing the unknown airfoil.

A paper based on this project was presented at the Republic of Singapore Air Force’s (RSAF) Aerospace Technology Seminar on February 2005.
Acknowledgements

The author would like to extend sincere gratitude to his project supervisor, Associate Professor Gerard Leng Siew Bing for his guidance, and above all patience in answering all queries pertaining to the project. Also, the authors would like to thank Mr Leong See Kit of Cradence Pte Ltd for allowing us to use his equipment during the course of this project.

Special thanks, also to Mr Low Jun Horng, Student, FYP AM23, for his hard work and dedication in getting the prototypes working and for integration of the control system.

The author will also like to extend his gratitude to the staff of Dynamics Laboratory, Encik Ahmad Bin Kasa, Ms Amy Chee, Ms Priscilla Lee, and Mr Cheng Kok Seng, for their assistance for the duration of the project, and as well as Mr Neo Ken Soon of the Advanced Manufacturing Laboratory for his assistance in the use of the 3D Laser.
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<td>$A_{\text{rotor}}$</td>
<td>Area of propeller disk / m²</td>
</tr>
<tr>
<td>$C_d$</td>
<td>Coefficient of Drag for airfoil</td>
</tr>
<tr>
<td>$C_L$</td>
<td>Airfoil lift coefficient</td>
</tr>
<tr>
<td>$C_{L,\text{max}}$</td>
<td>Maximum lift coefficient</td>
</tr>
<tr>
<td>$c$</td>
<td>Airfoil Chord / m</td>
</tr>
<tr>
<td>$C_P$</td>
<td>Coefficient of Power</td>
</tr>
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<td>$c_p$</td>
<td>Power coefficient for propeller</td>
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<td>$c_{p,\text{total}}$</td>
<td>Total Power coefficient of propeller</td>
</tr>
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<td>$C_T$</td>
<td>Coefficient of Thrust</td>
</tr>
<tr>
<td>$c_{T,\text{prop}}$</td>
<td>Thrust coefficient for propeller</td>
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<td>$c_{T,\text{total}}$</td>
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<td>$C_m$</td>
<td>Coefficient of Moments</td>
</tr>
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<td>$C_{m,c/4}$</td>
<td>Airfoil pitching moment about the quarter-chord point</td>
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<td>$d$</td>
<td>Diameter / m</td>
</tr>
<tr>
<td>$D$</td>
<td>Diameter of Propeller / m</td>
</tr>
<tr>
<td>$I$</td>
<td>Current / A</td>
</tr>
<tr>
<td>$I_{xx}$</td>
<td>Moment of Inertia about x axis</td>
</tr>
<tr>
<td>$I_{yy}$</td>
<td>Moment of Inertia about y axis</td>
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<tr>
<td>$I_{zz}$</td>
<td>Moment of Inertia about z axis</td>
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<tr>
<td>$I_{yz}$</td>
<td>Product of Inertia about y and z axis</td>
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<td>$I_{zx}$</td>
<td>Product of Inertia about z and x axis</td>
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I_{xy} \quad \text{Product of Inertia about the } x \text{ and } y \text{ axis}

L \quad \text{Length} / \text{m}

M \quad \text{Mass} / \text{kg}

M_{bi} \quad \text{Reading of digital balance due to mass} / \text{kg}

p \quad \text{Total pressure} / \text{Pa}

P \quad \text{Power} / \text{kW}

P_{\text{induced}} \quad \text{Induced power} / \text{kW}

P_{\text{total}} \quad \text{Total power due to Propeller and motor}

r_{\text{prop}} \quad \text{Radius of Propeller} / \text{m}

Re \quad \text{Reynold’s number}

T \quad \text{Thrust} / \text{N}

t_{\text{duct}} \quad \text{Thickness of duct} / \text{m}

V \quad \text{Voltage} / \text{V}

V \quad \text{Velocity at propeller disk} / \text{m/s}

W \quad \text{Weight of Prototype} / \text{kg}

\alpha \quad \text{Glide Angle}

\rho \quad \text{Density of air}, = 1.21 \text{ kg/m}^3 \text{ unless stated otherwise}

\Omega \quad \text{Angular Velocity of the Propeller} / \text{rad/s}
Aerodynamic Analysis of a Flying Wing UAV

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In partial fulfillment of the requirements for the Degree of Bachelor of Engineering
National University of Singapore

Session 2004 / 2005
Chapter 1 Introduction

1.1 Thesis Background

Unmanned Aerial Vehicles (UAVs) are remotely piloted or self-piloted aircraft that can carry cameras, sensors, communications equipment or other payloads. They have been used in a reconnaissance and intelligence-gathering roles since the 1950s, and more challenging roles are envisioned, including swarmed combat missions. The autonomous fixed- Flying Wing mini-UAV is the emerging class of vehicles in this family of UAVs and have significant importance to many fields, with applications including short range military reconnaissance and rural search-and-rescue.

Flying Wing MAVs are not just smaller versions of larger aircraft. They are fully functional, militarily capable, miniature flight vehicles in a class of their own where conventional aerodynamic theories do not always hold. The Reynolds number (a measure of size multiplied by speed) is the most useful single parameter for characterizing the flight environment. The low Reynolds number regime, where most MAVs fly within is an environment more common to the birds and where our basic understanding of the aerodynamics encountered is very limited up till today, making the task of mechanizing flight under these conditions a challenging one.

Unsteady flow effects arising from atmospheric gusting or even vehicle maneuvering are far more pronounced on small scale MAVs where inertia is almost nonexistent, that is, the wing loading is very light. Also given the limited wingspan available, MAVs have to achieve higher relative wing areas by having larger chords, i.e. by using configurations with low aspect ratio, like flying wings. Hence in general MAVs have to cope with fully three-dimensional aerodynamics where we have even less
low-Reynolds number data available, hence necessitating a complete and comprehensive aerodynamic analysis of the Golden Eagle Aircraft in this thesis.

Because small and lightweight mini-Flying Wings are quite difficult to fly manually, and a successful high-level interface is needed to increase the number of potential users and applications of these aircraft in the Industry. An on-board autopilot offers several levels of autonomy with the highest being the ability to automatically take off, follow a mutable set of GPS waypoints, and land automatically. Hence, the requirement by our industrial collaborators Cradence Services to automate the hyper-control sensitive Golden Eagle Flying wing MAV.

Combining the advantages of autonomous flight and high speed level flight, in a simple package, this concept is used by modern UAVs such as the Elbit Seagull, Sky Lark and the Rafeal, which is used extensively by the Israeli military for most of their aerial reconnaissance missions.

Figure 1. Modern Flying Wing UAVs in service today.
1.2 Objectives

This study is centered primarily about three main objectives with the first one being the complete reverse engineering of the aircraft structure, airfoil and aerodynamic data for a commercial UAV, the Golden Eagle. With this accomplished the focus is then on further aerodynamic analysis and scientific modification to the original design and power plant to increase the payload margin of the platform to carry an autonomous navigation system and a real time operating camera. Finally, the comprehensive integration of the structure, onboard controls and power plant.

The usual method of developing an aircraft is to decide what the mission requirements are, finding an aerofoil shape specific to it by testing, do a sizing and performance optimization and integrate it together with the other parts of the aircraft, i.e. controls, propulsion systems, payloads etc. As the original Unmanned Air Vehicle (UAV) platform was given without adequate aerodynamic, propulsion and stability data, the development chain was broken. This required a fair amount of reverse engineering, to determine the aerodynamic coefficients and forces, which were then used to obtain the stability derivatives by Mr. Low in designing the autonomous control system. As there was no previous literature or established system of reverse engineering for unconventional airfoil in hand, we had to structure our own simulation environments and empirical verification routines through out the course of this project.

The original craft was too costly to be tested on and hence identical test rig prototypes were to be constructed. Given this task, we were again faced with limited design specifications and materials used by the contractors, thus requiring detailed studies in structural and material analysis prior to the reverse-prototyping of the test beds.
1.3 The Golden Eagle Micro Air Vehicle (MAV)

The UAV we are working with is basically a flying wing but with a central fuselage that follows the reflex airfoil shape longitudinally and adapts to the curved ‘M’ shaped, tip to tip wing layout when viewed from the back.

![Figure 2. Shape of the UAV (rear view)](image)

The entire aircraft (modular wings and fuselage) is constructed using ultra-light weight composite Kevlar fibre. Its fuselage is specifically designed to house 4 Lithium batteries, a speed controller and a rear pusher propeller unit. The craft is estimated to be able to carry a payload of 1.2 kgs and fly at speeds up to 20 m/s. Effectively, there are only two control surfaces on the UAV. These are the left and right elevons found at the ends of the wings of the aircraft. These control the pitching and rolling on this UAV. The wing tips are angled upwards at about 30 degrees to the horizontal to compensate for the lack of the rudder surfaces, acting as a pair of winglets to provide lateral stability to the aircraft. Neither exactly a sweptback wing or a Delta wing, its unconventional airfoil structure was carefully analyzed and pre-existent aerodynamic theories have been adapted to suit it where possible.

- Wing Span- 650mm
- Overall Length- 770mm
- Weight- 1500g
- Flight Endurance- 2hrs
- Speed- 10-20m/s
- Altitude (up to)- 500m

![Figure 3. The Golden Eagle](image)
All Dimensions in millimeters.

Figure 4. Golden Eagle Dimensional Drawing
1.4 Goals to Achieve

The following were set to be achieved

1) Literature study of Flying Wing MAVs
2) Analysis of Reflexed Airfoils
3) CAD modeling of Golden Eagle’s Airfoil
4) Dimensional Slicing to get Design Specifications of unconventional airfoil
5) Construction of Test Rig and Prototypes.
6) Reverse Engineering of Aerodynamic Coefficients
7) Propulsion Studies and Integration
8) Control systems Integration
9) Fully equipped Test Flight with Autopilot Navigation
1.5 Structure of the Dissertation

This thesis is divided into 11 Chapters. Chapter 1 - introduces and defines the objectives of the project. Chapter 2 – Discusses the background on Flying Wings specific to its increased application today. Chapter 3 – Discusses aerodynamic theory as applied to Flying Wing aircrafts. Chapter 4 – Discusses the challenges we faced with the lack of design specifications and describes the reverse engineering process. Chapter 5 – Describes the Computational Fluid Dynamic computations that we did and presents our experimental data. Chapter 6 – Discusses the Longitudinal stability analysis, by CG positioning and mass of inertia estimation. Chapter 7 - Describes the preliminary glide tests and verification of our theoretical data. Chapter 8 – Covers our efforts in integrating the propulsion system. Chapter 9 – Describes the final phase in the integration of the autopilot control systems and the Flight Tests. Chapter 10 – Appropriately summarizes and concludes the lessons learnt and Chapter 11 – Suggests topics and areas for further research and work.

The stages set out for this project are: i) Developing a CAD and physical model of the given UAV, iii) Computational Fluid Dynamic (CFD) and semi-empirical estimation of aerodynamic coefficients, iv) Glide testing v) Propulsion Integration, and vii) Flight tests.
Chapter 2: Literature Survey

2.1 History and Evolution of Flying Wings

Early studies of delta wings led aircraft designers to ask if an entire airplane could consist only of a wing, with basically no fuselage whatsoever. Such all-wing aircraft would have excellent payload and range capabilities because they would produce less drag than a conventional aircraft. This was true because the tail and fuselage normally cause a significant amount of drag. Eliminate the tail and fuselage and you have eliminated a great deal of drag, enhanced performance, reduced the amount of fuel required, and generally improved the handling capabilities of the airplane. These so-called flying wing designs were long a dream of a number of designers but did not become practical until recently. The biggest problem found when building a flying wing aircraft is that such designs are inherently unstable and they do not easily stay level in flight.

The first jet powered all-wing aircraft flew in Germany on February 2, 1945, and at the time was also virtually undetectable by radar. In the United States, John Knudsen Northrop launched his first aircraft the “Flying Wing”. Over in the Soviet Union; the most successful Soviet designer was Boris Ivanovich Chernanovski, who developed a series of flying wing projects from 1921 to 1940. Although development of the all-wing aircraft began at about the same time in Germany, the Soviet Union and America, there was no collaboration whatsoever between designers. In spite of this, design teams in these widely-separated parts of the world were convinced that the all-wing aircraft was the best configuration and pursued the idea with much idealism. The all-wing concept had achieved its first practical success.
This fact has not been lost by the aeronautical engineers of today, who design Flying wings for use in various purposes ranging from hobbyist flights such as the RB-35 to stealth missions such as the famed B-2.

2.2 Concept and Theory of the Flying Wing

Every airfoil has three forces. Lift, weight (both vertical) and drag (horizontal). If lift and weight are placed on the same spot, the airfoil is stable. But most airfoils are not stable. The lift force is mostly located after the weight force. So it generates a turning moment - Nose down, pitching moment.
As we can see, there must be counter stabilizing force in the opposite direction to the downward pitching moment (negative pitching moments, $C_m$) of the nose to allow the aircraft to fly in a stable condition. This can be achieved by a downward force on the tail horizontal elevators (Tail Lift), as in conventional tailed aircrafts or by an upward vertical force on the horizontal surface to the front of the plane, as in canards.

Fig 6: Natural Nose down Moments of Conventional Airfoils

The problem now in designing a Flying wing is to achieve this very longitudinal pitch stability with the absence of the entire tail section (Rudder, elevators and tail-tips).

Fig 7: Instability of Conventional Airfoils

There are 4 basic ways by which this can be achieved in an All-Wing aircraft,
1. Give the wing an arrow form (sweep) and twist the wing. Usually swept backwards with a downward twist.

2. Use an auto stable airfoil (lift- and weight forces on the same point). Here we don’t have to use sweep, but instead the reflexed airfoil.

3. Place angled surfaces on the tips of the wing – winglets to provide lateral and horizontal stabilizing effects.

4. Place the center of gravity very low.

These 4 methods will be discussed in brief in the next sections highlighting their respective benefits and disadvantages.

2.2.1 Solution 1: Sweep and Twist

The narrowed wing tips provide the compensating down- (in case of backward sweep) force or up- (in case of forward sweep) force to the turning moment of the airfoil in the center. This neutralizes the inherent nose down moments of flying wings enabling stable flight. Sweep in the flying wing is analogous to a tail in that it allows for trimming the aircraft.

Fig 8: Sweep Angle on a backward swept wing

The angle of sweep is measured from the lateral axis to the line, which is placed on 1/4 of the wing chord length, also known as the quarter chord reference line.

Fig 9: Twist Angle on a wing section
The twist-angle is the angle between the airfoil at the root of the wing and the airfoil at the tip of the wing. When using a twisted wing, the airfoils do not have the same angle according to the longitude axis. This leads to good situations if we use a backward sweep. If the center section of the wing stalls, the tip airfoils are not near the angle to stall. If we place elevons on these tips, you can still control the aircraft and you can avoid getting the plane into a spin.

### 2.2.2 Solution 2: Reflexed Airfoils

These designs use an airfoil, which doesn’t require a sweep. Therefore they are the most compact version of a flying wing, also called auto-stabilizers or S-shaped wings.

Fig 10: Sketch of a Reflexed Airfoil

This airfoil (CJ-5) is an example of an auto stable or reflexed airfoil. Note that the trailing edge goes up. You can see a reflexed airfoil as a normal airfoil with a tail-airfoil in one.

**Advantages:**
- Auto stable means no stall and no spin provided the CG is placed correctly.

**Disadvantages:**
- Reflexed airfoils have less lift than normal airfoils. So more wing area is needed to have the same lift.
2.2.3 Solution 3: Tailed-Tip Wings

These wings have a large angle of sweep. The classic horizontal or angled tail surfaces are placed on the tips of the wing - also known as winglets. This way, we have the necessary down force to compensate the turning moment of the wing (the force-arm (distance between center of gravity and elevators) is long enough) and you need to have a long fuselage to hold the tail. Most known designs have the vertical tail also placed on the tip. Here you can also combine the elevators with the roll-rudders (elevons). Fig 11: Highly swept wing

Advantages:

- A large moment arm with respect to the CG makes these surfaces ideal lateral-directional controls. A great deal of control power can be generated by a relatively small surface by staggering the surface aft.

Disadvantages:

- Complex structures to be built due to winglets and an increased flight weight due to the added servos and servo mechanisms on both wing tips.

2.2.4 Solution 4: Low Centre of Gravity

The moment created by the wing gets (fully or partially) compensated by the very low CG. This technique is often used with ultra light. Mostly hang gliders (using weight shift as flight control) use this technique to its full use.
Advantage:

1. Very easy in design. No consideration for twists and sweep.
2. Still can use airfoil with some pitching moment $C_m$ like a "normal" airplane.

Disadvantage:

1. A cockpit hanging under the wing makes more drag than an integrated cockpit in the wing.

Upon careful analysis of the advantages and disadvantages we chose to use a reflexed unswept airfoil—Solution 2—in our prototype airfoil; One contributing factor is that the similar type of airfoil is currently adopted in the Golden Eagle aircraft and hence would be the easiest to implement, requiring minimal changes to the contractor’s choice of wing and aircraft design. Furthermore, we also concluded that the reflexed wing generates the highest amount of tail down counter moments within the given design limitations; hence also the most effective in stabilizing the flying wing body.
Chapter 3: Theoretical Analysis of Airfoil Aerodynamics

3.1 Aerodynamic Forces and Coefficients

Lift is the force acting at 90 degrees to the relative airflow as a result of the air flowing over an aerofoil, whilst drag is the air resistance opposing the direction of airflow. Lift and Drag forces depend on size, shape, attitude, fluid properties, and velocity. In addition to the shape and attitude of the body, the surface roughness also has an effect on these forces.
Theoretical Analysis of Airfoil Aerodynamics

\[ \sum F_x = 0: \quad L \cos \left( \frac{\pi}{2} - \alpha \right) - D \cos \alpha + Mg \sin (\gamma - \alpha) = 0 \]

\[ \sum F_y = 0: \quad L \sin \left( \frac{\pi}{2} - \alpha \right) - D \sin \alpha - Mg \cos (\gamma - \alpha) = 0 \]

\[ \sum m = 0: \quad m = 0 \]

For small angles, these equations are

\[ L\alpha - D + Mg(\gamma - \alpha) = 0 \]
\[ L - D\alpha - Mg = 0 \]
\[ m = 0 \]

Let \( S \) be the wing area, \( c \) the wing chord, \( q_\infty = \frac{1}{2} \rho V_\infty^3 \) the dynamic pressure, and divide by \( q_\infty c \) to find

\[ C_L \alpha - C_D + \frac{W}{q_\infty} (\gamma - \alpha) = 0 \]
\[ C_L = C_D \frac{W}{q_\infty} \]
\[ C_m = 0 \]

where \( w = Mg / S \) is the wing loading. The lift, drag and moment coefficients are

\[ C_L = \frac{L}{q_\infty c}, \quad C_D = \frac{D}{q_\infty c S}, \quad C_m = \frac{m}{q_\infty c S} \]

This comprehensive factor in the Lift equation is termed the coefficient of lift, \( C_L \) represented by the equation,

\[ \text{Lift} = \rho_\infty \times V_\infty^2 \times S \times \text{Factor} \left( \alpha, \frac{\rho_\infty V_\infty^3}{\mu}, \frac{V_\infty}{a_\infty}, \text{surface roughness, air turbulence} \right) \]

\[ C_L = \frac{1}{2} \rho_\infty V_\infty^2 \times S \]

Likewise for the force of drag,

\[ \text{Drag} = C_D \times \frac{1}{2} \rho_\infty V_\infty^2 \times S \]

\[ C_D = \frac{1}{2} \rho_\infty V_\infty^2 \times S \]

National University of Singapore
Department of Mechanical Engineering
3.2 Aerodynamic Moments and Pitching Moment Coefficients

Moments occur when the c.g is not placed directly above the a.c, hence generating either a nose down or tail down turning moments shown by the equation,

$$\text{Moment} = C_m \times \frac{1}{2} \rho \infty V^2 \times S \times \ell$$

$$C_m \equiv \frac{\text{Moment}}{\frac{1}{2} \rho \infty V^2 S \ell}$$

In the case of a tailed aircraft not much attention is paid to the airfoil pitching moment coefficient, $C_m$. A specific airfoil is selected usually because of performance criteria and stall characteristics and the negative (nose down) pitching moment is tolerated as a necessary evil. Horizontal stabilizers with large moment arms can be easily used to neutralize the moments. But in our case, the $C_m$ value has to be minimal and made positive, meaning no residual negative pitching moments, giving us a neutral and inherently stable airfoil shape throughout flight.

### 3.2.1 Effect of camber on $C_m$

Thin-Airfoil Theory determines that the pitching moment generated is dependent almost entirely on the camber and the distribution of camber of the airfoil. It requires
Theoretical Analysis of Airfoil Aerodynamics

A detailed calculation for each specific shape of camber line. Here, we simply note that, for a given shape of camber line the pitching moment about the aerodynamic center is proportional to the amplitude of the camber, and generally is negative for conventional camber shapes and the more camber the airfoil has the more negative its pitching moment will be. A symmetric airfoil has zero $C_m$ while a negative cambered will create a positive $C_m$, nose-up moment. The distribution of camber also significantly affects the $C_m$ and in general, the more forward the position of the camber, the larger its effect will be on creating a nose up moment by creating a large positive effect on the maximum lift of the airfoil and hence a positive $C_m$.

![Camber Profile Definition](image)

Fig 17: Camber Profile Definition

(A detailed Camber table with the airfoil Profile and their respective $C_m$ values is given in Appendix A)

### 3.3 Self Stabilizing Reflexed Wings

![Camber Profile Definition](image)

Figure.18. Golden Eagle’s matching reflex wing profile

The moments and forces for trimmed airfoil are denoted with an asterisk (*). The forces are the weight of the model $mg$, and the aerodynamic lift $L$, which have to cancel out (sum of forces in vertical direction equals zero). The drag forces are
neglected here. The sum of the moments around c.g. (caused by the airfoil moment $M$ and the lift force $L$, acting at a distance from c.g.) must also be zero.

<table>
<thead>
<tr>
<th>conventional airfoil with camber</th>
<th>airfoil with reflexed mean line</th>
</tr>
</thead>
<tbody>
<tr>
<td><img src="image1.png" alt="Diagram" /></td>
<td><img src="image2.png" alt="Diagram" /></td>
</tr>
<tr>
<td>$c_m &lt; 0$ (nose heavy)</td>
<td>$c_m &gt; 0$ (tail heavy)</td>
</tr>
</tbody>
</table>

This airfoil has a nose heavy moment. The center of gravity is also the center of rotation of the wing. When it is located behind the aerodynamic centre, $ac$ point, the air force $L^*$ in front of the c.g. counteracts the nose heavy moment $M^*$ to achieve equilibrium.

The reflexed camber line makes the moment coefficient positive, which means, that the moment around the $ac$ point is working in the tail heavy direction. Therefore the center of gravity has to be located in front of the $ac$ point to balance the moment $M^*$ by the lift force $L^*$. 
Theoretical Analysis of Airfoil Aerodynamics

Disturbed State

When the angle of attack is increased (e.g. by a gust), the lift force $L$ increases. Now $L > L^*$ and the tail heavy moment due to the lift is larger than the moment around $ac$, which still is $M = M^*$. Thus the wing will pitch up, increasing the angle of attack further. This behavior is instable and a tailplane is needed to stabilize the system.

Here, we have the air force acting behind the c.g., which results in an additional nose heavy moment, when the lift increases. With $L > L^*$, the wing will pitch down, reducing the angle of attack, until the equilibrium state is reached again. The system is stable.
3.4 Coupled Control Surfaces

Meanwhile, another important consideration for flying wings is the evolution of Elevons, which are the coupled and only control surfaces in the absence of Elevators and the Rudder for a flying wing platform.

Elevons control the flying wing’s movement in the pitch, yaw and roll axis during flight. They conveniently replace the bulky tail section and require only two servos to operate, thus reducing the overall flight weight of the flying wing aircraft.

ELEVATORS + AILERONS = ELEVONS
4.1 Rationale behind the need for CFD Analysis

To understand the characteristics of an airfoil, we need to know the precise aerodynamic forces of lift, drag and the aerodynamic pitching moments. The Golden Eagle airfoil we have in hand is currently with no such data, hence, impossible to study and analyze. Also with the aerodynamic coefficients, the Equations of motion can be derived by substituting them into these equations. From these equations, the reactions of the aircraft to specific inputs are known hence enabling us to design a specialized control system for the Golden Eagle. Conventionally, wind tunnel testing is done on the model with strain and pressure gauges and velocity indicators attached all around the airfoil. This allows the aerodynamic forces and moments to be experimentally measured and subsequently the coefficients computed. But we adopted another simpler but still equally reliable and less expensive method, given the scope of our Final year Project– Computational Fluid Dynamic Simulation.

Prior to this, an accurate CAD model needed to be generated. One which we could mesh and use for our various CFD simulation runs, simulating different wind speeds and angles of attack.

But with almost no specifications on the camber or spanwise curvature of this highly unconventional airfoil, creating a CAD model could not just be done from external physical measurement of the wing. Hence, a reverse engineering routine was established to profile this airfoil.
4.2 Airfoil Profiling

4.2.1 Laser Profile Photography

Using the Minolta, VIVID 900, Non-Contact-3D Digitizer Image Laser scanner, we photographed the entire wing profile and fuselage with a tolerance of ±1.5 mm. Each Wing section had to be photographed from at least six angles all around so that we could register the appropriate merging points for assembly later. The glossy surface of the wings had to be matted down with a fine layer of powder dusted through out. The Laser beams are absorbed by black surfaces; hence to get a proper edge definition, we photographed the sections against a black back drop. As the edges of the original Golden Eagle airfoils were painted with a black strip, we inevitably lost some edge details. But this was overcome in the assembly process.

Figure 20. 3-D Laser scanning and Reverse aerofoil CAD modeling procedure
The photographed mesh shells were then merged using the commercial scan programme RapidForm™ 2002- Reverse Modeler Version. Working with the photographed scattered points, we had to systematically connect each coordinate to attain the complex curves on the wing. Plot linearization and CAD editing was needed to marginalize the inaccuracy inherent in scanning. As we were primarily in search of the curvature coordinates, we were not too concerned about the inaccuracy along the edges of the wing which could be manually obtained by paper tracing and plotting.

4.2.2 CAD Modeling

The commercially available CAD software, Solidworks™ was used to edit the points to form a completed 3D model. The side profile of the fuselage end of the wing was traced and plotted out on paper. Coordinates at intervals of 5mm were assigned, measured and input into Solidworks. Nextly, with the sectional camber coordinates from the laser digital photographs we were able to create a guide curve by which we could loft the root end of the airfoil to the tip end. This procedure is repeated for the winglet, for which we also know the angle of inclination from the laser images, with the exception that instead of a guide curve in this case, a straight line was used to loft the winglet up to the tip.

The fuselage was more straightforward to model in that it did not consist of any complex geometry. The fuselage was assumed to be a box with a rectangular-base and a top that followed the curvature of the wing roots. The nose was also modeled using simple interpolation curves based on physical measurements.

The control surfaces (elevons) were modeled separately and assumed to be flat rectangular pieces that could be rotated in the CAD model to simulate deflections.
4.3 Performing the Preliminary Analysis

4.3.1 Preparation of CFD Mesh

The final CAD model was converted into a STEP file format and meshed using GAMBIT a mesh preprocessing program. A mesh analysis was done using meshes of different sizes to determine which mesh would give the most accurate result without a compromise on computational capacity and time. The model is meshed with an unstructured distributed triangular mesh, with courser mesh elements near the wall boundaries of the control volume and denser meshing at the leading, top and bottom surfaces of the airfoil. Usually, a finer mesh would give the most reliable results for a CFD calculation, but it is more computational expensive and thus not efficient. Therefore, we used an axis-symmetrical model and conducted several runs re-adapting the mesh, till the solution converged and displayed mesh independence.
One of the problems encountered was the skewed edges that the CAD model when imported to Gambit. Skewed edges are formed when large and long surface converge to line creating a sharp extended edge like the ones found on the trailing edges of our airfoil. The preprocessor has an extremely stringent condition for meshing and it did not recognize parts assembled in external CAD programs. Hence we actually had to do considerable mesh “tidying” in Gambit by exercising node and mesh control on these problematic edges.

4.3.2 Simulation Model

Prior to the calculations, a calculation on the UAV’s Reynold’s Number (Re) was done to investigate whether the flow over the airfoil would be considered laminar or turbulent. As \( Re = \frac{\rho ud}{\mu} \), we would need to estimate the density (\( \rho \)) and the viscosity (\( \mu \)) of the air flowing over the airfoil. The velocity of air over the airfoil (\( u \)) is assumed to be constant at 10m/s, which is the cruise airspeed specification given to us by the manufacturer, and the characteristic length is taken to be 0.77m, the overall length of the given UAV. Air at atmospheric pressure and the density is taken to be 1.1774 kg/m\(^3\) and the viscosity was found to be 1.8462 x 10\(^{-5}\) kg/ms at 300K.

Therefore, the Reynold’s number is:

\[
Re = \frac{\rho ud}{\mu} = \frac{1.1774 \times 10 \times 0.77}{1.8462 \times 10^{-5}}
\]

\[
= 4.91 \times 10^5
\]

The actual value of a critical Reynolds number that separates laminar and turbulent flow can vary widely depending on the nature of the surfaces bounding the flow and
the magnitude of perturbations in the flow. Since there is very limited knowledge
about the aerodynamics of mini-flying wing UAVs today, we had no tangible
platform to compare our $Re$ number. Given, the subsonic speed of 0.33 Mach we
neglected compressibility effects and used the simple Laminar flow assumption for
the initial simulations. (Results displayed in Appendix A)

Evidently, from the first few runs, we noted that the drag force was unrealistically low
and later via verification through glide tests discovered that the lift force and hence
the $C_L/C_D$ ratio was also inaccurate. Sighting the turbulence conditions that we test
and fly our aircraft in, and due to fabrication constraints, the airfoil constructed had
considerable surface irregularities, especially on the bottom side. From readings on
low speed aerodynamics, this factor should be sufficient to trigger an early transition
from laminar to turbulent flow. Hence, a suitable computational turbulence model
must be selected and used, because of the high possibility that turbulent, rather than
laminar, flow separation would occur the air foil. Also the practical environment
where the Golden Eagle operates in real life missions encounters much wind hence
reassuring us that the influence of turbulence must not and cannot be ignored.

From the Notes on Numerical Fluid Mechanics, modeling of flow separation for
aerofoils at low Reynolds number, we realized that Turbulence modeling in any detail
is an extensive subject and hence could not be covered in detail for this project. The
key to successful modeling of turbulent boundary layer flow is the selection of a
correct turbulent model. In any case, the turbulent models possible are limited by
those available within Fluent 6, which were state-of-the-art.
A K-Epsilon (RNG) model was used as the K-Epsilon method [19] has been generally used for turbulence modeling problems. Although another method which requires a lower computational time is available (Spalart-Allmaras model), it is a one-equation model, and thus the results would not be as accurate. The RNG K-Epsilon model [20] was used as it incorporates a formula for lower Reynolds number effects. Along with other features, it serves as a better model than the normal K-Epsilon model particularly in our project where we got consistent results that were later verified experimentally to be rather precise.

4.3.3 Relevant Parameters

Numerical Scheme: 2\textsuperscript{nd} Order Upward Scheme
Viscous Model: Turbulent Setting
Element Type: Triangular
Grid Size: Cells: 75918
Element Size: 2-10 mm
Fluid Type: Air with $\rho = 1.1774 \text{ kg/m}^3$ and $\mu = 1.8462 \times 10^{-5}$
BCDs :
– Inlet: Constant velocity
– Side: Axis-Symmetry
– Upper and lower: Periodic setting

Figure 24. 3D-Control Volume
4.4 Results and Discussion

The CFD runs were done with a wind angles ranging from 0 – 50 degrees and velocities ranging from 10 to 50 m/s, hence simulating the variety of Angle of attacks of the aircraft. The of $C_D$ and $C_L$ results of each run was tabulated and were later experimentally verified through glide tests.

Table 1: The Turbulent Flow CFD computation results

<table>
<thead>
<tr>
<th>Speed (m/s)</th>
<th>CD</th>
<th>CL</th>
<th>Cm</th>
<th>Drag/N</th>
<th>Lift/Kg</th>
<th>Mm/Nm</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>0.034291</td>
<td>0.202584</td>
<td>0.01685</td>
<td>0.78301</td>
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<tr>
<td>20</td>
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<tr>
<td>30</td>
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<tr>
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<tr>
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<td>-8.25977</td>
</tr>
</tbody>
</table>

Speed (m/s) | CD     | CL     | Cm   | Drag/N  | Lift/Kg  | Mm/Nm  |
<table>
<thead>
<tr>
<th></th>
<th></th>
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<th></th>
<th></th>
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<tbody>
<tr>
<td>0</td>
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<td>0.78301</td>
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<td>1.604231</td>
<td>29.22084</td>
<td>-2.35938</td>
</tr>
</tbody>
</table>
These theoretically obtained Forces and Moments were later verified in glide tests (Chapter 8) to be very accurate and reliable enough to be used for the next stage of our project, the derivation of stability derivatives for optimizing flight controls.

Figure 26: Polar Plots of Aerodynamic Coefficients

When graphed vs. velocity, these parameters can show you if an aircraft has enough lift to fly and we can identify a flight speed range. The "lift polar" shows the lift coefficient $C_L$, plotted versus the angle of attack. From this we found that the $C_{L_{\text{max}}}$ of the airfoil was 1.295 and the corresponding angle of attack to be 37-39 degrees, beyond which the stall behavior of the aircraft comes to play. This is significant as we now have successfully identified the flight envelope of this airfoil for high angle of...
attack flight. Hence, the aircraft would stall and start to drop like a stone, when it reaches angles of attack above 39 degrees. (Evidence of this behavior has been recorded on video during flight testing).

The lift against velocity curve shows the speed required to maintain flight. It was thus found that the Golden Eagle could generate a lift force of 4.646 N at a cruise speed of 10 m/s and encountered a turbulent drag of the magnitude of 0.783N.

- Maximum specified flight weight of Golden Eagle : 1200g
- Targeted Equipment load (Autopilot, GPS, Camera etc) : 250g
- Lift Force calculated at 10 m/s : 4646g × 9.81 N
- Required Thrust to overcome drag at 10m/s: > 0.783 N

Hence, with the detailed lift characteristics of the aircraft identified, we discovered that it can actually carry approximately 4 times its specified load and still fly without stalling or crashing. The drag force at this speed also guided us in propulsion sizing (Chapter 9), ensuring the trust force that the new power plant provides can overcome the induced drag and provide a speed range of between 10 - 15 m/s for straight and level flight.

Having high thrust/weight and lift/drag ratios are not enough to guarantee that a plane is capable of steady flight. Without properly balanced moments about the craft’s center of gravity even the smallest of perturbations to the vehicle’s flight path can potentially send the plane tumbling out of control. (CG balancing done in Chapter 6).
Chapter 5: Prototype Fabrication

5.1 Reason for Reverse Prototyping

The physical replica of the UAV is required as damage is foreseen during the flight-testing phase. It would be unwise to damage the original given model, as it is very expensive. Unable to match this particular wing with any of the standard NACA airfoils present, we had to generate a full 3 Dimensional CAD model of the craft from scratch. Also, the simplistic construction drawings provided could not accurately tell us the wing curvature at the concave leading edge and at the convex trailing edge.

Construction of the wing and fuselage were one of the toughest challenges we faced. We wanted our wing to be as smooth and accurate as possible. Also, an easily repeatable wing fabrication process could help us during flight testing in case the wing was damaged beyond repair and a new wing needed to be made. These features require a Reverse prototyping process, where we construct molds and tools specifically to re-construct the golden Eagle airfoil. This is almost the exact opposite design principle, where we dimension an existent airfoil and then design the procedure to build it. Since the fuselage is home to all of the expensive components, it needs to be built strong enough to protect the equipment inside during an impact. Strength, light-weight, and low-cost are nearly contradictory terms. Only a few materials available to us were considered. Among them were carbon fiber, glass fiber, foam and balsa wood.
5.1.1 Model Dimensioning

The model was then sectioned and sliced at critical intervals to obtain the exact structural coordinates to be used to design and construct the wings.

The fuselage dimensions were easily measured externally, as it did not contain any complex geometries.

5.2 Material Research and Selection

The original Golden Eagle wings were made of Kevlar composite fibers. Kevlar is a very light and extremely strong and tough material. But unfortunately, it is very costly and could not be purchased in small quantities. Material research had to be done to find an equally durable and light material to build our prototypes for testing. Because design development was heavily dependent on flight-testing, the ease, speed, and precision of manufacturing and repair was a fundamental consideration about the materials chosen and the manufacturing procedure adopted. All components were deliberately determined to be modular and are meant to break away during impact. This ensures minimal damage by allowing us to localize the damage to easily replaceable components (e.g the nose), hence reducing repair costs and time.
Traditional ways of making wings involves making reinforced cross-sectional spars covered with a plastic, heat-shrinking material or some other similar type of covering. Some problems with this design are its inaccuracy and its fragility. Also, aerodynamically speaking, the form of the airfoil is greatly compromised for the sections of the wing that lie in between the spars.

Although balsa wood and foam are conventional materials, they could not be adopted here due to the complex and unconventional airfoil shape (distributed camber) that we had to model and also due to the non-availability of precise contour machining tools. Various materials such as low and high density foam, stiff ¼-1/2 in. cardboards, paper marches and laminate resins together with different manufacturing processors were experimented with initially, primarily due to their ease of availability and extreme low cost. But unfortunately, they were either too heavy or not rigid enough to take the required wing loading. Hence we started looking into composites as a viable alternative. A comparison was made between different types of fibres and resins to choose the suitable one in terms of pricing, weight and mechanical strength.

Table 2: Comparison of Fiber properties (Ref: Hull & Clyne)

<table>
<thead>
<tr>
<th>Fiber</th>
<th>Cost</th>
<th>Density (Mg/m³)</th>
<th>Tensile Strength (GPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Carbon Mat</td>
<td>High</td>
<td>1.95</td>
<td>2.4</td>
</tr>
<tr>
<td>Glass Mat</td>
<td>Low</td>
<td>2.56</td>
<td>2.0</td>
</tr>
<tr>
<td>Kevlar Mat</td>
<td>High</td>
<td>1.45</td>
<td>2.3</td>
</tr>
</tbody>
</table>

Table 3: Comparison of Resin properties (Ref: Hull & Clyne)

<table>
<thead>
<tr>
<th>Resin</th>
<th>Cost</th>
<th>Density (Mg/m³)</th>
<th>Tensile Strength (GPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Polyester</td>
<td>Low</td>
<td>1.3</td>
<td>40-90</td>
</tr>
<tr>
<td>Epoxy</td>
<td>High</td>
<td>1.2</td>
<td>35-100</td>
</tr>
</tbody>
</table>
Finally, we singled out single ply bi-directionally laid tissue glass fiber (GFRP) as GFRP combined the mechanical properties of both the plastic resin as well as the strengthening fibers to give high rigidity, superior strength-to-weight ratio and displayed excellent mechanical properties upon impact—a crucial consideration for a UAV without landing gear mechanisms. It is also low in cost and readily available.

5.3 Structural Construction

5.3.1 Tissue Fiber Laying

Reusable male and female clay molds were created and checked for consistency against the acquired wing curvature dimensions. The glass fiber framework was then laid on the molds and covered with a thin layer of synthetic polymer (Ethylene Glycol, wt. % 99.9 - Polyester). Specifically measured quantities of resin were applied equally on each of the two wings, maintaining symmetry in weight. The viscous resin was poured down on the wing, with the mold propped vertically up. This ensures an even distribution of resin throughout the cast. It was then allowed to drip and air dry in an enclosed area. This procedure gave a smoother and more even exterior finish compared to the conventional method of brushing on the polyester. The entire manufacturing process is highly repeatable with the usage of durable and reusable molds and cost effective readily available materials.

![Figure.28. Pouring of the resin](image)

![Figure.29. GFRP right wing bottom shell](image)
5.3.2 Wing and Fuselage

A hand lay-up method was used to create the two bottom halves of the UAV’s wings. Via dimensional slicing we had the exact profile of sectional ribs which we cut out on balsa ply. Sections of the wing (closer to the fuselage) that had a high degree of span-wise curvature had more ribs assigned while those closer to the wing tips had fewer. This distributed method, allowed a more accurate modeling of our M-shaped wing allowing smoother transitions along the lateral curves. A thin layer of film is then wrapped onto the balsa wood profiles and the fiberglass bottom to give the complete airfoil shape.

The fuselage is built with light weight balsa wood hollowed out to house the equipment, and the modular nose section was made out of High density foam, capable of deforming and absorbing shock during impact. The nose was carefully anchored to the fuselage using pins that easily cut through the balsa fuselage, allowing the nose to break away during impact. A similar approach was used by employing short carbon rods as interfaces when attaching the wings to the fuselage.

5.4 Assembly
Chapter 6: Estimation of CG Position and Mass of Inertias

6.1 Aerodynamic Centre and Estimation of CG Position

Moment Contribution for the payload

- Total Moment For Payload = -0.3815 Nm
- Moment Coefficient of Airfoil = 0.01685
- Moment of Airfoil = +0.3848 Nm
- Residual Nose-up Moments = 0.003 Nm (Balanced manually with ballast)

The equipment was strategically placed in the fuselage to balance out the residual moments we discovered in our CFD analysis. Due to the symmetrical design of the aircraft, we concentrated only on the longitudinal positioning of the cg. Laterally, the craft was found to be off-balanced by a few grams (11g) when we experimentally balanced the craft on a pivot. But this was unavoidable, given our fibre laying technique and the vicious e resin used. But, this was easily resolved by manually off-setting the right wing with ballasts. (Extended Mass Table in Appendix G)
We also know that the CG must be located ahead of the ac, if we want a self stabilizing wing as the profile we are using is reflexed. Via the above theoretical estimation method, we found that the c.g is located 0.395m from the tip of the nose.

Stability is a very important criterion in the design of aircraft. For flying wings, two conditions must be met for longitudinal stability.

\[ C_{m\alpha} < 0 \, \& \, C_m > 0 \]

The validity of this expression is easy to see: as an increased pitch angle should be counteracted with a negative nose down pitching moment. The root airfoil has a pitching moment near zero; hence the normal down force required by the wing tips is not great. As shown in our \( C_m \) vs \( \alpha \) CFD curve, is that the gradient is negative \( C_{m\alpha} < 0 \) and the graph intersects the x-axis on the positive end \( C_m > 0 \).

Furthermore, the \( C_m \) values for this unconventional airfoil are much lower than those compared with conventional airfoils of similar configuration. (In Appendix F).

\[ \frac{C_{m\alpha}}{CL\alpha} = \frac{-0.0911}{2.0985} = -0.0434 \]

The \( C_{m\alpha} / CL\alpha \) calculation tells us where our aerodynamic centre lies, the point where the moment acting on the body is independent of the angle of attack, and since this is a flying wing with a comparatively small central fuselage which also rides the wing profile, we conclude that the neutral point too lies at the AC location calculated. The negative value tells us that the AC actually lies behind the CG location by 0.0434m.
Es
timation of CG Position and MOI

To experimentally verify our calculations, the conventional method of CG determination was employed - the entire assembled model was mounted on a self constructed level pivot, with a broad and sharp edge, and shifted accordingly to attain the mass centre of the individual components.

Figure 3.3. Location of AC with respect to the CG

Theory= CG located 395mm from the Nose tip
Experiment= CG located 392mm from Nose Tip

3mm of discrepancy

Figure 3.4. Experimental verification of CG position
6.2 Static Margin

Static margin is the distance between the c.g and the neutral point which is the AC in our flying wing body. If the c.g is ahead of the AC, then the static margin is positive, and the static stability is positive by an amount that is related to the static margin. If the AC is behind the neutral point, then the static margin and static stability are negative (i.e. the model is statically divergent, if you pull the nose up it pitches up even more).

Mean aerodynamic chord, MAC = 0.465m (Geometrical derivation in Appendix E)

Static Margin Calculations,

c.g ahead of AC = +0.0434m

\[
\text{Static Margin} = \frac{0.0434}{0.465} \times 100\% = 9.33\% \text{ of MAC}
\]

6.2 Mass of Inertia Estimation

The equations for moment of inertia, are also referred to as “second moment” equations. This is due to the squared moment arm that multiplies each infinitesimal volume during the integration. In the case of the \( I_{xx} \), the distance from the x-axis is the moment arm to be squared, and due to the Pythagorean Theorem, this squared distance is \( y^2 + z^2 \).

\[
I_{xx} = \rho \int \left(y^2 + z^2 \right) dV
\]

The same method is used for the other moments of inertia. But in order to safe computation effort in extensive integration, we can approximate the Inertias with the geometric summation of the various components of different masses in the structure,
as per equations (1)-(6) (Ref: Jon Roskam). We must assume that each component has a constant density and mass distribution throughout. The detailed integration formulae are illustrated in the appendix D.

\[
\begin{align*}
I_{xx} &= \sum_{i=1}^{i=n} m_i \left( (Y_i - Y_{cg})^2 + (Z_i - Z_{cg})^2 \right) \\
I_{yy} &= \sum_{i=1}^{i=n} m_i \left( (Z_i - Z_{cg})^2 + (X_i - X_{cg})^2 \right) \\
I_{zz} &= \sum_{i=1}^{i=n} m_i \left( (X_i - X_{cg})^2 + (Y_i - Y_{cg})^2 \right) \\
I_{xy} &= \sum_{i=1}^{i=n} m_i \left( (X_i - X_{cg})^2 (Y_i - Y_{cg})^2 \right) = 0 \\
I_{yz} &= \sum_{i=1}^{i=n} m_i \left( (Y_i - Y_{cg})^2 (Z_i - Z_{cg})^2 \right) = 0 \\
I_{xz} &= \sum_{i=1}^{i=n} m_i \left( (Z_i - Z_{cg})^2 (X_i - X_{cg})^2 \right)
\end{align*}
\]

Symmetrical Aircraft

Figure.35. Geometrical Estimation of Inertias
The individual components were weighed on an electronic weighing machine and tabulated (Appendix G). By drawing measured reference lines to the individual components, we were able to attain the approximate Ixy and the Izz values. A similar method was also used for the side profile of the aircraft to attain the remaining masses of Inertia values. The Ixy and Iyz, values were found to be 0 due to aircraft symmetry about the longitudinal axis.

Verification of the MOI theoretical calculations were verified on computer software program SolidWorks\textsuperscript{TM}. The program, required an input of the density of the various components, (estimated from vol. and mass of components) and the to-scale CAD 3D model. We found that the orders of the mass of inertias were the same on both occasions and that they differed marginally in exact numerical value.

Thus, we obtain the Inertia Tensor as,

\[
\mathbf{I} = \begin{bmatrix}
I_{xx} & -I_{xy} & -I_{xz} \\
-I_{xy} & I_{yy} & -I_{yz} \\
-I_{xz} & -I_{yz} & I_{zz}
\end{bmatrix} = \begin{bmatrix}
0.07305 & -0.01718 & -0.01129 \\
-0.01718 & 0.13592 & -0.02418 \\
-0.01129 & -0.02418 & 0.14663
\end{bmatrix}
\]
Chapter 7: Glide Tests and Verification

7.1 Preliminary Glide Test

7.1.1 Glide Test Theory

![Diagram of glide slope angle definition]

Figure 36. Glide Slope Angle definition

\[
\text{Ratio: } \frac{\text{LIFT}}{\text{DRAG}} = \frac{L}{D} = \frac{1}{\tan \Omega} = \frac{d}{h}
\]

\[
L = 0.5 \times C_L \times \rho \times V^2 \times S
\]

\[
D = 0.5 \times C_D \times \rho \times V^2 \times S
\]

\[
\frac{L}{D} = \frac{C_L}{C_D}
\]

7.1.2 Experimental Set-up and Calibration

We needed an elevated platform to launch the craft from and at the same time be able to identify its glide slope. Hence, we chose a test site with an inclined measurable slope that opened up into a wide field as our launching platform. The plane was

![Image of test grounds with dimensions]

Figure 37 Glide Test Grounds
to be hand launched at zero angle of attack and at an approximate velocity of 10m/s. Two video cameras were set up on stands to help gauge the glide slope (one on each side of the glide slope).

There was a parallel path running, along the field and we aligned our launches along this path. This was primarily because; we could later use it as a straight gauge to measure the glide distance. Furthermore, the slope we launched from had a flight of stairs running alongside it (green railings of stairs, seen in picture above), and this served us in measuring the launch height.

To measure the launch speed and to regulate the speed as a constant throughout the experiment, we built our own simplistic speed gauge. It consists of an anchored spool of thread stuck to the bottom side of the fuselage. The side of the spool has a reflective tape stuck to it, so that we can read its rpm with a tachometer. The total length of the thread is 12 meters, and once the aircraft travels further then 12 meters, it gets dislodged from the spool and follows the craft. The spool is free rotating and the thread used is very light, causing negligible resistance during flight. The loose method of attaching it to the aircraft with a tape also allows it to get dislodged easily if the tension is too great.

Figure 38: Speed Measurement Setup
7.1.3 Test Environment

- Glide Velocity: 10 m/s
- Conditions: Static conditions – Windless
- Launching Mechanism: Hand Launched
- Engine Status: Mounted with propeller, but unfeathered.
- Flight Weight: Fully equipped with dummy weights, Weight total = 1569g

7.1.4 Aero Coefficient Verification

- Calculations:
  - Simulation Results: \( \frac{C_L}{C_D} = \frac{0.202584}{0.034291} = 5.91 \)
  - Glide Test Results: \( \frac{C_L}{C_D} = \frac{d}{h} = \frac{20.000}{3.203} = 6.14 \)

- Theoretical vs Experimental value: 3.72 % error

7.2 Discussion

Through the glide tests performed above we were able to verify that the theoretical simulation results we had attained thus far, were indeed accurate and that our CAD model and simulation environment were indeed realistic. These aerodynamic coefficients together with the CFD environment could now be used by Mr. Low for the next stage of our project, to get the aerodynamic derivatives. This reassures us of the integrity of the reverse engineering procedure adopted in aero. data generation. Another appreciable result of the glide tests is that the prototype we built does structurally represent the aerodynamic characteristics of the Golden Eagle accurately in that it glides flawlessly, along the calculated glide slope, reassuring our efforts in
the structural reverse engineering procedures adopted too. Finally, we were able to prove that with the correct c.g positioning and weight and balance matching done, the Golden Eagle could in fact carry in access a payload of about 260g.

With, these results we were certain that our project had indeed progressed in the right direction so far and our main efforts now would be in propulsion research and selection to provide sufficient lift generating thrust for straight and level flight.
Chapter 8: Propulsion System Integration

8.1 Propulsion Systems

8.1.1 Theoretical Analysis

Generation of thrust during flight requires the expenditure of power. In steady level flight, the thrust is equal to the aircraft drag.

\[
\text{Power} \equiv \text{Thrust} \times \text{Vel.} = \text{Drag} \times \text{Vel.}
\]

Similar to airfoils and wings, the performance of propellers can be described by dimensionless (normalized) coefficients. A propeller is described in terms of \textit{advance ratio}, \textit{thrust coefficient}, and \textit{power coefficient}. The relevant equations are as follows,

\[
\begin{align*}
\text{Thrust:} \quad c_T &= \frac{T}{\rho \cdot n^2 \cdot D^4} \\
\text{Power:} \quad c_P &= \frac{P}{\rho \cdot n^2 \cdot D^4} \\
\text{Advance Ratio:} \quad \frac{v \cdot nD}{n \cdot D} &= \frac{v}{n \cdot D} \\
\text{Efficiency:} \quad \eta &= \frac{v}{n \cdot D} \cdot \frac{c_P}{c_T}
\end{align*}
\]

Where,

\[
\begin{align*}
v \quad &\text{velocity} \quad \text{m/s} \\
D \quad &\text{diameter} \quad \text{m} \\
n \quad &\text{revolutions per second} \quad \text{1/s} \\
\rho \quad &\text{density of air} \quad \text{kg/m}^3 \\
P \quad &\text{power} \quad \text{W} \\
T \quad &\text{thrust} \quad \text{N}
\end{align*}
\]

For a given battery voltage, the shaft power and motor efficiency depend on the rotation rate (rpm). Figure 29 shows this dependence for our SP brushed motor.

![Figure 39: \(P_{\text{shaft}}\) and \(\eta_m\) versus \(\Omega_m\) for a Speed-400 motor, voltage of 12V](image)
The propeller then converts the shaft power to thrust power as such,

\[ P = \eta_p P_{\text{shaft}} = \eta_p \eta_m P_{\text{elec}} \]

We could measure the electrical power output, the motor rpm and the shaft power inputs, but that would still require us to know the propellers and motor efficiencies. Since this data was not readily available to us from the motor vendors, we opted to directly measure the thrust force with our own experimental set-up.

### 8.1.2 Experimental Set-up

![Figure 40. Thrust measurement: Test Stand Setup](image)

Procedure: 1) We tested two motor units primarily, the Himax and the SP.

2) Each with 5 different pusher propellers of various pitch and diameter.

3) The downward rotating moments are measured for 3 thrust settings.

4) Voltage drawn per thrust setting is also measured with a voltmeter.

5) Airspeed behind the propeller is also measured with an anemometer.

6) Results are tabulated and discussed.
8.1.3 Analysis of Data

Goals: 1) Selecting correct Electric motor and propeller combination

2) Power requirements on board (Number of batteries needed)

The calculations involve moment balancing,

\[ \text{Measured weight} \times 0.5 \times 9.81 (\text{g}) = \text{Thrust} \times 0.25 \]

Table 3: Thrust experimental results

<table>
<thead>
<tr>
<th>Motor</th>
<th>Thrust (N)</th>
<th>Prop</th>
<th>Idle</th>
<th>Mid</th>
<th>Max</th>
</tr>
</thead>
<tbody>
<tr>
<td>Himax .15kg</td>
<td></td>
<td>9x6</td>
<td>0.706</td>
<td>4.571</td>
<td>6.887</td>
</tr>
<tr>
<td></td>
<td></td>
<td>8x7</td>
<td>0.471</td>
<td>4.042</td>
<td>5.709</td>
</tr>
<tr>
<td></td>
<td></td>
<td>8x4</td>
<td>0.706</td>
<td>4.258</td>
<td>6.416</td>
</tr>
<tr>
<td></td>
<td></td>
<td>7x5</td>
<td>0.746</td>
<td>3.551</td>
<td>4.944</td>
</tr>
<tr>
<td></td>
<td></td>
<td>6x4</td>
<td>0.314</td>
<td>1.844</td>
<td>2.511</td>
</tr>
<tr>
<td>GS .17kg</td>
<td></td>
<td>9x6</td>
<td>0.530</td>
<td>4.277</td>
<td>6.533</td>
</tr>
<tr>
<td></td>
<td></td>
<td>8x7</td>
<td>0.275</td>
<td>3.826</td>
<td>5.140</td>
</tr>
<tr>
<td></td>
<td></td>
<td>8x4</td>
<td>0.530</td>
<td>4.252</td>
<td>6.318</td>
</tr>
<tr>
<td></td>
<td></td>
<td>7x5</td>
<td>0.628</td>
<td>3.571</td>
<td>4.768</td>
</tr>
<tr>
<td></td>
<td></td>
<td>6x4</td>
<td>0.177</td>
<td>1.766</td>
<td>2.237</td>
</tr>
</tbody>
</table>

Extended tabulation of voltage, airspeed and moments are shown in Appendix B.

8.2 Discussion

The thrust force generated for the motors depend heavily on the type of propellers we used. While higher pitched propellers provided an increased thrust, the increase in propeller diameter proved to be a more significant factor due to their xD^4 factor of influence in the Thrust formula. We chose the Himax motor coupled with the 9 inch diameter, Air screw pusher propeller with a pitch of 6, because it was lighter and also delivered the highest thrust at all power settings and could overcome drag forces encountered by the craft up to angles of attack of 40 degrees. It also proved to be the most efficient, in terms of battery power consumption. It ran at 100% throttle setting on 4, 9-volt, 50 mAh NiCd cells for approximately 29 mins.
Chapter 9: Flight Testing

9.1 Outdoor Testing: Powered

9.1.1 Test Sites

Testing was performed in the following venues.

a) Open air area at Jurong East
b) Open air area at Marina South
c) Open field at Ports Down Road

A large area was sought in order to allow the craft an unobstructed area to maneuver during the flight tests. This not only allows a more realistic assessment of the prototypes’ performance and dynamic characteristics on-site on in-video, but also prevents the craft from damage when crashing on walls. With no landing gears the Golden Eagle prototypes had to either fly into a recovery net or crash land on soft ground. Lack of man-power to mend the large nets forced us to use the later option.

9.1.2 Test Routines

Fully integrated with RC control unit, a dummy camera weight and the auto-pilot system we hand launched the prototype from an elevated terrain, so that it would have enough ground clearance to glide and climb altitude. The auto pilot card on board, collected flight velocity and altitude data which we retrieved once the craft landed.

We measured flight endurance by constantly measuring the voltage drop in the batteries with a multi-meter, each time the plane is grounded. Flight distance is measure with a measuring tape, from the launch point to the landing site and flight duration is timed with a conventional stop watch. In addition, we documented every
single test attempt with video equipment, which were analyzed in the lab frame by frame to identify stall angles and response of control surface deflections.

9.1.3 Problems

Induced oscillations from hand launching, together with inconsistent launch angles, plagued the initial attempts and caused severe damage to the wings and fuselage of our first prototype during crashes.

Uponpatching the wing of the model, we discovered that it actually altered its aerodynamic characteristics and weight balancing (C.G and Inertia), so we had to build our second, third and subsequently our forth prototype throughout our extensive flight test routines. On our other prototypes, we paid greater attention to launching at a consistent angle of attack, ensuring both proper alignment of the thrust line and wings in the intended direction of flight.

9.2 Test Data Gathered

Test Data Collected:  1) Flight Velocity = 10.6 – 22.3 m/s  
                  2) Max.Flight altitude = 25m above launch ground  
                  3) Flight Weight = 1625g  
                  4) Flight Endurance = 12s  
                  5) Battery endurance = 29minutes  
                  6) Structural stability = Excellent  
                  7) Thrust = >Sufficient  
                  8) Noise Level = < Original  
                  9) Critical Stall Angle = 40-45 degrees
9.3 Flight Test Results

Lift Characteristics: 1) Achieved take-off
   2) Achieved altitude ascend to 25m
   3) Achieved Straight and level flight
   4) Increased Payload Margin +260g

This verified that our reverse engineered prototype was structurally and aerodynamically stable and could generate enough lift to sustain flight. It could also effortlessly carry the additional payload of 260g and was balanced in the longitudinal axis, allowing it to attain self stabilized level flight. A recorded flight ceiling of 25m was also attained.

Thrust Characteristics: 1) Achieved take-off speed of >10m/s
   2) Max AoA 40-45 degrees
   3) Straight and level flight
   4) Cruise conditions of 8-12m/s
   5) Max Recorded Speed 22.3m/s
   6) Low motor noise levels

The new propulsion system allowed cruise sustained speeds of 8-12 m/s and also provided sufficient thrust to overcome drag even during high angled of attack loops. It also proved to be quieter than the original one and displayed a full throttle flight endurance of 29 minutes.

Visual evidence of the flight testing is available in video. Interested readers are requested to refer to the attached CD-R, to obtain understanding of the following discussion.
9.4 Discussion

From our flight test we were able to deduce that the structural aerodynamics of the craft, the propulsion system and c.g positioning were precise, it that the aircraft could take-off climb, and fly straight and level. A stable platform which replicates the Golden Eagle is now available for Mr. Low to test his control systems.

Sighting the, slow human response to the aircraft perturbations and the rapid respond rates of the aircraft in the control axis, we decided to fly with the aid of the micro pilot card. Together with Mr. Low, we integrated the system he designed on our 4th prototype and prepared it for autonomous, way point (2-point) flight. Once the aircraft was airborne Mr. Low activated the autopilot routine and proceeded to test the onboard control system. We were able to see that the aircraft trying to auto correct itself in the air and we have video evidence of autonomous flight over a short distance of 50 m, but these results could not be repeated over longer test distances. Sustaining stable controlled flight in the presence of wind perturbations proved to be difficult, even with autonomous flight as it responded extremely sensitively to control inputs making it difficult to maneuver in the air.
Chapter 10: Conclusion

This project has successfully validated the usefulness of reverse engineering in aerodynamic analysis of unconventional small scale aircraft structures. Through a realistic CFD environment the $C_L$, $C_D$, and $C_M$ values for the Golden Eagle’s unconventional reflex airfoil were generated and verified via glide tests to be accurate and consistent. This revealed the self stabilizing and increased lift characteristics of the airfoil design, which are an asset to Cradence Services, who had limited idea of their airfoil’s characteristics till today. We were also able to completely reproduce the airfoil structure and prototype four modular, aerodynamically identical replicas of the UAV, at the fraction of the original cost for testing, and hence incurred no damage at all to the original expensive model given to us.

Studying the lift and drag polars, the payload margin was duly optimized by 20-23 % at flight speeds of 10 m/s. Stability and weight balancing procedures too gave the correct estimates of the centre of gravity, aerodynamic centre and the Inertia tensor, which could be used by my colleague to establish the equations of motion.

The propulsion studies showed that the combination of propeller geometry and power of the motor were crucial in delivering additional thrust. The new propulsion system finally integrated in the prototypes delivered approximately 3 times more than the previous attainable thrust effortlessly.

The final integrated aircraft could lift its own weight and fly in a level configuration and ascend altitude displaying key aerodynamic characteristics. This allowed Mr.Low
Conclusion

to establish and test some key characteristics of his onboard control system, in the context of the prototypes dynamic behaviors.

Throughout this multidisciplinary project we have achieved our objectives by,

- Developing, validating, and applying aerodynamics models for the analysis and design of realistic, three-dimensional configurations for the aero Industry.
- Understanding the impact of uncertainty and variability on aerodynamic predictions and the resultant impact on aircraft system performance estimates;
- Computational Fluid Dynamic Simulations;
- Adopting the complimentary roles of theory, experiment, and computations in aerodynamic analysis and design.

Figure 44: Flight Testing of Prototype-4 with 6x4 Propeller and GS Motor
Recommendations

Chapter 11: Recommendations

Future recommendations would include the following steps.

a) Familiarizing oneself with Gambit, the CFD preprocessor supporting Fluent. This would be useful in assembling CAD models directly before creating the mesh, without other CAD programs such as CaTia or Solidworks as an interface, thus avoiding numerous part file exports and cumbersome “mesh tidying”.

b) Develop more advanced techniques for composite parts construction. Future students may attempt this as part of an UROP project to support their future FYP work to ensure sufficient time. Parts that can benefit from this skill include lightweight fuselages, airfoils and custom-made propellers.

c) Working with more leading edge materials such as Carbon or Kevlar fibers. This would allow us to generously fabricate both the bottom and top airfoil shells without much concern for weight.

d) A mechanical Launcher can be designed and built to allow consistent launching, each and every time, at an adjustable Angles of attack and velocity.

e) A less destructive flight test approach can be undertaken by proper on ground preparation of recovery nets prior to glide and flight testing.
List of References


10. Fletcher H.S., “Experimental Investigation of Lift, Drag and Pitching Moment of Five Annular Aerofoils:, Langley Aeronautical Laboratory, Langley Field, Va, USA, October 1957


12. “Gambit 2.0.4 Online Documentation”, Fluent Incorporated, 2001


Appendix A: CFD Results and Aerodynamic Plots

The Turbulent Flow CFD computation results are tabulated as,

<table>
<thead>
<tr>
<th>Speed</th>
<th>CD</th>
<th>CL</th>
<th>Cm</th>
<th>Drag/N</th>
<th>Lift/Kg</th>
<th>Mm/Nm</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>0.034291</td>
<td>0.202584</td>
<td>0.01685</td>
<td>0.78301</td>
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<td>-0.01683</td>
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<td>-0.01666</td>
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<td>0.189608</td>
<td>-0.01452</td>
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<td>69.27213</td>
<td>-5.30538</td>
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<td>0.189142</td>
<td>-0.01447</td>
<td>11.34138</td>
<td>107.9715</td>
<td>-8.25977</td>
</tr>
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</table>

<table>
<thead>
<tr>
<th>AOA</th>
<th>CD</th>
<th>CL</th>
<th>Cm</th>
<th>Drag/N</th>
<th>Lift/Kg</th>
<th>Mm/Nm</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0.034291</td>
<td>0.202584</td>
<td>0.01685</td>
<td>0.78301</td>
<td>4.625796</td>
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<td>10</td>
<td>0.03008</td>
<td>0.578148</td>
<td>-0.03617</td>
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<td>13.20143</td>
<td>-0.82592</td>
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<tr>
<td>20</td>
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<td>0.926144</td>
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<td>0.547247</td>
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<td>-1.06374</td>
</tr>
<tr>
<td>30</td>
<td>0.029058</td>
<td>1.223494</td>
<td>-0.05519</td>
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</tr>
<tr>
<td>40</td>
<td>0.060669</td>
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<td>1.38532</td>
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</tr>
<tr>
<td>50</td>
<td>0.070256</td>
<td>1.279707</td>
<td>-0.10333</td>
<td>1.604231</td>
<td>29.22084</td>
<td>-2.35938</td>
</tr>
</tbody>
</table>

Due to fabrication constraints, the airfoil constructed had considerable surface irregularities, especially on the bottom side. From readings on low speed aerodynamics, this factor should be sufficient to trigger an early transition from laminar to turbulent flow. Hence, a suitable computational turbulence model must be selected and used, because of the high possibility that turbulent, rather than laminar, flow separation would occur the airfoil.

From the Notes on Numerical Fluid Mechanics, modeling of flow separation for aerofoils at low Reynolds number is among the major topics discussed. The key to successful modeling of turbulent boundary layer flow is the selection of a correct
turbulent model. In any case, the turbulent models possible are limited by those available within Fluent 6, which were state-of-the-art. However, due to insufficient time for the project, a detailed turbulence model could not be developed. Future projects can then take over from here in order to develop a workable turbulence numerical model. For this project, we created the turbulent environment with the Fluent preset turbulent setting to attain, rather successful results.

The accompanying table format provides probably the clearest comparison of the listed airfoils. Looking at the numbers we can quickly examine the values and trends, and determine which section would be best for an anticipated flight envelope. The chart can also be used to establish performance comparisons between aircraft.

Laminar flow assumption CFD computations results are displayed for comparison. Note the low drag coefficient and force.

<table>
<thead>
<tr>
<th>AOA</th>
<th>CD</th>
<th>CL</th>
<th>Cm</th>
<th>Fx</th>
<th>Fz</th>
<th>Mm</th>
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<tr>
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<td>0.010842</td>
<td>0.542473</td>
<td>0.03161</td>
<td>0.250743</td>
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<tr>
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<td>0.008478</td>
<td>0.915872</td>
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<td>0.196087</td>
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<td>0.039565</td>
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<td>1.48141</td>
</tr>
<tr>
<td>0.698132</td>
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</tr>
</tbody>
</table>
Appendix A

Plots of Turbulent Aerodynamic Forces and moments,

Lift Force vs Angle of Attack

Drag Force vs Angle of Attack

Moments vs Angle of Attack
Polar plots of the aerodynamic coefficients,

- Coefficient of Drag vs Velocity
- Coefficient of Lift vs Velocity
- Coefficient of Moment vs Velocity
Coefficient of Lift vs Angle of Attack

Coefficient of Moments vs Angle of Attack

Coefficient of Drag vs Angle of Attack
And finally the Lift and Drag Polar,

These curves represent the results of the reverse engineering procedure and our understanding and increased knowledge of the Golden Eagle’s airfoil. We see unique characteristics such as increased lift and reduced drag at speeds of 10-20 m/s.
# Appendix B: Typical Airfoil Section Characteristics

Airfoil Section Characteristics  
Two dimensional properties only  
Reynold’s Number = 6,000,000  
Lift coefficient values for l/d characteristics: .1, .4, .6

<table>
<thead>
<tr>
<th>Airfoil Shape</th>
<th>Drag Coefficient Values</th>
<th>Max. Cl</th>
<th>L/d @ Cl = .1</th>
<th>L/d @ Cl = .4</th>
<th>L/d @ Cl = .6</th>
<th>Max. Section Cl</th>
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</thead>
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<td>.0060</td>
<td>.0068</td>
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<td>.0065</td>
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<td>.0055</td>
<td>.0055</td>
<td>.06</td>
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<td>72.72</td>
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</tbody>
</table>

The following are a sample of Harry Riblett’s sections for general aviation.
Appendix C: Propulsion system selection

Speed 400

<table>
<thead>
<tr>
<th>Motor</th>
<th>Watts (out)</th>
<th>Volts</th>
<th>Load RPM</th>
<th>Prop Size</th>
<th>Amp</th>
<th>Thrust</th>
<th>Case long</th>
<th>Dia. mm</th>
<th>Shaft mm</th>
<th>Wt. gm</th>
<th>Uses</th>
</tr>
</thead>
<tbody>
<tr>
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<td>50</td>
<td>6</td>
<td>14.2k</td>
<td>6x3</td>
<td>17</td>
<td>300g</td>
<td>38</td>
<td>27.8</td>
<td>2.3</td>
<td>73</td>
<td>Speed 400 type, to 24 oz, to 275 sq.,</td>
</tr>
<tr>
<td>4.8V</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>gliders to 400 sq. in.</td>
</tr>
<tr>
<td>SPEED 480</td>
<td>48</td>
<td>7.2</td>
<td>16.7k</td>
<td>4.7x</td>
<td>10</td>
<td>240g</td>
<td>47.2</td>
<td>29.2</td>
<td>3.2</td>
<td>103</td>
<td>Speed 400 racers, to 20 oz, to 150</td>
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<tr>
<td>RACE</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>sq.in., 74mph speed</td>
</tr>
</tbody>
</table>

We initially approached this problem in a conservative manner, by trying out the cheaper and lower end brushed motor 400 series. But, evident from our first few experiments where it could not even sustain the torque of our 9x6 propellor, we decided to switch over to brushless motors instead, which are generally more efficient, lighter and powerful.

From the experiment we conducted in the lab, using the Himax 400 series motor we managed to arrive at the following data. The voltage drawn by the Himax 400 series motor together with its wind speed and thrust are tabulated as such, for different propellers ranging in diameter of 6-9 inches and pitches varying from 4 to 7.
The airspeed, thrust force and voltage drawn data is tabulated,

<table>
<thead>
<tr>
<th>Motor</th>
<th>Prop</th>
<th>Windspeed</th>
<th>Weight (Kg)</th>
<th>Voltage</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>Idle</td>
<td>Mid</td>
<td>Max</td>
</tr>
<tr>
<td>Himax</td>
<td>9x6</td>
<td>0.716</td>
<td>0.752</td>
<td>0.949</td>
</tr>
<tr>
<td></td>
<td>8x7</td>
<td>0.729</td>
<td>0.753</td>
<td>0.935</td>
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<td></td>
<td>8x4</td>
<td>0.729</td>
<td>0.765</td>
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<td></td>
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<td></td>
<td>6x4</td>
<td>0.723</td>
<td>0.739</td>
<td>0.817</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

We also experimented and tabulated the characteristics of the GS-400 motor,

<table>
<thead>
<tr>
<th>Motor</th>
<th>Prop</th>
<th>Windspeed</th>
<th>Weight (Kg)</th>
<th>Voltage</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>Idle</td>
<td>Mid</td>
<td>Max</td>
</tr>
<tr>
<td>GS</td>
<td>9x6</td>
<td>0.712</td>
<td>0.739</td>
<td>0.930</td>
</tr>
<tr>
<td></td>
<td>8x7</td>
<td>0.729</td>
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</tr>
<tr>
<td></td>
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</tr>
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<td>6x4</td>
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<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

We, had to compare, at least two top end motors to arrive at a fair decision. Requiring with aero enthusiasts on our field research trips to aero modeling shops we discovered the
GS motor. Unfortunately, technical specifications for this particular model was not readily available, hence we relied purely on experimental data.

We found that the thrust specifications for the smaller sized propellers were insufficient for both motor at all three thrust settings, but as the pitch and diameter increased, the differences in performance became more divergent. Although both motors provided the required thrust, the Himax motor displayed longer sustainability, in that it could run at 100% throttle setting for much longer. Furthermore the himax was also a lighter motor and good option for our power plant.

Via performance matching is so that we optimized the performance of the propulsion system with the 9x6 pusher propeller to give us maximum thrust.

Our motor of choice is the himax, series; we have displayed here its recommended usage and specifications that we discovered.

**Operation:**

1. Himax Brushless motors require brushless sensor less speed controls. Failure to use a brushless sensor less electronic speed control (ESC) can result in damage to the motor and/or ESC. A Castle Creations Phoenix series ESC is recommended for best
performance. The standard setting for timing advance is recommended for best operation.

2. The three wires of the motor can be connected to the three output leads of the ESC in any order. Check the direction of rotation of the motor. If the motor spins in the wrong direction switching any **two** of the motor wires will reverse rotation. Be sure to insulate the wires to prevent shorting which may damage the ESC.

3. Allow for proper cooling of the motor during operation. With extremely high capacity batteries, care must be taken to prevent excessive motor temperature. Overheating of the motor is not covered under warranty. Insufficient cooling can result in overheated motors, even when operated at moderate power levels.

**Features:**

High Efficiency - High Power - Light-weight - Replacement for 500-600 Motors

**Specifications:**

Weight: 138g, (4.9oz), Motor Only

Max Power: 400W, (This is dependent on several factors)

Max RPM: 50,000 RPM

Diameter: 28mm, (1.10”)

Length: 36.8mm, (1.45”)

Shaft Diameter: 4.0mm, (.1575”) or 3.17mm (.125”)

Mounting Screw Thread: 3.0mm, max depth 4.0mm, on 16mm, (.625”) bolt circle

Maximum Case Temperature: 100°C, (212°F)

**Electrical Specifications:**

HA2825-2300 Kv = 2300, Rm = .078, Io = 0.5, Efficient Operating Current = 4-23A
Himax HA2015 series brushless motors - Speed 280/300 replacement

- Slotless design for high efficiency. High performance replacement for Speed 280/300 type can motors,
- For light electric planes under 20oz or 3-D performance under 13oz,
- Fit most slowflyer/parkflyers - GWS planes, micro helicopters - Hummingbird & Piccolo, micro cars - mini T & HPI micro,

Himax HA2025 series brushless motors - Speed 370/400/480 replacement

- Slotless design for high efficiency. High performance replacement for Speed 370/400/480 type can motors,
- For light electric planes under 30oz or 3-D performance under 18oz,
- Perfect for 3-D aerobatic, pylon race, and any application maximum power/weight ratio is needed,

Himax HA2825 series brushless motors

- Slotless design for high efficiency. High performance replacement for 500/550 type can motor,
- Gear motors are suitable for small electric trainers, sport fun flyers,
- Direct drive motors are best for ducted fans & pylon racers where high rpm is a must,
- For light electric planes under 70oz.

HA2825-3600 Current vs. Voltage with Propeller

We are using the 3600 series motor, which displays current to voltage consumption as such, comparing against this graph, we were able to, select the power supply of 4 cells to drive the propulsion system.
Appendix D: Aerofoil concepts

Research of airfoils and planforms was conducted referencing several aerodynamics texts. Two of these texts, the American Institute of Aeronautics and Astronautics (AIAA) published *Fixed and Flapping Wing Aerodynamics for Micro Air Vehicle Applications* and their independently written *Model Aircraft Aerodynamics* by Martin Simons, contained a substantial amount of material concerning low-speed airfoil research. From these texts we researched several thin and thick airfoils.

Separation bubble on low speed airfoils (Ref: AIAA, pg.135)

Both texts stated that at low Reynolds numbers thin airfoils outperform thicker profiles. It might seem as though a very porous surface would create a substantial amount of extra drag. This is true for larger aircraft that fly at much higher Reynolds numbers. A smooth surface is a much better aerodynamic body at these speeds. However, at low speeds a smooth surface is not always the best choice. Low speed, laminar flows are prone to producing separation bubbles along the upper surface of a wing within a Reynolds
number range between 50,000 to 100,000. This separation bubble is a portion of the flow that temporarily separates from the wing. It is a location where the boundary layer of the flow reaches a very large proportion of the airfoil thickness, see Fig. above. While the separation bubble may not be a physical part of the wing itself, its impact on the L/D performance of the wing has the same effect as if it were. The flow over a wing with a separation bubble has significantly increased drag. One way to reduce or eliminate the negative effects of a separation bubble is by perturbing the flow around the airfoil before the separation bubble has a chance to form. Features such as wires, grooves, or small deformities make the laminar flow slightly more turbulent therefore altering the boundary layer. For Reynolds numbers less than 100,000 rough airfoils actually tend to have higher lift to drag ratios than smooth airfoils, see Fig. below.

![Fig: Cl/Cd vs Reynolds number for smooth and Rough airfoils (Ref AIAA, Pg5)](image)

Fig: Cl/Cd vs Reynolds number for smooth and Rough airfoils (Ref AIAA, Pg5)
Sweep and Twist

Basic lift distribution consists of lifting root, downloaded tip -> positive $C_m$.

Camber distribution

An airfoil is defined by first drawing a “mean” camber line. The straight line that joins the leading and trailing ends of the mean camber line is called the chord line. And the length of this chord line is called chord,c. To the mean camber line, a thickness distribution is added in a direction normal to the camber line to produce the final airfoil shape. Equal amounts of thickness are added above the camber line, and below the camber line. An airfoil with no camber (i.e. a flat straight line for camber) is a symmetric...
Appendix D

airfoil. Whilst, that with appositive camber infront and a negative camber behind is termed a reflexed airfoil.

**Center of Pressure:**

the center of pressure is defined as the point about which the pitching moment is zero. As the flow conditions change (example, angle of attack $\alpha$ changes), the center of pressure will change.

**Aerodynamic Center:** The aerodynamic pressure is defined as the point where the pitching moment (or the pitching moment coefficient) is independent of $\alpha$. That is, if we computed the pitching moment about the aerodynamic center,

$$\frac{\partial M}{\partial \alpha} = \frac{\partial C_m}{\partial \alpha} = 0$$
Appendix E: Weight and Balance Matching

One of the critical areas of conceptual design is that of weight and balance. As can be imagined, the total weight of the vehicle will determine if it can fly or not, but there are much more subtle aspects of weight and balance that play a vital role in the conceptual design of an aircraft. The term “weight and balance” refers to the mass properties of an aircraft and the resulting stability or lack thereof as a consequence of its mass properties. The term mass properties usually includes the following values: volume (or mass or weight), center of mass (or center of gravity), and the moments and products of inertia. The weight of the vehicle is trivial to calculate, but can be the difference between the success and failure of a mission. It is essential for calculating the residual moments by way of derived equations from a “free body” diagram. A free body diagram is used to visually sum all of the forces and moments on an object to determine its static equilibrium or it’s dynamic response due to a particular loading.

The distribution of the vehicle’s weight is also of major concern. The locations of the aircraft’s components and their corresponding weights will determine where the aircraft’s center of gravity is located. The center of gravity (cg) is the point where the object’s gravity force (or weight) acts. Also, it is around this point that the object naturally rotates if a torque is applied. If a force is applied to an object and the force vector does not intersect the CG, then a torque is also created about the CG proportional to the moment arm(perpendicular distance from CG to the force vector).
In the case of aircraft, as it flies through the air, each of its components experiences aerodynamic forces (not at the aircraft CG), thus creating torques about the CG. To balance these torques, generally a control surface is deflected a certain amount to produce a counter torque to nullify the overall moment felt at the CG. For this reason, the CG location is critical for balancing the vehicle.

Finally, the moments and products of inertia (MOI) of the aircraft are also determined by the distribution of the mass throughout the vehicle. Each component’s MOI and distance from the aircraft CG affect the overall MOI. The MOI are used in dynamic analysis of the aircraft and also to size the control surfaces. One example of a study where MOI are critical would be to determine the dynamic response of the aircraft to a sudden gust of wind.

The equations that command weight and balancing are,

\[ M = \rho V = \rho \int dV \]

\[ \bar{x} = \frac{\int x \, dV}{V} \]

\[ \bar{y} = \frac{\int y \, dV}{V} \]

\[ \bar{z} = \frac{\int z \, dV}{V} \]
These equations are a bit complex due to the integration involved in them. The c.g equations can also be referred to as “first moment” equations. This is due to the fact that each infinitesimal volume is multiplied by a moment arm. This moment arm adds the concept of position of each particle into the equation. The process requires that the moments (instead of masses) are summed, and then divided by the total volume. This is like taking an average of all the positions of the points in an object. The final result is the coordinates of the CG (or centroid).

The equations for moment of inertia, are also referred to as “second moment” equations. This is due to the squared moment arm that multiplies each infinitesimal volume during the integration. In the case of the $I_{xx}$, the distance from the x-axis is the moment arm to be squared, and due to the Pythagorean theorem, this squared distance is $y^2 + z^2$. The same method is used for the other moments of inertia.
The usefulness of this squared moment arm is derived from the fact that the amount of time needed for a point mass at a particular radius to reach a given speed is proportional to its mass and the square of the radius. The moments of inertia of an object can be understood as its resistance to rotation (angular accelerations and decelerations).

By assuming a constant mass throughout the different components we are analysing, we can use the simplified estimation equations, (Ref: Jon Roskam)

\[
\begin{align*}
I_{xx} &= \sum_{i=1}^{i=n} m_i \left( (Y_i - Y_{cg})^2 + (Z_i + Z_{cg})^2 \right) \\
I_{yy} &= \sum_{i=1}^{i=n} m_i \left( (Z_i - Z_{cg})^2 + (X_i - X_{cg})^2 \right) \\
I_{zz} &= \sum_{i=1}^{i=n} m_i \left( (X_i - X_{cg})^2 + (Y_i - Y_{cg})^2 \right) \\
I_{xy} &= \sum_{i=1}^{i=n} m_i \left( (X_i - X_{cg})^2 (Y_i - Y_{cg})^2 \right) = 0 \\
I_{yz} &= \sum_{i=1}^{i=n} m_i \left( (Y_i - Y_{cg})^2 (Z_i - Z_{cg})^2 \right) = 0 \\
I_{zx} &= \sum_{i=1}^{i=n} m_i \left( (Z_i - Z_{cg})^2 (X_i - X_{cg})^2 \right)
\end{align*}
\]

The Component masses are found in Appendix G and the geometrical drawing and solution follows below,
By drawing reference lines to the individual components, we were able to attain the approximate $I_{xy}$ and the $I_{zz}$ values. A similar method was also used for the side profile of the aircraft to attain the remaining masses of Inertia values. Due to the relative small masses and the large wing, realized that they were rather small. The $I_{xy}$ and $I_{yz}$, values were found to be 0 due to aircraft symmetry about the longitudinal axis.
The Inertia tensor was,

\[
\mathbf{I} = \begin{bmatrix}
I_{xx} & -I_{xy} & -I_{xz} \\
-I_{xy} & I_{yy} & -I_{yz} \\
-I_{xz} & -I_{yz} & I_{zz}
\end{bmatrix} = \begin{bmatrix}
0.07305 & -0.01718 & -0.01129 \\
-0.01718 & 0.13592 & -0.02418 \\
-0.01129 & -0.02418 & 0.14663
\end{bmatrix}
\]
Appendix F: Determining the Static Margin

Static margin is the distance between the C/G and the neutral point. If the C/G is ahead of the neutral point, then the static margin is positive, and the static stability is positive by an amount that is related to the static margin. If the C/G is behind the neutral point, then the static margin and static stability are negative (i.e.: the model is statically divergent, if you pull the nose up it wants to go up even more, shove the nose down and it wants to tuck).

The well known Geometrical approximate method of determining the MAC is illustrated for a Delta wing below,
Following the same procedure as for the delta wing we get,

\[ \text{MAC} = 0.465 \text{m} \]

c.g ahead of AC = +0.0434 m

\[ \text{Static Margin} = \frac{0.0434}{0.465} \times 100\% = 9.33\% \text{ of MAC} \]
## Appendix G: Component Weights Breakdown

<table>
<thead>
<tr>
<th>Components</th>
<th>Mass/g</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pusher Propeller</td>
<td>5</td>
</tr>
<tr>
<td>Motor</td>
<td>145</td>
</tr>
<tr>
<td>Left Servo</td>
<td>25</td>
</tr>
<tr>
<td>Right Servo</td>
<td>25</td>
</tr>
<tr>
<td>Servo wire and linkages</td>
<td>15</td>
</tr>
<tr>
<td>RC receiver</td>
<td>8</td>
</tr>
<tr>
<td>Speed Controller</td>
<td>12</td>
</tr>
<tr>
<td>Video electronics (camera, transmitter)</td>
<td>55</td>
</tr>
<tr>
<td>2 x Batteries (9-volt, 50 mAh NiCd)</td>
<td>250</td>
</tr>
<tr>
<td>1 x Battery (Micropilot card)</td>
<td>100</td>
</tr>
<tr>
<td>Micro Pilot Card</td>
<td>25</td>
</tr>
<tr>
<td>Micro Pilot Cables</td>
<td>4</td>
</tr>
<tr>
<td>Left Wing (with Glue)</td>
<td>295</td>
</tr>
<tr>
<td>Right Wing (with Glue)</td>
<td>302</td>
</tr>
<tr>
<td>Fuselage</td>
<td>150</td>
</tr>
<tr>
<td>Nose Section</td>
<td>52</td>
</tr>
<tr>
<td>Elevons</td>
<td>5</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>1569g</strong></td>
</tr>
</tbody>
</table>
### Appendix H: Camber Distribution of Airfoils

<table>
<thead>
<tr>
<th>REF</th>
<th>DESIGNATION</th>
<th>$C_m$</th>
<th>$a_{\theta}$</th>
<th>SECTION PROFILE</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>E 205</td>
<td>-0.046</td>
<td>-2.37</td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>E 205.inv</td>
<td>+0.046</td>
<td>+2.37</td>
<td></td>
</tr>
<tr>
<td>3</td>
<td>Symmetrical</td>
<td>0.000</td>
<td>0.00</td>
<td></td>
</tr>
<tr>
<td>4</td>
<td>EH 2/10</td>
<td>+0.00165</td>
<td>-0.74</td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>E 228</td>
<td>+0.0143</td>
<td>+0.34</td>
<td></td>
</tr>
<tr>
<td>6</td>
<td>E 230.Eppler/MTB 1/2</td>
<td>+0.053</td>
<td>+1.73</td>
<td></td>
</tr>
<tr>
<td>7</td>
<td>E 230.Panknin</td>
<td>+0.025</td>
<td>+1.73</td>
<td></td>
</tr>
</tbody>
</table>
Appendix I: Benefits and Disadvantages of Flying Wings

2.4 Benefits and Disadvantages of the Flying Wing

The advantages of the Tailless Flying wing aircrafts are,

a) Fuselage and tail sections create extra drag reducing performance.

b) Reduced weight due to absence of tail components, (servos, rudder etc.)

c) Lower lateral radar visibility (flatter profile) due to absence of vertical tails.

d) Simple structure. Supposed to be easily made using composite materials

However, the following disadvantages also apply

a) Due to absence of vertical tail surfaces the aircraft experiences on its longitudinal and vertical axes, relatively large inertia moments, while around its transverse axis it is much smaller.

b) Aerodynamic damping of the rotation movements around its transverse axis is less than that of conventional tailed aircrafts, leading to very sensitive and rapid movement in this axis. (Hypersensitive aircraft)

c) Unstable and difficult to handle as control surfaces are relatively close to c.g. (needs artificial stability resources – autopilot controls)