Design, Development and Manufacture of a Search and Rescue Unmanned Aerial Vehicle

Draft Report v 3.5

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Executive Summary

The Search and Rescue Unmanned Aerial Vehicle (UAV) was designed, developed and manufactured by a group of eight undergraduate engineering students from the School of Mechanical Engineering at The University of Adelaide. With backgrounds in Aerospace and Mechatronic disciplines, the students aimed to develop an autonomous UAV platform, with ground based control and surveillance capabilities. Dubbed 'UAV iSOAR' (UAV for intelligent Surveillance for Outback Aerial Rescue) the design was driven towards a search and rescue mission profile, with ultimate testing of the effectiveness of the platform to be conducted at the inaugural Australian Research Centre for Aerospace Automation (ARCAA) 2007 UAV Outback Challenge - Search and Rescue event.

The conceptual design of the airframe was derived using a classical approach, based on an extensive feasibility and statistical analysis of the global UAV industry. Driven by controllability and stability requirements, the airframe is of a conventional configuration, with a high wing, electric propulsion system, proprietary control system, and internal and external payload carrying capabilities. The aircraft is primarily constructed from composite materials with removable wings, undercarriage and hatches, allowing access to internal components and easy transportation.

The use of an electric propulsion system allowed for a reduction in the complexity of the dynamics of the aircraft, as well as development time of the prototype. The use of a brushless electric motor, and lithium-polymer battery technology, was designed to provide the UAV with a cruise speed of 90km/hr and an endurance in excess of 60 minutes, as dictated by the mission requirements.

The Micropilot 2028g proprietary autopilot system was integrated into the airframe to provide straight and level autonomous flight. The system had the scope to automate the vehicle in all regimes of flight, however integration problems resulted in this not being achieved by the project. During in-flight tuning of the system, interference between the autopilot communication link and RC receiver resulted in a crash, and a shift in the focus of work within the project towards resolving this interference problem.

The internal payload of the aircraft consists of an analogue video camera, with 450TV lines of resolution, and a field of view of 90 degrees. This camera was successfully used to stream live video footage to a ground station, providing the necessary information for successful surveillance missions.

An externally mounted payload system was designed and located on the aircraft’s undercarriage. Through the use of two servo motors, the system successfully deployed a 600mL bottle of water,
and thus demonstrated the capability of the aircraft in deploying mission specific payloads to designated targets.

The primary goals of this project were ambitious given the sequential order in which each goal needed to be accomplished. However, the resourcefulness of the group meant that these goals were able to be realised in full, even when unforseen problems arose. Several extended goals were also specified, and these were particularly ambitious. All of these goals were partially achieved, and the work completed by the project thus far provides a solid base for future work in the development of a fully autonomous UAV system.
Acknowledgments

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- City Holden

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Nomenclature

Acronyms

AR Aspect Ratio
ARCAA Australian Research Centre for Aerospace Automation
CASA Civil Aviation Safety Authority
CASR Civil Aviation Safety Regulations
CCD Charge-coupled device
CEP Circular error probable
CGR Climb Gradient
CPU Central Processing Unit
CSIRO Commonwealth Scientific and Industrial Research Organization
DSTO Defence Science and Technology Organization
EIRP Effective Isotropic Radiated Power
EMI Electromagnetic Interference
ESC Electronic Speed Controller
FBD Free Body Diagram
FHSS Frequency hopping spread spectrum
FOV Field of View
GPS Global Positioning System
ICE Internal combustion engine
IMU Internal Measurement Unit
INTA Instituto Nacional De Tecnica Aeroespacial
iSOAR Intelligent Surveillance for Outback Aerial Rescue
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LiPo  Lithium polymer
LoHR  Lines of horizontal resolution
LoVR  Lines of vertical resolution
Ni – MH Nickel Metal Hydride
MAC   Mean aerodynamic chord of main wing
MAV   Micro Air Vehicles
MTOW  Maximum take off weight
PCM   Pulse Code modulation
PID   Proportion, Integral and Derivative based control
QUT   Queensland University of Technology
RC    Radio Control
RF    Radio Frequency
TODR  take off distance requirement
UAS   Ultrasonic Altitude Sensor
UAV   Unmanned Aerial Vehicle

Symbols

A     Area, Aspect ratio
a     Lift curve slope of main wing
$A_{dB}$ Absorption loss (dB)
$A_{disc}$ Equivalent disc area of propeller
$A_{H}$ Aspect ratio of horizontal stabiliser
$a_h$ Lift curve slope of horizontal stabiliser
$A_i$ Area of element
$A_V$ Aspect ratio of vertical stabiliser
b     Span of main wing
$b_{aileron}$ Span of aileron
$B_{dB}$ Correction factor (dB)
$b_{\text{elevator}}$  Span of elevator  

$b_H$  Span of horizontal stabiliser  

$b_{\text{rudder}}$  Span of rudder  

$b_V$  Span of vertical stabiliser  

$C$  Chord length of main wing  

$\overline{C}$  Mean Aerodynamic Chord length of main wing  

$C_1$  Chord length of aileron, (wing root side)  

$C_2$  Chord length of aileron, (wing tip side)  

$C_{\text{aileron}}$  Chord length of aileron  

$C_{\text{battery}}$  Battery charge  

$C_{D0}$  Zero angle of attack drag coefficient  

$C_{D_{\text{induced}}}$  Induced drag coefficient of main wing  

$C_{\text{elevator}}$  Chord length of elevator  

$C_H$  Chord length of horizontal stabiliser  

$\overline{C}_H$  Mean aerodynamic chord length of horizontal stabiliser  

$C_{\text{Hinge Ei}}$  Chordwise position of hinge line of elevators  

$C_{\text{Hinge Rad}}$  Chordwise position of hinge line of rudder  

$C_{\text{Hinge Ail}}$  Chordwise position of hinge line of ailerons  

$C_L$  Main wing lift coefficient  

$C_l$  Airfoil section lift coefficient  

$C_{l_{a1}}$  Main wing local additional section lift coefficient  

$C_{l_b}$  Main wing local basic section lift coefficient  

$C_{L_{\text{cruise}}}$  Main wing lift coefficient at cruise speed  

$C_{L_{\text{des}}}$  Main wing lift coefficient on decent  

$C_{L_{\text{max}}}$  Maximum main wing lift coefficient  

$C_{l_{\text{max}}}$  Maximum airfoil section lift coefficient  

$C_M$  Main wing moment coefficient (quarter chord)  

$C_m$  Airfoil section moment coefficient (quarter chord)
\( C_{\text{root}} \) Root chord length
\( C_{\text{root}_H} \) Root chord length of horizontal stabiliser
\( C_{\text{root}_V} \) Root chord length of vertical stabiliser
\( C_{\text{rudder}} \) Chord length of rudder
\( C_{\text{tip}} \) Tip chord length
\( C_{\text{tip}_H} \) Tip chord length of horizontal stabiliser
\( C_{\text{tip}_V} \) Tip chord length of vertical stabiliser
\( C_V \) Chord length of vertical stabiliser
\( D \) Drag
\( d_{\text{Hinge}} \) Distance between control surface hinge line and control surface centre of load
\( \frac{dC}{d\left(\frac{1}{2}\right)} \) Change in chord length of main wing with respect to half span of wing
\( D_{\text{prop}} \) Propeller diameter
\( d_{zi} \) Vertical distance from centroid
\( \frac{de}{d\alpha} \) Main wing downwash angle derivative
\( e \) Oswald span efficiency factor
\( F_{\text{surface}} \) Force acting on control surface
\( f \) Frequency
\( G \) Conductivity relative to copper
\( g \) Gravitational constant, \(9.81 \text{ms}^{-2}\)
\( I \) Moment of inertia
\( I_{\text{motor}} \) Current drawn by motor
\( I_x \) Moment of inertia about the x axis
\( J \) Main wing angle of attack correction factor, polar moment of inertia
\( K \) Drag-due-to-lift factor
\( K_A \) Aerodynamic Terms
\( K_F \) Control surface load factor
\( K_T \) Thrust Terms
\( kts \) Knots
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$K_V$  Motor constant
$L$  Lift
$L_{airfoil}$  Lift - drag ratio of airfoil section
$L_{vehicle}$  Lift - drag ratio of aircraft
$L_{wing}$  Lift - drag ratio of main wing
$l_{servo}$  Actuation lever arm of actuator servo
$l_{surface}$  Actuation lever arm of control surface
$M$  Bending moment
$n$  load factor
$nm$  Nautical mile
$n_s$  safety factor
$P$  Motor power, Profile of the wing
$P_{battery}$  Battery power
$P_{max}$  Maximum motor power
$P_{motor}$  Motor Power
$P_{prop}$  Propeller pitch
$P_{Shaft}$  Motor shaft power
$P_{shaft_{cruise}}$  Motor shaft power at cruise speed
$P_{shaft_{max}}$  Maximum motor shaft power
$P_{thrust}$  Power provided to freestream from propeller
$Q$  First moment of inertia
$q$  Dynamic pressure
$q_s$  Dynamic pressure at stall velocity
$R_{dB}$  Reflection loss (dB)
$Re$  Reynolds Number
$S$  Main wing planform area
$S_{aileron}$  Aileron planform area
$S_{dB}$  Screening effectiveness (dB)
$S_G$  Ground roll distance

$S_H$  Horizontal stabiliser planform area

$SM$  Static margin

$S_{ref}$  Reference surface area

$S_{surface}$  Planform area of control surface

$SV$  Vertical stabiliser planform area

$S_{wet}$  Wetted surface area

$T$  Torque

$t$  Battery discharge time, skin thickness

$t_{mm}$  thickness in millimeters

$T_{static}$  Static motor thrust

$V$  Voltage, Volume, Shear force

$v_{cr}$  Nominal cruise velocity

$v_{crwind}$  Wind gust nominal cruise speed

$V_f$  Final Velocity

$V_H$  Volume coefficient of horizontal stabiliser

$V_i$  Initial Velocity

$V_{motor}$  Operating voltage of motor

$V_s$  Stall velocity

$V_V$  Volume coefficient of vertical stabiliser

$W$  Weight

$W_e$  Empty weight

$W_{payload}$  Payload weight

$W_{TO}$  Take off weight

$X_{acA}$  Neutral point of aircraft

$X_{acw}$  Neutral point of main wing

$x_{centroid}$  The centroid on the x axis

$x_H$  Horizontal stabiliser lever arm
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$x_i$  
Longitudinal position

$\mathbf{X}_V$  
Vertical stabiliser lever arm

$y$  
Lateral/spanwise position

$z_{\text{centroid}}$  
The centroid on the vertical axis

$z_i$  
Vertical position

Greek Symbols

$\alpha$  
Angle of attack of main wing

$\alpha_{H_{\text{main-wing-stalled}}}$  
Angle of attack of horizontal stabiliser when main wing is stalled

$\alpha_{\text{installed}}$  
Installed angle of main wing, relative to longitudinal axis of aircraft

$\alpha_{\text{installed}}_{H}$  
Installed angle of horizontal stabiliser, relative to longitudinal axis of aircraft

$\alpha_{\text{installed}}_{V}$  
Installed angle of vertical stabiliser, relative to longitudinal axis of aircraft

$\alpha_{\text{loa}}$  
Angle of attack of main wing when zero lift produced at wing root

$\Delta \alpha_{OL}$  
Reduction in zero-lift angle of attack of main wing due to flap deflection

$\alpha_{\text{stall}}$  
Stall angle of attack of main wing

$\beta$  
Field of View

$\gamma$  
Climb angle

$\varepsilon$  
Twist angle of main wing

$\zeta_{\text{main}}$  
Angle between main gear axle line and vehicle centre of gravity (tail dragger undercarriage)

$\lambda$  
Taper Ratio

$\lambda_{H}$  
Taper ratio of horizontal stabiliser

$\lambda_{V}$  
Taper ratio of vertical stabiliser

$\mu$  
Friction Coefficient, magnetic permeability relative to copper

$\eta_{\text{motor}}$  
Motor efficiency

$\eta_p$  
Propeller efficiency

$\eta_s$  
Horizontal stabiliser efficiency

$\eta_{\text{total}}$  
Total propulsion system efficiency
\[ \theta_{\text{servo}} \] Deflection of actuator servo

\[ \theta_{\text{servo}}^{\text{max}} \] Maximum deflection of actuator servo

\[ \theta_{\text{surface}} \] Deflection of control surface

\[ \theta_{\text{surface}}^{\text{max}} \] Maximum deflection of control surface

\[ \rho \] Freestream air Density

\[ \sigma \] Normal stress

\[ \tau \] Shear stress

\[ \tau_{\text{max}} \] Maximum shear stress

\[ \tau_{s} \] Shear stress due to shear force

\[ \tau_{\text{servo}} \] Torque requirement of actuator servo

\[ \tau_{\text{servo}}^{\text{design}} \] Maximum design torque of actuator servo

\[ \tau_{\text{servo}}^{\text{max}} \] Maximum torque requirement of actuator servo

\[ \tau_{\text{surface}} \] Moment acting on control surface

\[ \tau_{\text{surface}}^{\text{max}} \] Maximum moment acting on control surface

\[ \omega_{\text{motor}} \] Motor speed

\[ \omega_{\text{motor}}^{\text{max}} \] Maximum motor speed

Coordinate Frame Definitions

The coordinate frame used throughout this report is shown in the Figure 1 below.
Figure 1: Coordinate Frame (Mahoney, 2007)
Chapter 1

Introduction

The development of Unmanned Aerial Vehicles (UAVs) is one of the fastest growing areas of the international aerospace industry. The development of Australian designed and manufactured UAVs is an ideal product in which Australian aerospace engineers can demonstrate their technical knowledge to the world.

To date, Unmanned Aerial Vehicles have been used for defence and security operations, vehicles such as the Aerovironment ‘Puma’ being used for battlefield reconnaissance, and the General Atomics ‘Globalhawk’ has being considered by the Australian Government for border security. Less publicised are the potential uses of UAVs for domestic, civil applications. UAVs have the potential to perform a large number of civil tasks, with a high level of efficiency and cost effectiveness, in a manner which reduces work load and risk to the persons involved with the task. Employing suitable payloads, possible civil applications for a UAV include; search and rescue missions, traffic monitoring, surveillance (such as shark patrol), bushfire early detection and monitoring, mapping, topography, surveying, planning and meteorological data acquisition.

The increased use of UAVs in civilian applications will depend on the development of a greater number of suitable airframes and control systems and a increased awareness of UAV capabilities. The primary goal of the Search and Rescue UAV project being conducted at the University of Adelaide is to progress both of these areas.

1.1 Aim

The aim of this project is to develop an autonomous UAV with ground based surveillance capabilities. This UAV is designed primarily for the purpose of entrance into the UAV Outback Challenge Search and Rescue Division, and hence named iSOAR, that is; an aircraft capable of Intelligent Surveillance for Outback Aerial Rescue. It is intended however, that the platform also be suitable for other civil surveillance applications.

The UAV iSOAR has been designed, developed and manufactured using existing techniques and readily available materials and components. The project included the design of the airframe, as well the control, navigation, and imaging systems that were integrated into the vehicle allowing
the objectives of the design to be met. Once manufactured, remote control equipment was to be installed and testing was to be carried out to ensure that a stable airframe had been created. Control, navigation, and imaging systems will be integrated onto this custom airframe, primarily using commercially available equipment. This equipment will be configured to allow autonomous control of iSOAR, and the streaming of a live image feed from iSOAR to a custom designed ground station. The requirements of these on-board components were determined utilising the requirements of the UAV Outback Challenge. Certification in line with the regulations of the Australian Civil Aviation Safety Authority (CASA) were required for the vehicle. To attain this certification, extensive structural and flight testing of the vehicle was to be performed. The structural testing will involved both engineering analysis and non-destructive testing to ensure integrity. Manual, radio control (RC) of the vehicle was to establish the stability of the airframe for flight, and such flight testing, in conjunction with autonomous flight testing was to be conducted to meet the requirements of CASA certification. The design would be validated through participation in the inaugural Australian Research Centre for Aerospace Automation (ARCAA) UAV Outback Challenge.

1.2 Project Objectives

The project objectives were specified by the project group very early in the project. These consisted of both primary and extended project goals. A review of the goals, and the level to which they were achieved is presented in the conclusion of this report.

Primary project objectives

1. Development of a UAV platform capable of being flown via remote control, and which has the potential for being configured for autonomous flight and civil surveillance purposes.

2. Fitment of surveillance equipment to UAV platform capable of capturing imagery observed by the UAV during flight.

3. Development of a deployable payload system for the UAV, capable of carrying approximate 600g, and delivering it to a specified target.

4. The integration of an auto-pilot system, which allows the vehicle to maintain straight, horizontal flight autonomously.

Extended project objectives

1. Encourage continued undergraduate and postgraduate development of UAVs at the University of Adelaide.

2. Automation of the UAV platform for all flight regimes through the implementation and configuration of various UAV auto-pilot componentry externally sourced.
3. Development of a surveillance system for a UAV which can stream to a ground based station and allow for autonomous search and identification of ground based targets.


5. World recognition for aeronautical research at the University of Adelaide.

**ARCAA UAV Outback Challenge**

The ARCAA UAV Outback Challenge Search and Rescue is a competition run by the Australian Research Centre for Aerospace Automation in conjunction with the Queensland University of Technology. The intention of this competition is to raise awareness of the potential civilian applications of UAVs. This competition requires the surveillance of a specified area with the intent of locating a missing person and deploying a small rescue package to that person. Points are given for the level of automation which could be achieved. The design of iSOAR aims to, ultimately, produce a craft capable of completely autonomous flight for ground-based surveillance. In addition to testing the design of the vehicle, the participation of the aircraft in this competition will contribute to the exposure of UAV capabilities to the World and ultimately the expansion of civil UAV usage throughout the World.

**1.3 Scope**

This report details the first prototype design, development and manufacture of iSOAR. Details regarding ground and flight testing are presented along with a detailed discussion of the results of the project.
Chapter 2

Literature Review and Feasibility Study

This phase of the project focused on a review of current literature related to Unmanned Aerial Vehicle technology. A review of aircraft design texts that were utilized during the design was carried out, along with a review of current control and imaging systems technology, capabilities, and appropriate literature. The process then moved onto the review of commercial UAVs for a detailed statistical analysis that was compiled for the purposes of design resulting in the development of a database of UAV statistics. These statistics were used to determine the feasibility of this project and finally a study was conducted to develop the design requirements for this aircraft.

2.1 Literature Review

The following literature review is a brief summary of the extensive investigation into Unmanned Aerial Vehicle technology that was conducted at the beginning of the project. This section has been divided into literature on aircraft design, control systems and typical payloads.

Review of Aircraft design literature

There are a large range of aircraft design texts and guides available for assisting the design process. A discussion about the principle design text, Daniel Ramyer’s ‘Aircraft Design: A Conceptual Approach’ is presented below, followed by a short discussion of other useful texts and their contribution to the design of this aircraft.

Raymer’s Aircraft Design: A Conceptual Approach (Raymer, 2006) is an aircraft design text book that has received world wide recognition. The process discussed in this text starts from a conceptual point of view and focuses on the development of a feasible design concept. From this point the text uses a classical analysis approach for each of the major aircraft design disciplines such as aerodynamics, performance, structures and propulsion. The detail presented in this text gives the general method of analysis and the important points to focus on, however additional text were required to substantiate the analysis.
2.1. LITERATURE REVIEW

Roskam’s Aircraft Design Series (Roskam, 2004b) are aircraft design texts for Aeronautical Engineering II and tend to approach aircraft design from a ‘cookbook’ perspective. This was seen as less favourable for this project, however this series was used for a variety of technical references on specific topics.

Niu’s Aircraft Structural Design (Niu, 1992) texts were used for additional structural design information. These texts consisted of books on stress analysis, composite structures, and airframe structural design, all which were applicable to the development of an airframe.

Abbott’s Theory of Wing Sections (Abbot & VonDoenhoff, 1959) was used extensively throughout the aerodynamic analysis, as both a guide and a source of detailed technical information. Despite its obvious age this text is still used in commercial design applications.

A variety of other references and texts were used extensively throughout this project however a detailed analysis of each of these is not necessary. Relevant information for individual sections will be presented as required.

Control Systems

The development of control systems for UAVs has been in progress for many decades, beginning with the simplest missile guidance applications and advancing to completely autonomous control. A literature review of systems allowing for the automation of in-flight control was conducted, highlighting both the capabilities of currently available commercial systems, as well as the capabilities of UAVs in defence and the research that is currently in progress. In brief, this showed that the in-flight automatic control that was considered for the UAV in this report was well within the current capabilities of the UAV market in general.

Civil Applications of UAVs

A literature review of current commercially available autopilot systems was conducted in order to gain an understanding of the range of technologies available. This highlighted a large number of available self contained systems that allowed for the complete automation of in-flight control, with some also providing autonomous take-off and landing capabilities. These systems rang in price from $US5000 to $US50000, and, of the systems investigated in depth, none allowed for hardware modification or provided the user with the embedded code used.

A small number of open-source autopilots that were being developed at the commencement of the project were identified. These systems were all initially targeting the level of automation required for this design, and hence were investigated for use in this project. It was noted that these systems were being developed by hobbyists and at the commencement of this project it was not clear whether any of these options had successfully been integrated into an airframe. Review of these systems continued throughout the duration of the project and it was found that some of these systems made significant development progress during 2007. The most noteworthy open control autopilot system, the Paparazzi, has had various successes with implementation into Micro Air Vehicles (MAVs) and small UAVs (Paparazzi, 2007).
Defence Applications of UAVs

The largest user of UAVs at present are defence departments, in particular those in the USA. To give some perspective, the National Defence Authorization Act in 2001 stated a target that, within ten years, one-third of U.S. military operational deep strike aircraft will be unmanned (Bone & Bolkom, 2003). In terms of control, the capabilities of the defence department cover the entire range of currently available technology. Although the most well known defence UAVs such as the Predator and Global Hawk are considerably larger than the airframe detailed in this report, smaller designs such as the BUSTER, Silver Fox and Scan Eagle are also used which are of sizes comparable to iSOAR. The control systems of these UAVs are capable of fully autonomous flight with various methods of autonomous take-off and landing. Additional features on the different UAVs include the ability to fly to the cursor position on the ground station and a flight endurance of up to 30hrs (OSD, 2005).

Current research

UAV control systems is an area that is currently under heavy research both in commercial, defence and educational institutions. Indeed, all of the 'Group of Eight' universities in Australia currently have some research or projects based on UAV systems and their control. The literature survey indicated that dominant areas of study at present are the control and co-ordination of multiple UAVs, autonomous collision avoidance and adaptive control.

Imaging System

2.2 Market Evaluation and Benchmarking

An extensive market evaluation of operating UAVs from around the world was conducted. This market evaluation was conducted in parallel with the aforementioned literature review, and provided extensive design knowledge to the project. By recording the statistics of those vehicles that possess similar payloads and payload capacities to the proposed aircraft a statistical database was generated providing valuable benchmark figures from which design work could be compared.

In addition to the extensive statistical database generated from the market evaluation, four UAVs were selected to provide detailed benchmarks to base the design on. The selection of these prototype vehicles was based on the following criteria:

- Similarity of mission requirements and application
- Similar weight to the expected final design
- Success in industry
- Availability of knowledge
The primary selection criterion for the prototypes was vehicle weight and application. From the results of the weight estimation, the prototypes selected had to be small UAVs with a take off weight of up to 30 kg. Using this criterion in conjunction with the other previously mentioned criteria, four prototypes were selected for benchmarking. These four aircraft are:

**Dutch Space ‘Mate’** is a close-range reconnaissance UAV capable of missions with up to 60 minutes duration (Dutch-Space, ND). The Mate is of a conventional configuration, as can be seen in Figure 2.1, and the vehicle features a single forward facing camera in the lower fuselage that streams video back to a ground station. The maximum takeoff weight of the vehicle is 6kg, which is similar to that expected for the project (Dutch-Space, ND). For these reasons the Mate was deemed a suitable prototype for the project.

![Figure 2.1: The Dutch Space 'Mate' (Dutch-Space, ND)](image)

**Aerosonde ‘Aerosonde’** is a successful UAV design that has been utilized throughout the world for missions including surveillance and meteorological investigations (Aerosonde, 2007). An Australian design, the vehicle is currently in operation with the Defence Science and Technology Organization (DSTO) for the ‘Automation of the Battlespace Initiative’ currently being developed (Aerosonde, 2007). Although the Aerosonde is not a conventional configuration, as seen in Figure 2.2, the success of the vehicle makes it a prime source of information with regards to design parameters for a successful UAV system. Furthermore, the maximum takeoff weight of the vehicle is around 14kg (Aerosonde, 2007) which is similar to that expected for the project aircraft. Additionally, a large amount of knowledge regarding the Aerosonde system and its success is readily available. Hence the Aerosonde was selected as a suitable prototype for the project.

![Figure 2.2: The Aerosonde UAV](image)
Instituto Nacional De Tecnica Aeroespacial (INTA) 'Alo' is a surveillance UAV capable of real-time video streaming to a ground station. The vehicle is launched by catapult and is recovered by a parachute landing system. The Spanish army has used this aircraft but it is also intended for civil purposes including the detection and early warning of fires (Instituto-Nacional-De-Tecnica-Aeroespacial, ND). In addition to this the Alo is also of a conventional configuration as can be seen in Figure 2.3 and thus was deemed to be a suitable prototype for comparison.

Faraz Asia Technologies 'Faraz II' is a hand-launched surveillance UAV developed in Iran. The vehicle is of a conventional configuration, and is capable of autonomous flight (Arjomandi, 2007). Being hand launched the vehicle is not of an ideal size. However, features such as the simple stowage and assembly methods of the vehicle make it an attractive design for further investigation (Arjomandi, 2007). Furthermore, knowledge of the performance of the vehicle was also readily available to the group courtesy of the project supervisor whom worked on the vehicle. These reasons made the Faraz II a suitable prototype to benchmark. The vehicle is shown in Figure 2.4.
2.3 Statistical Analysis

A statistical analysis method was utilised by the project group to design the vehicle. This method was as suggested by both Raymer (2006) and Roskam (2004b). The statistical analysis method involves investigating the performance and designs of existing vehicles and using the information from these designs to construct a baseline design of the vehicle, which can then be optimized for the required technical task. The statistics used for this method were those collected as a result of the market evaluation.

2.3.1 Analysed UAVs

A number of commercial UAV platforms were analysed for their design parameters. These included aircraft up to and including 30 kg in take off weight, aircraft with both electric and hydrocarbon propulsion systems and with an endurance of at least 30 minutes.

2.3.2 Parameters to be Analysed

A number of critical design parameters were analysed on appropriate aircraft. Easier to obtain parameters were gathered for all aircraft if available, which included:

- Maximum take off weight (MTOW)
- Payload weight
- Empty weight
- Engine type
- Maximum speed / cruise speed
• Endurance

• Geometry
  – Configuration
  – Span
  – Length

• Launch / recovery method

• Payload type

However a number of specific design parameters were collected on a small sample of UAV platforms, such as the design prototypes. These specific parameters included.

• Wing loading

• Power loading

• Wing area

• Tail volume coefficient

• Airfoil cross section

This statistical data was utilised extensively throughout the design phase.

2.4 ARCAA UAV Outback Challenge

The ARCAA UAV Outback Challenge (ARCAA, 2007), held in late September, 2007, was a chance to prove the UAV system developed for this project. For this mission, an unmanned system was required to search a predetermined area, locate a lost bush walker and deploy a small rescue package, in the form of a 500ml water bottle, to them and return home in one hour. A schematic of the required search area is included in Figure 2.5. A brief detailing of the competition requirements follows.
2.4. ARCAA UAV OUTBACK CHALLENGE

Mission boundary is defined as an area which is 2 nm by 3 nm and at a height above ground level between 200ft and 1500ft. A flight corridor of approximately 1 nm in length in which aircraft are to fly to the mission boundary from the competition arena is also included within the mission boundary (Figure 2.5).

Search area is an area of 1 nm by 2 nm defined inside the mission boundary where the 'lost bushwalker' is located (Figure 2.5).

Time limitations are imposed on the competition. A time of 45 minutes will receive maximum score, whilst a time of over 1 hour and 15 minutes will results in the loss of all competition points.

Automation was a condition of entry, such that any vehicle entered into the challenge must have some level of automation. The development of flight control systems were deemed feasible as per the control system survey discussed in Section 2.1. The level of automation achieved in this project was somewhat limited by financial constraints, and the availability of proprietary flight control systems. Individual points were awarded based on the level of automation present in flight regimes such as:

- Take off
- Search and identification of the target
- Deployment of the payload to the target
- Landing

Although automation was a requirement of the challenge by no means was a fully autonomous vehicle required for entry, and thus manual control input could be used for certain aspects of the mission.
2.5 Mission Profile Specification

The development of design specifications stemmed from the requirements of the aircraft’s mission profile. Thus a number of civil UAV applications were analysed in order to determine specifications that would provide for a flexible and robust system, suitable for multiple applications. The analysed missions for this system are covered under two general headings; search and rescue, and surveillance.

Search and rescue missions are characterised by the ability to deliver a payload to a ground target, whether it be a shark deterrent in an ocean based mission, or a bottle of water to a lost bushwalker. The typical mission considered consists of standard take-off and fly out regimes, followed by a search regime which culminates in the deployment of a package or loitering about a specific area of interest, a return phase to the base station then follows, employing landing or recovery manoeuvres. This particular mission profile is suitable for use in the ARCAA UAV Outback Challenge.

Civil surveillance applications have a similar mission profile to search and rescue missions, however the ability to deploy a payload is not required. A mission profile was determined for various civil surveillance missions such as:

- Shark patrol
- Bushfire surveillance
- Traffic monitoring
- Environmental monitoring
- Agricultural monitoring
- Wildlife surveillance

Slight differences exist in the performance and operational requirements for each of these mission. A brief analysis of surveillance applications has been conducted and is summarized in Table 2.1.
Table 2.1: Summary of mission profile analysis for several civil surveillance applications

<table>
<thead>
<tr>
<th>Application</th>
<th>Maximum cruise speed (km/h)</th>
<th>Operational cruise speed (km/h)</th>
<th>Minimum feasible endurance</th>
<th>Altitude of flight (ft)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Shark patrol</td>
<td>100</td>
<td>70-90</td>
<td>1 hour</td>
<td>400-700</td>
</tr>
<tr>
<td>Bushfire surveillance</td>
<td>150</td>
<td>70-90</td>
<td>1 hour (preferably more)</td>
<td>1000-2500</td>
</tr>
<tr>
<td>Traffic monitoring</td>
<td>100</td>
<td>50-70</td>
<td>1 hour</td>
<td>400-700</td>
</tr>
<tr>
<td>Environmental monitoring</td>
<td>80</td>
<td>50-70</td>
<td>30 min</td>
<td>400-700</td>
</tr>
<tr>
<td>Agricultural monitoring</td>
<td>80</td>
<td>50-65</td>
<td>30 - 45 min</td>
<td>200-500</td>
</tr>
<tr>
<td>Wildlife surveillance</td>
<td>75</td>
<td>40-60</td>
<td>30 min</td>
<td>200-400</td>
</tr>
</tbody>
</table>

Table 2.1 was developed using a critical analysis of the objectives of each of the missions and a reasonable range of performance parameters that would suit each application. By evaluating these parameters it was deduced that a design with an endurance of at least one hour would meet the minimum requirements of all of the aforementioned surveillance applications. Similarly, low altitude flight was also desired, along with a moderate to high cruise speed, in the vicinity of 70 to 90 km/h. In an effort to quantify the operational requirements of this system the UAV Outback Challenge requirements was used, which ensured that the final design met these objectives as well as the requirements of a number of typical civil applications. The selected mission profile is shown in Figure 2.6.

Figure 2.6: Mission Profile of a Typical Search and Rescue Mission (adapted from Brock, 2006)

2.6 System Requirements

The following section details the development of the design requirements for the UAV system developed. These requirements have been obtained with an aim to produce a UAV capable of operating in typical surveillance missions, as discussed in Section 2.5, while using the ARCAA Outback Challenge to quantify the exact parameters required.
2.6.1 Determination of Cruise Speed Requirement

A brief trade study is presented to estimate the required aircraft velocity. As the mission profile defines a search area and the required endurance a relationship between cruise velocity and camera requirements was generated. Understanding the limitations of imaging systems in the budget was essential when determining an appropriate cruise velocity. From this trade study the aircraft design parameters were specified.

An estimate of the cruise requirements was determined by restricting the time of search, as indicated by Table 2.2.

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Total time</strong></td>
<td>+60 minutes</td>
</tr>
<tr>
<td><strong>Set up time</strong></td>
<td>-5 minutes</td>
</tr>
<tr>
<td><strong>Loiter, confirmation and payload deployment</strong></td>
<td>-5 minutes</td>
</tr>
<tr>
<td><strong>Contingency</strong></td>
<td>-10 minutes</td>
</tr>
<tr>
<td><strong>Total time for search</strong></td>
<td>+40 minutes</td>
</tr>
</tbody>
</table>

Although this time allocation may seem unnecessarily ambitious it does allow for the completion of the entire searching phase within the points-maximizing time of 45 minutes.

As camera requirements alter depending on altitude and field of view, a parameter referred to as search width was defined as the effective search width covered during a single pass by the imaging system. Using a simple ladder search style, an approximation of the required search speed has been developed for a given search width. The result of this analysis is shown in Figure 2.7.

![Search Width Vs Cruise Speed](image)

Figure 2.7: Relationship between Search Width and Cruise Speed

The relationship provided in Figure 2.7 shows that a higher cruise speed is favourable, however should the cruise speed be increased excessively then it will become difficult to distinguish a person on the ground due to forward translation and the associated time the target will spend
A qualitative trade-off between search width and operational cruise speed was made in order to proceed with the design. Original estimates for search height were of the order of 150m in width, however through analysis of in-flight footage from other UAV systems this was deemed to be insufficient for this mission as the target was expected to be difficult to locate. After reanalyzing Figure 2.7 it was determined that a cruise speed of 90 km/h would be sufficient to meet both camera operational requirements and airframe development requirements.

### 2.6.2 Propulsion Concept Selection

Both petrol and electric power plant alternatives exist to provide propulsion to a small UAV. Propulsion for this UAV system will be via a propeller and driving mechanism as this will provide adequate thrust and efficiency over the speed range of the system. Hydrocarbon based internal combustion engines were one option to drive such a propeller. Electric motor and battery technology has had significant improvement both technically and economically over the past few years. Both of these power sources would be adequate in order to power an aircraft of approximately 10kg. A short background of each of these systems is detailed below, followed by power plant selection criteria, a direct comparison and the selection of the propulsion system type.

**Petrol Engine**

For an aircraft of this size and power requirements a suitable Internal Combustion Engine (ICE) was a glow plug engine. These engines are common amongst the model aircraft community and have been the principle propulsion system of choice for modellers. The glow plug engine differs from the conventional internal combustion engine found in automobiles. The fuel is ignited through means of a catalytic chemical reaction between the fuel and the filament of the glow plug, rather than with a spark plug. This allows for the system to be miniaturised however this results in inconsistent, unreliable performance and requires an additive of nitro-methane to the fuel to promote combustion. This increases the levels of pollution emitted into the air.

**Electric Motor**

Electric motors were an alternative to the glow plug engines. Recent developments in motor and, in particular, battery technology have allowed electric systems to match the performance of internal combustion engines in both power output and overall system weight. This has been achieved with the integration of electric motors with advanced lithium polymer (LiPo) batteries.

**Selection Criteria**

When selecting the propulsion system a number of issues were analysed to determine the most suitable propulsion system. These included:
CHAPTER 2. LITERATURE REVIEW AND FEASIBILITY STUDY

1. Development and integration time
2. System requirements and cost
3. Maintenance and reliability
4. Power to weight ratio
5. Complexity
6. Operational costs
7. Noise

Concept Selection

An electric power plant is more suitable for this UAV system. The obvious benefits for ease of development and integration, maintenance and reliability and reduced system complexity make this propulsion system the most ideal choice. In addition the electric engine is capable of providing the required endurance for the mission profiles. Selection of an electric propulsion system is detailed Section 3.8.

2.6.3 Launch and Recovery Requirements

The UAV design incorporated traditional landing gear and an alternative launch and recovery method. A brief discussion of the possible alternative launch and recovery methods will be presented followed by the selection of the proposed launch and recovery methods to be developed further.

Alternative Launch Methods

**Catapult or bungee systems** use high accelerations from a ground based energy source to launch a UAV. This method required extra ground based equipment and a significant development time along with testing to ensure sound operation. Due to the high load factors experienced by the airframe during launch, the airframe may require some specific structural design for launch conditions. The launching method is also inherently risky.

**Car launch** is used to launch vehicles from the roof top of a vehicle. Often modifications to the pre-existing roof rack system can be made to achieve this. This method involves the design and development of some launch specific ground based components but required only minor structural modifications to an existing airframe, due to the reduced load factors as compared with the catapult method.
Launch trolley consists of a ground based trolley which is used as wheels during takeoff. The trolley does not leave the ground and therefore the aircraft does not require external landing gear during flight. This method is simple concept however serious difficulties may arise during integration and operation.

Alternative recovery methods

Parachute recovery involves the deployment of a parachute in order to reduce kinetic energy and descend the aircraft to the ground. The integration of a pilot chute and deployment system can increase the reliability of this system as it may double as an emergency recovery system. Significant aerodynamic loads can be generated from a parachute system if it is not designed correctly. A parachute recovery system is also very easy to automate using GPS waypoint navigation and flight control systems.

Net or wire recovery is implemented by flying the aircraft directly into a ground based net or wire system which 'catches' the aircraft. The automation of this system required a great deal of technical knowledge and commonly incorporates differential GPS to increase accuracy during recovery. This system also requires a complicated ground based infrastructure.

Belly landing involves the controlled flight into terrain and does not require any additional infrastructure however this would place abrasive and impact shock loads on the airframe, thus requiring some structural considerations during design.

Alternative launch and recovery selection

A car launch system was selected as the alternative method of launch as this was perceived to be the easiest to integrate with the airframe, required the least development time and lowest operational risk. The successful development of this system would allow for a relatively simple automatic launch method, and thereby increase the level of automation achieved.

The alternative recovery method was a parachute system. This method was determined to be the easiest to integrate, automate and to out source due to the modular design of parachute systems. The parachute mechanism doubles as an emergency recovery method. This allows the aircraft to be recovered successfully in the event of a failure during flight, such as loss of power or control authority. This was another key characteristic of the parachute system that led towards its integration into the design.

The successful development of these alternative launch and recovery methods increased the level of autonomy and operational flexibility of the UAV system.

2.6.4 Automation Requirements

An important specification, in terms of control systems development, is the level of autonomy required. These specifications have an impact on both the hardware and development time
required. Although the car launch and parachute recovery systems were included in the design to increase the level of automation, for the duration of this project the flight path, that is aircraft control and navigation was to be automated, however a manual system would be used for take off and landing. Automation of the payload drop for the Outback Challenge was also excluded due to the complexities involved in the imaging system to achieve this. A brief discussion of the three main flight phases with regard to control is presented below.

**Takeoff/Landing**

Although many of the commercial autopilots claimed the system was capable of autonomous take-off and landing additional hardware was required, namely ultrasonic altitude sensors, to enable this capability. This was due to the requirement during these phases (in particular landing) for the control system to accurately measure altitude, to within centimeters, which cannot be provided using either pressure transducers or GPS altitude due to accuracy limitations. Thus, to decrease the cost of the control system and reduce the development time automated take off and landing were discounted in terms of control. However, it is noted that the development of the alternative car launch and parachute recovery systems would allow for the automation of these phases without the need for the additional hardware as the less accurate altitude measurements would be sufficient.

**Pre-defined Flight Path and Flight Path Modifications**

The ability of the autopilot to fly between a set of waypoints which are pre-defined before flight is considered the basic necessity of a true autopilot system. For the UAV Outback Challenge and other civil applications, this needs to include the ability to change altitudes, as well as allow for in-flight modifications to the flight path.

**Payload Drop**

Automation of the payload drop would rely predominantly on the ability to accurately identify the location of the target using GPS. With an accurate location, sensory knowledge of the position and orientation of the aircraft as well as the wind conditions would allow automation of the payload drop, provided testing had provided knowledge of the behavior of the payload during descent. However, a requirement for conferral with the on-site judges prior to the payload drop limited the degree of automation which could be used during the payload drop phase, and the accuracy to which GPS co-ordinates would be available was unknown. Thus a manual overlay on the imaging system similar to a gun sight combined with a manually controlled payload release, rather than automation of this process was utilised.

2.6.5 **Imagery Requirements**

To fulfill the UAV Outback Challenge goals to the highest level, a completely autonomous target identification process was required. This was deemed to be beyond the scope of the skills of the
2.7. SUMMARY OF DESIGN REQUIREMENTS

A brief specification of the design requirements is stated below.

Performance Requirements

Range of the aircraft is limited by the quality of the communications equipment which was specified to a 10km radius.
Endurance is the duration that the aircraft is able to remain airborne. This requirement was set to in excess of one hour to increase the practicality of the system and extend the surveillance capabilities.

Cruise velocity is closely linked to camera performance and search area requirements, it was set to 90 km/h. This high velocity made the design suitable for a number of alternative time critical surveillance operations such as traffic surveillance and shark patrol.

Maximum operational speed was designated to be 120 km/h to increase the dash out and dash in capabilities of the system. Increasing this parameter exponentially complicates the design of the power plant and structural members.

Take off and landing field length is the length of the field required for take off and landing. For this aircraft the maximum take off field length was 200 ft, which was deemed to be short enough to maintain application flexibility and long enough to reduce power requirements.

Climb performance is set by the required climb gradient. The CASA regulation for UA 25.65 for climb under all motors operating states that the aircraft must maintain a climb gradient of at least 8.33% at a speed greater than 1.3Vs with the maximum take off weight.

Operational altitude is governed by the camera performance and mission requirements. After a brief analysis this was expected to range between 200 ft to 600 ft depending on camera specifications and application requirements. Remote control and autopilot testing was to be conducted at altitudes below 400 ft to meet CASA requirements for unmanned aerial vehicles. The service ceiling is set by CASA to be 1500 ft for small UAVs. It was expected that the designed UAV would be physically capable of flight well above this altitude.

Level of flight autonomy required affects the complexity of the control system required as well as the integration time. As such, the autonomy requirement has been specified as control over a complete pre-defined flight path, excluding landing, take-off, and payload deployment.

In flight monitoring capabilities are to be present when under autopilot control, and able to report the position and orientation of the UAV at all times, along with any in-flight warnings, to a base station up to 10 km away.

Imaging Systems were to be constructed to give the opportunity to identify a person in an outback setting from a height of 200-600 ft, and provide this imagery to a base station up to 10 km away.
2.7. SUMMARY OF DESIGN REQUIREMENTS

Operational Considerations

**Electric motor** is to be utilised to minimise development time, easy integration with airframe and electronics and provide a clean system. This also results in unrestricted testing and a low noise signature.

**Size** of the UAV is not limited however it would be ideal for the system to be compactable into a station wagon for transportation. This would make the system flexible in its applications as the platform could easily be transported.

**Mass** of the airframe must be below the CASA restriction of 150 kg to meet the requirements of a small UAV. It was envisaged that the mass of the final system would be less than 20 kg, based on a predicted payload of approximately 2kg. This consists of up to 1.2 kg of imaging and control systems, 0.6 kg for the deployable payload and 0.2 kg for fixtures and fittings.

**Payload modularity** was required to retain the flexibility of this system. An externally mounted deployable payload allows payload modularity.

**Maintenance and access** was required for all onboard electronics, actuators, motor and power sources.

**Alternative launch and recovery systems** were investigated as previously mentioned. A car launch system and parachute recovery system are discussed in more detail as these were the most feasible options to increase the operational flexibility and autonomy of this system.
Chapter 3

Platform Conceptual Design

The conceptual design aimed to select and develop a feasible design concept that met all design requirements. This process was conducted using a classical conceptual design approach and broadly followed the process presented in Raymer (2006). This process set about determining the major design parameters and decisions that drove the resultant design and thus performance of the final system.

3.1 Configuration Design

The UAV’s configuration determines the process through which it can be sized, hence the configuration was the first design parameter which needed to be addressed. The feasibility study indicated that there were five general configurations that should be considered. These five configurations were conventional, pod and boom, twin tail, flying wing and canard, as shown with conceptual sketches in Figure 3.1. Each of these has specific design and performance characteristics that produce advantages and disadvantages. The merits of each design were considered in a qualitative manner to select the most appropriate configuration for this design.
3.1. CONFIGURATION DESIGN

To select the configuration most suitable for this UAV aspects of the design were grouped into ten categories, each of which may have a number of sub categories. These categories were then ranked from 1 to 10, 10 being of the greatest importance, this ranking scheme was then converted to a percentage based scheme with the following results:

- Vehicle controllability (18.2%)
- Vehicle stability (16.4%)
- Vehicle weight (14.5%)
- Aerodynamic efficiency (12.7%)
- Vehicle maneuverability (10.9%)
- Flight regime performance (9.1%)
• Configuration knowledge (7.3%)
• Ease of manufacturability (5.5%)
• Project profile (3.6%)
• Appearance (1.8%)

More desirable characteristics such as controllability, stability and vehicle weight were weighted more heavily, while aspects that were less important, such as project profile and appearance were weighted lightly. These weightings reflect the requirements of the design.

Each configuration was given a score for all design aspects that was determined by applying the relevant weighting values to each category score and then summing the results as can be seen in Table 3.1. The results of this analysis indicated a conventional design was the most suitable configuration for this application. This is primarily due to the ease of control, inherent stability, simple design and associated short development time.

Table 3.1: Detailed Decision Matrix displaying the Analysis of the Configuration Design

<table>
<thead>
<tr>
<th>Aspect</th>
<th>Weighting (Out of 55)</th>
<th>Possible Weighted Score</th>
<th>Conventional</th>
<th>Pod &amp; Boom</th>
<th>Twin Tail</th>
<th>Flying Wing</th>
<th>Canard</th>
</tr>
</thead>
<tbody>
<tr>
<td>Controllability</td>
<td>10</td>
<td>18.20</td>
<td>17.59</td>
<td>17.59</td>
<td>15.17</td>
<td>7.28</td>
<td>10.31</td>
</tr>
<tr>
<td>Stability</td>
<td>9</td>
<td>16.40</td>
<td>16.40</td>
<td>16.40</td>
<td>16.40</td>
<td>3.28</td>
<td>0</td>
</tr>
<tr>
<td>Weight</td>
<td>8</td>
<td>14.50</td>
<td>5.80</td>
<td>5.80</td>
<td>2.90</td>
<td>14.50</td>
<td>11.60</td>
</tr>
<tr>
<td>Efficiency</td>
<td>7</td>
<td>12.70</td>
<td>6.89</td>
<td>6.35</td>
<td>2.72</td>
<td>12.16</td>
<td>6.89</td>
</tr>
<tr>
<td>Maneuverability</td>
<td>6</td>
<td>10.90</td>
<td>4.36</td>
<td>4.36</td>
<td>0.00</td>
<td>8.72</td>
<td>10.90</td>
</tr>
<tr>
<td>Flight Regime</td>
<td>5</td>
<td>9.10</td>
<td>8.65</td>
<td>6.83</td>
<td>5.46</td>
<td>3.19</td>
<td>5.01</td>
</tr>
<tr>
<td>Configuration Knowledge</td>
<td>4</td>
<td>7.30</td>
<td>7.30</td>
<td>7.30</td>
<td>3.65</td>
<td>2.19</td>
<td>2.92</td>
</tr>
<tr>
<td>Manufacturability</td>
<td>3</td>
<td>5.50</td>
<td>3.30</td>
<td>2.93</td>
<td>1.83</td>
<td>4.77</td>
<td>1.83</td>
</tr>
<tr>
<td>Project Profile</td>
<td>2</td>
<td>3.60</td>
<td>2.16</td>
<td>1.44</td>
<td>1.44</td>
<td>3.60</td>
<td>2.88</td>
</tr>
<tr>
<td>Appearance</td>
<td>1</td>
<td>1.80</td>
<td>1.08</td>
<td>0</td>
<td>0.72</td>
<td>1.80</td>
<td>1.44</td>
</tr>
<tr>
<td>Total (Out of 100)</td>
<td>-</td>
<td>100.00</td>
<td>73.53</td>
<td>69.00</td>
<td>50.29</td>
<td>61.48</td>
<td>53.79</td>
</tr>
</tbody>
</table>

3.2 Initial Concept Designs

A loosely governed idea generation process was used to develop design concepts. Each design was generated to satisfy the technical task, without technical limitations, including the positions of the payload. A critical analysis of five design concepts using lists of advantages and disadvantages of each design was performed. The advantageous aspects of each design were noted and amalgamated to form the final design concept.

The idea generation process produced several consistent trends, the most profound being the high mounted wing. The high mounted wing increases the stability of the aircraft about the
3.3. WEIGHT ESTIMATION

longitudinal axis, whilst allowing the internal volume of the fuselage to be used more efficiently than a centrally mounted wing structure. A vertical tail for simpler control dynamics was also common amongst the designs, along with a tractor motor / propeller configuration. This forward mounted motor configuration enables both the parachute and payload to be deployed without the risk of entanglement with the propeller. Other factors that were considered in the analysis of the design concepts were the internal volume of the fuselage for carrying flight control systems, the location of the payload and its effect on the stability of the aircraft when deployed.

The final design concept is shown in Figure 3.2. This concept is a combination of the desirable features of the ideas generated. It has a mid to high mounted wing, tractor propeller, high mounted parachute bay, modular payload bay located at the centre of gravity and the ability to use modular wings.

![Figure 3.2: Final Configuration Design Conceptual Sketch](image)

3.3 Weight Estimation

Successful design of an aircraft requires the knowledge of the weight of the vehicle, including the empty weight, and the design takeoff weight. These weight values were required during the preliminary stages of design to determine the performance and dimensions of the aircraft, however these values are difficult to predict without some form of formal analysis. The method suggested by Roskam (2004b) and Raymer (2006) is developed from the statistical analysis. This analysis reveals a relationship between vehicle takeoff and empty weights, based on the statistics of similar aircraft that have already been designed and which may or may not be in active service. The foundation of such analysis is technology limitations, with the premise being that most designs are limited by current and existing technology, and thus must satisfy some form of relationship dictated by these limitations (Roskam, 2004b). This graph is known as a ‘Technology Diagram’.
and it was used to find the required design takeoff and empty weights based on specifications from the project specifications.

**Weight Build-up**

To calculate the design takeoff and empty weights of the project vehicle, the above technique was employed, with a technology diagram constructed from weight values for UAVs of similar size. Details of the methodology undertaken for this process are discussed in detail below. The use of the statistical analysis approach for weight estimation required an estimation for the difference between the takeoff and empty weights of the plane. Takeoff weight, $W_{TO}$, is defined by Equation 3.1, for an electric powered aircraft.

$$W_{TO} = W_{payload} + W_e$$  \hspace{1cm} (3.1)

Equation 3.1, when used in conjunction with the statistical analysis, was used to estimate takeoff and empty weights.

**Payload Requirements**

As defined by the feasibility study, the UAV was to be configurable for a number of civil purposes, primarily centered on surveillance type missions. Consequently the payload is to include a camera system, with video streaming equipment. Furthermore, as stipulated by the rules of the ARCAA 2007 UAV Outback Challenge, the aircraft was required to feature a deployable payload system capable of supplying 500mL of water to a lost person. As a result, the payload requirements of the vehicle also included such a deployment system. As stated in the feasibility study, the total payload requirement was specified as a mass of 2 kg, which consisted of approximately 1.2 kg of imaging equipment and independent power, 0.6 kg of deployable payload and 0.2 kg of mountings and fittings.

**Statistical Analysis**

Having determined the required payload weight to be carried by the UAV, suitable UAVs were selected to construct a technology diagram. The project vehicle had to be classified based on propulsion system and payload requirements to successfully identify these UAVs.

The feasibility study specified the vehicle to be powered by an electric propulsion system. Thus the UAVs used for this analysis all needed to be electrically powered.

Based on investigation into the payload capacity of many UAVs, it was found that small UAVs with takeoff weights between 5kg and 30kg were capable of carrying 2kg of payload. Consequently, the project vehicle was classified as a small UAV with a design takeoff weight between 5kg and 30kg.

Eight existing UAVs, which fitted the classification of the project vehicle, were identified from a market investigation. The empty and takeoff weights of these vehicles were researched, so as to
form the basis for the technology diagram. The vehicles discovered, and the corresponding data for each can be found in Table 3.2.

### Table 3.2: Weight Data for Electric UAVs with Takeoff Weight between 5 kg and 30 kg

<table>
<thead>
<tr>
<th>UAV</th>
<th>Manufacturer</th>
<th>Empty Weight (kg)</th>
<th>Payload Capacity (kg)</th>
<th>Max Takeoff Weight (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sender</td>
<td>NRL</td>
<td>3.41</td>
<td>1.13</td>
<td>4.54</td>
</tr>
<tr>
<td>Duigan</td>
<td>Monash University</td>
<td>5.50</td>
<td>1.00</td>
<td>6.50</td>
</tr>
<tr>
<td>Biodrone</td>
<td>Alcore</td>
<td>7.00</td>
<td>3.00</td>
<td>10.00</td>
</tr>
<tr>
<td>Azimut</td>
<td>Alcore</td>
<td>4.50</td>
<td>2.00</td>
<td>6.50</td>
</tr>
<tr>
<td>Pointer</td>
<td>AeroVironment</td>
<td>3.44</td>
<td>0.91</td>
<td>4.35</td>
</tr>
<tr>
<td>Extender</td>
<td>NRL</td>
<td>10.90</td>
<td>3.20</td>
<td>14.10</td>
</tr>
<tr>
<td>Finder</td>
<td>NRL</td>
<td>20.70</td>
<td>6.10</td>
<td>26.80</td>
</tr>
<tr>
<td>Swallow</td>
<td>NRL</td>
<td>23.56</td>
<td>4.54</td>
<td>28.10</td>
</tr>
</tbody>
</table>

**Technology Diagram**

As previously mentioned, the technology diagram is a plot of takeoff weight versus empty weight, generated from the results of a statistical collection. To find a suitable relationship between the two parameters, Roskam (2004a) suggests the fitment of a linear curve between the logarithms of the two parameters. This method was adopted to develop a relationship between these two parameters for the UAVs selected to generate a suitable technology diagram.

The technology diagram produced from the statistics, in Table 3.2, has been represented graphically in Figure 3.3. This graph shows a highly linear relationship between the logarithms of empty weight and takeoff weight. The equation of the line of best fit to the statistical data was determined and the relationship between empty weight and design takeoff weight is defined by Equation 3.2.

\[ \log(W_e) = 1.0247\log(W_{TO}) - 0.1387 \]  

This equation was used to determine the design point, as seen in Figure 3.3, which specifies the design maximum take off weight, for the specified payload mass of 2 kg.

\[ W_{TO} = 9kg \]

This take off weight value was used during the design phase as a guide for the required loadings of the aircraft.
3.4 Design Specifications and Parameters

A brief review of the design specifications that were considered in this section is presented below.

**Maximum design take off weight** of the aircraft is 9 kg, as determined from the weight estimation procedure.

**Nominal cruise speed** was been defined as 90 km/h.

**Take off distance** was limited to 200ft, for the field length, however an analysis for a range of take off distances will also be presented.

**Ground level of operation** has the potential to reduce the take off performance. Due to competition requirements the sizing process was completed for operation of the UAV at a ground level of approximately 1500 ft.

**Climb requirements** are dictated by CASA regulations, and require a climb gradient in excess of 8.33% for a velocity of 1.3Vs. This factor included a safety margin of 40% in the following calculations to increase this parameter to a value 11.67%.
3.4. DESIGN SPECIFICATIONS AND PARAMETERS

Stall requirements were specified internally. The stall requirements were determined during the sizing process by understanding the relationship between stall speed, design cruise lift coefficient, as well as optimal design conditions. This depended on the airfoil selected for the wing, however preliminary estimates were determined from the analysis of a couple of preliminary airfoil selections, which are specified in more detail in Section 7.1.2. Through manipulation of this stall speed requirement, a semi-optimal aircraft design could be realised.

For straight level flight

\[ \frac{W}{S} = qC_L \]  
(3.3)

Equating wing loading for the cruise and stall cases the following relationship was derived:

\[ \nu_S = \sqrt{\frac{\nu_{cr}C_{L_{des}}}{C_{L_{max}}}} \]  
(3.4)

Understanding that aerodynamic efficiency is a function of the design lift coefficient, and that endurance is a function of aerodynamic efficiency. A design trade off between optimal aerodynamic efficiency at cruise conditions and low stall speed needed to be made.

\[ \frac{L}{D} = f(C_{L_{-des}}) \rightarrow \text{Endurance} = g \left( \frac{L}{D} \right) \]  
(3.5)

A critical analysis of Figure 3.4 was conducted. A stall speed of 60 km/h was selected as it was concluded that it was a reasonable trade off between cruise aerodynamic efficient and sufficiently low stall speed, to minimise operational risk.

(a) Preliminary drag polar for an aspect ratio of 8  
(b) Aerodynamic efficiency versus design lift coefficient

Figure 3.4: Preliminary Aerodynamic Plots

Preliminary Design Specifications

The preliminary specification of several other design variables such as the wing geometry and wetted surface area ratio was required in order to develop the design further. Each of these design decisions was made considering the type of aircraft, the required performance and its intended operation.
**Wing geometry**   The geometrical shape of the wing influences the aircraft’s performance. Parameters that define the wing’s geometry needed to be given consideration as each of them affect performance characteristics such as efficiency, structural weight, stability, stall characteristics and controllability.

**Aspect ratio, \( A \)**   is essentially the ‘fine-ness’ of the wing. The aspect ratio is an important parameter as it determines the lift distribution and performance of the wing. A high aspect ratio tends to have high aerodynamic efficiency, but high structural mass, whereas a low aspect ratio wing tends to have low structural mass, but low aerodynamic efficiency (Raymer, 2006). To achieve good structural and aerodynamic efficiency, a moderate to high aspect ratio of 8 was used for the design. A statistical analysis to determine the aspect ratio for this aircraft was not valid in this case as UAV manufacturers invest relatively large number of resources and time in developing low weight, high aspect ratio designs, which were considered difficult to replicate given the time and budget constraints of the project.

**Wing sweep**   is defined as the angle at which the wing centreline is sweep aft of the aircraft. This is often employed to reduce compressibility effects and delay transonic flight (Raymer, 2006). As this aircraft will never experience compressibility effects the design has a zero sweep angle for optimal aerodynamic efficiency.

**Taper ratio, \( \lambda \)**   is defined as the ratio of the tip chord length to the root chord length. It is suggested that a taper ratio of between 0.4 and 0.5 is most preferable for wings with low sweep angle and for a wing with zero sweep angle the ideal aerodynamic taper ratio should be approximately 0.45 (Raymer, 2006). It was also suggested that when vehicle weight is taken into account, then the ideal taper ratio would be approximately 0.4 (Raymer, 2006). Thus a taper ratio of 0.4 was selected based on Raymer’s recommendations.

**Dihedral angle**   is the angle of the wing relative to the horizontal, (positive above). A positive dihedral gives increased lateral stability. Since this wing is mounted relatively high on the fuselage the design already possesses inherent lateral stability and hence a dihedral angle of 0 degrees was chosen. This value correlates with the recommendations of Raymer (2006).

**Wetted surface area ratio**   is the ratio between the wing reference area and the total surface area of the design concept. This has been estimated, from preliminary sketches, as presented in Figure 3.2, as 4.0. This value correlated to that of similar profile aircraft presented in Raymer (2006).

**Propeller efficiency**   is required for accurate estimation of thrust. Raymer suggests that propeller efficiency is approximately 0.7 for a fixed pitch propeller, and ranges up to 0.85 for variable pitch propellers. A conservative estimate of 0.6 was used for preliminary sizing, however this could be reasonable considering the efficiency of propellers available for small scale aircraft.
3.5. AIRCRAFT PRELIMINARY SIZING

Preliminary Performance Parameters

Several performance parameters needed to be determined so the preliminary sizing process could be performed.

**Zero angle of attack drag coefficient**, $C_{D0}$, was required for all drag predictions during the sizing process. This can be estimated using a process developed in Raymer, which is a product of an equivalent skin friction coefficient and the wetted area ratio. For this type of aircraft it was assumed, from Raymer (2006), that the equivalent skin friction coefficient is approximately 0.0055. Thus the zero angle of attack drag coefficient, $C_{D0}$, would be of the order of 0.022 (Raymer, 2006).

**Oswald span efficiency factor** is a measure of the 'non-elliptic' lift distribution of a given wing. This parameter was estimated using Equation 3.6 (Raymer, 2006), as approximately 0.81.

$$e = 1.78(1 - 0.045A^{0.68}) - 0.64 \quad (3.6)$$

**Aerodynamic efficiency estimate** was determined using a method presented in Raymer (2006). This method required the calculation of a parameter the wetted area aspect ratio defined in Equation 3.7.

$$Wetted\_Aspect\_Ratio = \frac{A}{\left(\frac{S_{wet}}{S_{ref}}\right)} = 8.0/4 = 2 \quad (3.7)$$

Using this parameter an estimate of the aerodynamic efficiency, or $L/D$, was determined from Raymer (2006). For a design with landing gear this parameter would be approximately 12.6 and for a concept with an alternative launch and recovery method it would result in an $L/D$ estimate of approximately 15.

3.5 Aircraft Preliminary Sizing

The aim of the preliminary sizing process was to determine a design that met all design requirements. The approach undertaken in the project was similar to that presented in Raymer (2006) and Roskam (2004a); a classical aircraft design approach. The aim was to determine the major aircraft design parameters; wing loading and power loading. This process ensured that the final design met all of the design requirements including stall speed, cruise speed, take off distance requirements and climb requirements. This process culminated in the development of a matching diagram, a tool used to select an aircraft design point which met all design requirements.

The sizing process developed relationships that formed the limits of a feasible design, which met all design requirements. The following section presents the formulae required to develop the
matching diagram. From this graphical representation of the developed relationships, an initial design point was selected.

All relationships are defined in terms of wing loading and power loading.

**Wing loading**, $W/S$, is the wing area normalised weight of the aircraft.

**Power loading**, $W/P$, is the power normalised weight of the aircraft.

By determining these two parameters, and having the estimate of the final maximum take off weight, the required wing area, motor power and wing dimensions. These parameters are functions of the required vehicle performance, and were derived from the elementary equations of aircraft motion presented below.

**Stall requirements** are determined by the maximum lift coefficient of the wing and the stall speed required, (Equation 3.8). An estimate of the wing maximum lift coefficient, $C_{L_{\text{max}}}$ = 1.2, was determined for a clean wing (Raymer, 2006).

$$\left(\frac{W}{S}\right)_{\text{stall}} = q_s C_L$$

(Equation 3.8)

**Take off requirements** were determined for take off speed and the take off distance requirement (TODR). The take off speed is a requirement for CASR and is defined as 1.3 times the stall velocity. As there are no set runway lengths for model aircraft or UAVs a number of different take off distances were analysed, as lower take off distances were desirable, yet not critical. An analysis of the take off distance, taking into account the flexibility of the design and the implications of reducing take off distance further, was conducted using Equation 3.9.

$$S_G = \left(\frac{1}{2gK_A}\right) \ln \left(\frac{K_T + K_AV_t^2}{K_T + K_AV_i^2}\right)$$

(Equation 3.9)

Equation 3.9 can be further developed, using elementary aircraft equations and definitions, into the form of.

$$S_G = \left(\frac{1}{2g\left(\frac{\rho}{2W/S}\right)}\right) \left(\frac{\mu C_L - C_{D0} - KC_{L}^2}{\left(\frac{V}{V_T}\right) - 1 - \mu + \left(\frac{W}{P}\right) - 1 - \mu}\right)$$

(Equation 3.10)

It is worth noting that despite the seeming complexity of Equation 3.10 this is only an approximation of take off length. The actual take off distance of the resultant design may differ slightly from the TODR due to approximations for ground roll friction. The analysis was conducted assuming ground roll over a smooth tarmac surface.
3.5. AIRCRAFT PRELIMINARY SIZING

Climb requirements were defined by the speed of climb and the required climb gradient. The speed for climb was determined from CASA regulations for climb, to be at least 1.3 times the stall velocity with a climb gradient of 8.33%. A reserve factor of 40% was placed on this parameter to increase this gradient to 11.67%. The climb angle was then determined from trigonometry, (Equation 3.11).

$$\tan(\gamma) = CGR$$ \hfill (3.11)

Rearranging aircraft equations of motion for the climb, Equation 3.12 was developed. This equation defines the limitation for power loading and wing loading to meet these climb requirements.

$$\gamma = \sin^{-1} \left( \left( \frac{V}{\eta_p} \right) \left( \frac{W}{P} \right)^{-1} - C_{D0}q \left( \frac{W}{S} \right)^{-1} - \frac{1}{\pi A e} \left( \frac{W}{S} \right) \left( \frac{\cos^2 \gamma}{q} \right) \right)$$ \hfill (3.12)

Cruise requirements were specified in the feasibility study. This study predicted that a cruise speed of the order of 90 km/h would be required to successfully complete the mission. Equation 3.13 was determined from the equations of aircraft motion, to develop a relationship for cruise requirements.

$$\left( \frac{V}{\eta_p} \right) \left( \frac{W}{P} \right)^{-1} = C_{D0}q \left( \frac{W}{S} \right)^{-1} + K \left( \frac{W}{S} \right)$$ \hfill (3.13)

A matching diagram was developed from the above equations / relationships, (Figure 3.5). The matching diagram shows the ‘met area’, which is limited by cruise requirements, stall requirements and a take off distance requirement of 50 m. Thus it was determined that an aircraft with design parameters within this region would meet all design requirements.

Figure 3.5: Matching Diagram for the Predefined Design Requirements
3.6 Design Point

A design point was selected from the matching diagram in Figure 3.5. A brief discussion of the design choices is presented below, followed by the final selection of a design point.

A higher wing loading results in higher cruise efficiency. This reduces the requirements for batteries which is essential when trying to increase the endurance for an electric powered aircraft. A vehicle with a higher wing loading is also less susceptible to perturbations arising from wind gusts and adverse conditions. This advantage was identified as being extremely beneficial for the selected range of potential applications of this UAV, including the ARCAA UAV Outback Challenge.

A lower wing loading reduces stall speed and thereby reduces the operational risk during take off and landing. In addition, climb and take off distance performance requirements are reduced. However, this has an effect on the aerodynamic efficiency of the aircraft, as explained previously, and thereby increases the battery requirements.

A higher power loading implies the need for a motor with less power. This favourably reduces system weight, but reduces operational performance for climb, ground roll and acceleration in flight. A lower power loading implies the opposite.

The design point, shown in Figure 3.6 was chosen by relating the aforementioned choices to the aircraft performance and operational requirements. As the system has been proposed to be electric powered, cruise efficiency is imperative. Thus a relatively high wing loading was selected. Having briefly investigated electric motors and their power requirements, it was understood that increasing motor power results in a marginal increase in system weight but has a marked improvement in power and thus overall system performance. As the required power is inversely proportional to power loading, a large reduction in power loading will increase the system weight by too much. The chosen design point, is considered a reasonable trade off between operational performance and system weight. This design point resulted in the conceptual design of an aircraft that meets all design requirements with a take off distance of approximately 30m.

![Figure 3.6: Matching Diagram showing the Design Point](image)
The values of wing loading and power loading at the design point are as follows:

\[
\frac{W}{S} = 20.2 \text{kg/m}^2 \\
\frac{W}{P} = 13 \text{kg/kW}
\]

These design parameters result in the following wing area and required motor shaft power, for the design take off weight of 9kg:.

\[
S = 0.4455 \text{m}^2 \\
P = 690 \text{Watt}
\]

### 3.7 Tail Plane Sizing

The conceptual sizing of the tail plane provided details and dimensions of the required planforms of the horizontal and vertical stabilisers, as well as the associated lever arms of these surfaces. The sizing was conducted using the volume coefficient approach presented by Raymer (2006), and Roskam (2004a). The equations governing this method are presented below in Equation 3.14 and Equation 3.15.

\[
\bar{V}_H = \frac{S_H x_H}{S_C} \quad (3.14)
\]

\[
\bar{V}_V = \frac{S_V x_V}{S_b} \quad (3.15)
\]

### Design Specifications

The conceptual design of the tail plane was required to:

- Determine suitable tail volume coefficients based on commissioned UAVs and single-engined general aviation planes
- Provide suitable lever arms for the tail plane assuming a desired fuselage fineness ratio of 10
- Provide the planform areas for the tail plane
- Determine the require geometry for the tail plane
Determination of Volume Coefficients

Statistical data about the volume coefficients of commissioned UAVs was retrieved through careful investigation of images of the commissioned vehicles and used as baseline estimates for the coefficients of the UAV iSOAR. The derived estimates were compared with published data from Roskam (2004a) and Raymer (2006) for single-engined general aviation aircraft to produce more credible values. Table 3.3 contains the collected data from commissioned UAVs and the recommendations from Roskam (2004a) and Raymer (2006) can be found in Table 3.4.

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>$V_H$</th>
<th>$V_V$</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>'WSUV'</td>
<td>0.534</td>
<td>0.0368</td>
<td>Lee (2004)</td>
</tr>
<tr>
<td>BAI 'Javelin'</td>
<td>0.6364</td>
<td>0.0372</td>
<td>Janes-Information-Group (2002)</td>
</tr>
<tr>
<td>INTA 'Alo'</td>
<td>0.5935</td>
<td>0.0337</td>
<td>Janes-Information-Group (2002)</td>
</tr>
<tr>
<td>Aerosonde 'Aerosonde'</td>
<td>0.93</td>
<td>0.0201</td>
<td>Janes-Information-Group (2002)</td>
</tr>
<tr>
<td>Hangar 9 'American P-51D Mustang'</td>
<td>0.497</td>
<td>0.0429</td>
<td>N/A</td>
</tr>
<tr>
<td>Average</td>
<td>0.638</td>
<td>0.0341</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>$V_H$</th>
<th>$V_V$</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Single-engined General Aviation</td>
<td>0.7</td>
<td>0.04</td>
<td>Raymer (2006)</td>
</tr>
<tr>
<td>Single-engined General Aviation</td>
<td>0.67</td>
<td>0.0436</td>
<td>Roskam (2004a)</td>
</tr>
</tbody>
</table>

The resultant volume coefficients for the UAV derived from this method are as follows.

\[
\overline{V_H} = 0.8
\]

\[
\overline{V_V} = 0.04
\]

Discrepancies exist between the selected coefficient values and values that can immediately be drawn from the data, as a result of the specified control authority and stability level that the tail plane is to provide. By specifying larger volume coefficients, greater control authority and stability is achieved through the increased surface planform areas and / or lever arms. However, care was taken not to make these surfaces excessively large. This would create redundant weight in the tail plane and also create structural integrity problems within the airframe due to the excessive lift and moment generation caused by the larger surfaces and lever arms. Thus the sizing ensured the values of the coefficients fell within the bounds of the available, collected data.

The selected vertical tail volume coefficient is 17% larger than the average value from the collected data. This provides improved control authority and stability whilst remaining suitably sized, being the recommended value from both Raymer (2006) and Roskam (2004b). Similarly, the selected horizontal tail volume coefficient is 25% larger than the average value, providing
increased control authority and stability, whilst positioning the design of the UAV within the collected data range.

**Determination of Lever Arms**

The lever arms of the horizontal and vertical stabilisers are the distances between the quarter chord points of the mean aerodynamic chords of the surfaces and the quarter chord point of the mean aerodynamic chord of the main wing (Raymer, 2006). The lever arms are measured in percent of the Mean Aerodynamic Chord (MAC) of the main wing. The definition of the lever arms is shown in Figure 3.7.

Data on the lever arms of commissioned UAVs was used to assist the specification of the lever arms for the horizontal tail. This data is presented in Table 3.5.
Table 3.5: Collected Lever Arm Data - Commissioned UAVs

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>XH [%MAC]</th>
<th>XV [%MAC]</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>‘WSUAV’</td>
<td>343</td>
<td>343</td>
<td>Lee (2004)</td>
</tr>
<tr>
<td>Hangar 9 ‘American P-51D Mustang’</td>
<td>293</td>
<td>333</td>
<td>Measured by group</td>
</tr>
<tr>
<td>Average</td>
<td>346.2</td>
<td>354.2</td>
<td></td>
</tr>
</tbody>
</table>

With the assistance of the data from table 3.5, the following lever arm sizes were derived.

\[ x_H = 400\% \bar{C} = 975mm \]

\[ x_V = 400\% \bar{C} = 975mm \]

To assist in achieving the desired fuselage fineness ratio, the lever arms were selected to be greater than the average value from the data. In addition, since there is no restriction on having unequal horizontal and vertical lever arm values, both were specified to be equal, to assist the manufacturing process and improve aesthetics. Uneven tail arms are used to improve performance during adverse conditions (Raymer, 2006) such as spins, however as this aircraft will not be performing complicated manoeuvres and so this was deemed unnecessary.

**Determination of Planform Areas**

Using the above results in conjunction with equation 3.14 and equation 3.15, the planform areas of the horizontal and vertical stabilisers were calculated. These areas were found to be:

\[ S_H = 0.0863m^2 \]

\[ S_V = 0.0345m^2 \]

**Determination of Tail Plane Geometry**

Recommendations from Raymer (2006) were used to determine suitable aspect ratios and taper ratios of the tail plane. These values were then used to determine the span of the stabilisers, as well as the root and tip chord lengths of the tail plane using Equation 3.16 for aspect ratio.
\[ A = \frac{b^2}{S} = \frac{2b}{C_{\text{root}}(1 + \lambda)} \]  

where \( \lambda = \frac{C_{\text{tip}}}{C_{\text{root}}} \)

The above analysis produced the following results for the geometry of the tail plane.

**Horizontal Tail**

\[ A_H = 6 \]
\[ \lambda_H = 0.5 \]
\[ b_H = 0.719 m \]
\[ C_{\text{root}_H} = 0.16 m \]
\[ C_{\text{tip}_H} = 0.08 m \]

**Vertical Tail**

\[ A_V = 2 \]
\[ \lambda_V = 0.5 \]
\[ b_V = 0.263 m \]
\[ C_{\text{root}_V} = 0.175 m \]
\[ C_{\text{tip}_V} = 0.088 m \]

### 3.8 Propulsion System Selection

An electric propulsion system was to be utilised for this UAV, as specified from the base specifications of the project. The propulsion system encompasses the components which provide the vehicle with thrust. These components are the motor, propeller, batteries, and Electronic Speed Controller (ESC).

Given the time and cost constraints of the project, as well as the availability of proprietary electric propulsion systems for small scale aircraft, it was deemed a poor use of resources to develop custom propulsion componentry. Instead, potentially suitable system elements were identified from market surveys and compared to find those components which best satisfied the relevant specifications.

The market surveys focused on identifying components available from sources within Australia. In addition to making component acquisition simple, the benefits related to replacement units as well as the provision of after sales assistance lead to sourcing components locally.
### 3.8.1 Specifications of Propulsion System

Each of the four individual components of the propulsion system have unique requirements, these are discussed separately in the relevant sections below. However, collectively, the components were required to complement each other and satisfy the underlying specifications of the propulsion system. Arising from the results of the conceptual design of the aircraft, these requirements were:

- Single motor
- $P_{ShaftMax} = 690\, W$
- $P_{ShaftCruise} = 215\, W$
- $v_{cr} = 90\, km/h$
- Provide an endurance of a minimum of 1 hour
- Minimal weight
- Efficient
- Cost effective
- Composed of readily available components

In addition, it was essential that the power consumed by the aircraft during the mission was estimated. Assumptions were drawn from the mission profile, resulting in the following.

- Aircraft utilizes maximum shaft power for 2 minutes (takeoff and climb)
- Aircraft utilizes cruise shaft power for 60 minutes (cruise, loiter and recovery)
- Power supply has a 10% reserve factor

### 3.8.2 Motor Selection

**Detailed Specifications of the Motor**

Used to produce the propulsive power of the vehicle, the selected motor was required to satisfy the following specifications:

- Minimal weight
- High efficiency
- Cost effectiveness
- Be readily available
3.8. PROPULSION SYSTEM SELECTION

Motor Type

Two types of electric motor are commonly used to power small scale aircraft, these two types being brushed and brushless motors. According to Model Flight Pty. Ltd. brushless motors suffer from less frictional losses than brushed motors. In addition, higher power output and longer service life can be attained with a brushless motor when compared to a similar brushed device (ModelFlight, 2007). The above information is supported by data from electric motor manufacturer, ModelMotors (2006). As can be seen in Table 3.6, a comparison between a brushed motor and a similar sized brushless motor, (based on recommended aircraft weight,) shows that a brushless device is significantly more efficient. In addition, the data shows that the brushless motor is also lighter, another significant benefit of brushless technology.

Table 3.6: Comparison of Brushed and Brushless Motor Technology

<table>
<thead>
<tr>
<th>Motor</th>
<th>Maximum $\eta_{motor}$ (%)</th>
<th>Aircraft Weight (g)</th>
<th>Motor Weight (g)</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>VM 2410 (Brushed)</td>
<td>74</td>
<td>2000</td>
<td>242</td>
<td>ModelMotors (2006)</td>
</tr>
<tr>
<td>AXI 2814/20 Gold line</td>
<td>83</td>
<td>1800</td>
<td>106</td>
<td>ModelMotors (2006)</td>
</tr>
</tbody>
</table>

The use of an efficient motor has the follow on effect of decreased battery requirements, and hence weight, for a given endurance target. It is for this reason, as well as the additional power potential, that a brushless electric motor type was selected for the propulsion system.

Motor Candidates

An extensive market survey of available motors within Australia identified five potential candidates for the motor, all five being brushless types. It should be noted that the power figures that were able to be obtained from the market survey are motor power figures. Thus, the ability of each motor to meet the shaft power specifications of the propulsion system required further analysis. Relevant information on the five candidates is presented in Table 3.7.

(Note: Motor power output is dependent upon batteries and propeller used in conjunction with motor.)

Table 3.7: Motor Candidates

<table>
<thead>
<tr>
<th>Motor</th>
<th>$K_v[\text{RPM/V}]$</th>
<th>$[\text{g}]$</th>
<th>$P_{\text{max}}[\text{W}]$</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dualsky 50-60</td>
<td>310</td>
<td>377</td>
<td>1300</td>
<td>Dualsky (2002)</td>
</tr>
<tr>
<td>AXI 4130-16</td>
<td>385</td>
<td>409</td>
<td>1000</td>
<td>ModelMotors (2006)</td>
</tr>
<tr>
<td>AXI 4130-20</td>
<td>305</td>
<td>409</td>
<td>1000</td>
<td>ModelMotors (2006)</td>
</tr>
<tr>
<td>E-Flite Power 60</td>
<td>400</td>
<td>380</td>
<td>1200</td>
<td>E-Flite (2007)</td>
</tr>
</tbody>
</table>
Comparison of Candidates

Without the finances or time to purchase all five motor candidates, and independently test them all, a virtual test was conducted using available motor data, and some assumptions, to determine the most suitable device. The most suitable would be the motor allowing for the lightest motor and battery combination.

The virtual test involved the determination of the battery charge required by each motor to perform a hypothetical mission. The following assumptions were made with regards to this mission.

- Max power flight for 2min

- Cruise power flight for 60min

- Operation with Lithium-Polymer, (Li-Po,) battery packs, with 8 cells connected in series, as recommended to allow each motor to produce $P_{shaft_{max}}$. (7 for E-Flite Power 60.)

In addition, assumptions were made regarding the efficiency of the motors in maximum power use and cruise use, as well as the operating voltage of each motor. Motors with $Kv$ values were assigned higher voltage values, as information obtained from manufacturer’s suggested that such motors were capable of exploiting more battery voltage. Manufacturer’s data was only available for the AXI motors, and as such better efficiencies were assumed than for the other motors.

The selected motor was to be the one which required the least battery charge requirement, and hence weight. The charge requirements for each motor were calculated using Ohm’s Law, and the relationships between battery power and motor power, shown in Equation 3.17.

$$C_{battery} = \sum I_{motor} t = \frac{tP_{shaft}}{\eta_{motor} V_{motor}} \bigg|_{max} + \frac{tP_{shaft}}{\eta_{motor} V_{motor}} \bigg|_{cruise}$$  \hspace{1cm} (3.17)

The results of the analysis can be seen in Table 3.8.
### Table 3.8: Required Battery Charge for Candidate Motors based on Hypothetical Mission

<table>
<thead>
<tr>
<th>Motor</th>
<th>$\eta_{\text{motor}}$ (Max Power) (%)</th>
<th>$\eta_{\text{motor}}$ (Cruise Power) (%)</th>
<th>No. of LiPO Cells</th>
<th>$V_{\text{motor}}$ (V)</th>
<th>$C_{\text{battery}}$ (Ahr)</th>
<th>Manufacturer’s Data</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dualsky 50 − 60</td>
<td>83</td>
<td>83</td>
<td>8</td>
<td>23.27</td>
<td>13.25</td>
<td>No</td>
</tr>
<tr>
<td>Model Motoras AXI4130 − 16</td>
<td>85</td>
<td>85</td>
<td>8</td>
<td>22.73</td>
<td>11.76</td>
<td>Yes</td>
</tr>
<tr>
<td>Model Motors AXI4130 − 20</td>
<td>85</td>
<td>85</td>
<td>8</td>
<td>22.73</td>
<td>11.76</td>
<td>Yes</td>
</tr>
<tr>
<td>E-Flite Power 60</td>
<td>82</td>
<td>82</td>
<td>7</td>
<td>26.92</td>
<td>15.33</td>
<td>No</td>
</tr>
<tr>
<td>E-Flite Power 110</td>
<td>83</td>
<td>83</td>
<td>8</td>
<td>23.27</td>
<td>12.04</td>
<td>No</td>
</tr>
</tbody>
</table>

From the above results, the AXI motors were considered suitable choices. The Dualsky 50-60 and E-Flite Power 110 were disregarded due to their high battery requirements. The E-Flite Power 60 was still considered an option, even though it required more battery power. This was because the Power 60 operated on seven Li-Po cells, rather than eight like the AXI motors, and as such may have required less battery mass.

Using information from ModelFlight (2007), it was determined that the Power-60 would require approximately 100 grams more battery weight than the AXI motors to achieve the same performance. Consequently, it too was disregarded.

### Selected Motor

The motor comparison highlighted two candidates, which appeared almost identical. The difference between the two was in the motor constant. Advice was sought from Model Motors, as to which would be most suitable for the UAV. The group was informed that the higher motor constant of the 4130-16 means it produces less torque than the 4130-20, and consequently suffers from three problems compared with the 4130-20.

- Is required to spin faster to produce an equivalent amount of power, as $P_{\text{motor}} \propto \omega_{\text{motor}}$
- If the motor spins faster, a smaller propeller diameter must be used to prevent sonic flow at the propeller tips. This affects the efficiency of the propeller as larger diameter propellers are more efficient to smaller devices.
- The motor may not have enough torque to overcome the resistive forces produced by the propeller under load, and hence may produce less power.
Consequently, based on these reasons, the Model Motors AXI 4130-20 Gold Line, (Figure 3.8) was selected as the motor for the propulsion system. This selection was supported by advice from Jan Vaclavik, (2007,) an employee of the motor manufacturer Model Motors.

![Selected Motor for Propulsion System - The Model Motors AXI 4130-20 Gold](ModelMotors_2006.png)

Figure 3.8: Selected Motor for Propulsion System - The Model Motors AXI 4130-20 Gold (ModelMotors, 2006)

### 3.8.3 Propeller Selection

#### Detailed Propeller Specifications

- Allow aircraft to reach $v_{cr} = 90km/hr$
- Optimum efficiency at $v_{cr}$
- Diameter greater than twice the hydraulic diameter of the fuselage, $D_{prop} > 0.30m$ to allow for adequate prop wash
- Diameter small enough to allow adequate ground clearance for RC flight tests, $D_{prop} < 0.56m$
- Diameter small enough to prevent sonic flow at tips. Information from Model Motors, (2006,) suggest that the maximum speed that the motor will attain, $\omega_{motor_{max}} \approx 7000RPM$. Thus to prevent sonic flow at the tips at operational altitude, $D_{prop} < 0.90m$
- Suitable for electric motors
- Light weight
- Folding type for vertical recovery
- Proprietary type
- Be available

#### Propeller Sizing

The sizing of the propeller was performed using recommendations from the motor manufacturer as well as Simons (2002). Proprietary type propellers are specified in terms of diameter and pitch, and thus these were the two values which needed to be determined for the UAV application.
An estimation of propeller pitch was attained from Figure 3.9. The figure suggested a propeller pitch of;

\[ P_{\text{prop|simons}} = 9\text{inches} \]  

(3.18)

based on

\[ \omega_{\text{motor,max}} = 7000\text{RPM} \]  

(3.19)

\[ v_{cr} = v_{cr\text{wind}} = 100\text{km/hr} \]  

(3.20)

(desired cruise speed with 10% reserve factor for wind gusts)

The suitability of Simons estimate was checked via personal communication with the motor manufacturer. ModelMotors (2006) supported the results of Simons (2002), suggesting a similar pitch size

\[ 8\text{inches} < P_{\text{prop|model-motors}} < 17\text{inches} \]  

(3.21)

Both recommendations were carefully considered before a final pitch was specified. The ultimate pitch size,
was based on the desire to achieve optimal performance at cruise speed. The 9inch pitch would allow the vehicle to achieve a higher cruise speed than an 8inch pitch propeller, as a higher pitch allows for higher vehicle speeds to be attained (Raymer, 2006). High pitch propellers do however suffer from poorer takeoff and climb performance, but the use of a car launch system with the UAV, along with the need for only shallow climb gradients to satisfy the mission, meant that these negative effects would have little impact on the performance of the aircraft.

The diameter of the propeller was again selected from recommendations from the motor manufacturer. ModelMotors (2006) suggests the following:

\[ 16 \text{inches} < D_{\text{prop}|\text{model\textendash}motors} < 17 \text{inches} \]  

(3.23)

These values are within the diameter limits specified for the propeller.

Further analysis and consideration had to be made before a final propeller diameter was chosen. Larger diameter propellers provide improved propeller efficiency (Raymer, 2006) but the increased weight requires more motor power to spin, and may prevent the motor from attaining maximum speed if the device cannot produce enough torque to turn the device at maximum speed. However, weight tests conducted (Table 3.9) on two propellers of different materials and sizes showed that a larger propeller may be lighter than a smaller one if it is constructed of a different material. Hence the propeller was specified to be the largest of this range.

\[ D_{\text{prop}} = 17 \text{inches} \]  

(3.24)

Table 3.9: Propeller Material and Diameter Weight Comparison

<table>
<thead>
<tr>
<th>Propeller</th>
<th>Material</th>
<th>Diameter (inches)</th>
<th>Pitch (inches)</th>
<th>Weight (g)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aeronaut CAM Carbon</td>
<td>Carbon fibre</td>
<td>17</td>
<td>9</td>
<td>32</td>
</tr>
<tr>
<td>APC</td>
<td>Nylon</td>
<td>16</td>
<td>8</td>
<td>23</td>
</tr>
</tbody>
</table>

Propeller Type

A carbon fibre folding type propeller was selected for the propulsion system. Not only did such a system satisfy the specifications associated with the vertical recovery of the UAV, it also followed the recommendations of the motor manufacturer, Model Motors, who specify the use of folding type propeller with the AXI 4130-20 Gold Line for maximum system efficiency.

The use of a carbon fibre device also allowed for a propeller which was lighter than comparable plastic units. The results presented in Table 3.9, show that a 25% decrease in weight can be achieved by using a carbon fibre system over a similar sized nylon propeller.
3.8. PROPULSION SYSTEM SELECTION

Selected Propeller

**Mission Propeller**  The Aeronaut CAM Carbon 17inches × 9inches (diameter \× pitch) carbon fibre folding propeller was selected as the optimal propeller for the UAV, and its required mission. This propeller was found to satisfy all of the propeller specifications.

**Propeller for Initial RC Tests**  For RC tests, it was expected that damage to the propeller may occur due to poor landings. Consequently, APC brand, nylon propellers were used for RC testing. These propellers were more readily available than the carbon fibre units, and only one third of the price. Thus, they could easily and affordably be replaced if damaged during initial flight tests.

These propellers were suitably sized for the motor, with dimensions 16inches × 8inches (diameter \× pitch). It can be seen that the nylon units were not optimally sized for the mission requirements. However, since initial RC testing did not require optimal cruise efficiency, it was deemed acceptable to use these devices initially, and then use the carbon fibre units for actual missions.

**Electronic Speed Controller Selection**

The Electronic Speed Controller (ESC) is an electronic throttling device, used to control the speed and power output of the electric motor. A suitable ESC had to be matched to the motor, and battery supply, to provide adequate performance, efficiency and safety.

**Specifications of ESC**

The ESC was required to satisfy the following requirements related to the selected motor and Li-Po battery supply:

- Suitable for brushless motors
- Capable of operating with 8 Li-Po cells connected in series, as per the motor power requirements
- Capable of providing undervoltage protection to Li-Po cells. This ability inhibits operation of motor if cell voltage falls below 3.0V, the minimum voltage to be maintained by cells to prevent destruction of batteries, (Autography-Flight-Technology-Ltd, 2005).
- Safely handle the maximum current that the motor can draw of 55A (ModelMotors, 2006)
- Cost effective
- Be available
ESC Options

Given the time and cost constraints of the project, as well as the availability of proprietary ESC systems, it was deemed a poor use of resources to develop a custom ESC for the propulsion system. Suitable, available ESCs were instead identified from a market survey and compared for suitability. The market survey identified three systems which were subsequently compared for suitability. Table 3.10 contains the information on the three systems, which was used to base a selection on.

Table 3.10: Specifications of ESC Options

<table>
<thead>
<tr>
<th>ESC</th>
<th>Brushless Motor Compatible</th>
<th>Li-PO Operating Cell Count</th>
<th>Max Continuous Current [Amps]</th>
<th>Weight [g]</th>
<th>Cost (source)</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Kontronik Jazz 55 – 10 – 32</td>
<td>Yes</td>
<td>4-10</td>
<td>55</td>
<td>59</td>
<td>$321.61 (Perth RC Models and Hobbies)</td>
<td>Kontronik (ND)</td>
</tr>
<tr>
<td>Dualsky XC9036HV</td>
<td>Yes</td>
<td>2-12</td>
<td>90</td>
<td>125</td>
<td>$249 (Model Flight Pty. Ltd.)</td>
<td>Dualsky (2002)</td>
</tr>
</tbody>
</table>

Selected ESC

All identified ESCs satisfied the specifications of the device. Consequently, the ultimate selection of the device was based on weight, cost and current handling capabilities.

The cheapest option, the XC9036HV was dismissed due to its substantial weight. In addition, the current handling capabilities of the device were deemed excessive for the UAV application. Similarly, the Jazz 55-10-32, was discounted. Not only is the Jazz 55-10-32 heavier and more expensive than the MasterSpin OPTO 75, it has no safety buffer to handle the maximum expected current draw of the motor. Consequently, the MasterSpin 75 OPTO (Figure 3.10) was the selected ESC for the aircraft.
3.8. PROPULSION SYSTEM SELECTION

3.8.4 Battery Selection

Detailed Battery Specifications

The selected batteries are required to satisfy the following specifications:

- Minimal weight
- Provide enough charge for a minimum endurance of 1 hour such that the aircraft can utilize maximum shaft power for 2 minutes, and cruise shaft power for 60 minutes

Battery Type

The selected motor is capable of operating from Lithium-Polymer (Li-Po) or nickel metal hydride (Ni-MH) battery packs. The choice of battery type was driven by the need to keep the weight of the propulsion system to a minimum.

A weight test conducted by the project group revealed that for a given voltage and charge capacity, Ni-MH batteries are approximately twice as heavy as a comparable Li-Po battery pack (Table 3.11). Consequently, Li-Po type batteries were selected for the propulsion system.

<table>
<thead>
<tr>
<th>Battery Type</th>
<th>Voltage [V]</th>
<th>Charge [Ahr]</th>
<th>Weight [g]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ni-MH</td>
<td>10.8</td>
<td>1.000</td>
<td>175</td>
</tr>
<tr>
<td>Li-Po</td>
<td>11.1</td>
<td>1.000</td>
<td>81</td>
</tr>
</tbody>
</table>

Required Battery Power

From the motor selection section, it was found that the selected motor requires battery packs with 8 Li-Po cells connected in series, which collectively can produce 11.76Ahr of charge. With the specified charge reserve factor of 10%, the propulsion system requires approximately 13Ahr of charge.
Selected Battery Units

Following a market survey, it was found that these requirements could be satisfied with the use of 8 FlightPower FPEVO20-33004S battery packs. Each of these battery packs have 4 Li-Po cells connected in series, and are capable of holding 3.300Ahr of charge. By connecting two of these packs in series, a ‘twin’ pack with 8 Li-Po cells connected in series, and capable of holding 3.300Ahr of charge, can effectively be created. If four of these ‘twin’ packs are connected in parallel, then a battery pack with 8 Li-Po cells connected in series, and capable of holding 13.200Ahr of charge, can be created, which satisfies the required specification.

3.9 Material Availability

Material selection for this airframe was driven by two main factors, namely specific strength and weight. The availability of materials was a factor considered for the construction of this aircraft. Commonly used composite materials were an attractive material to be used for most components, however the properties of these materials needed to be quantified in order to ensure structural integrity.

A variety of commonly available materials were identified to be used in the construction of this aircraft. The structural requirements, construction method and manufacturing methods were considered for each component individually to select the appropriate material. Properties of each material need to be carefully considered as each component has different requirements for strength and electrical transparency. Table 3.12 shows the materials that were identified as both available and suitable for the purposes of this construction process. Most components were designed with these materials in mind.
Table 3.12: Available Materials (AutomationCreations, ND & Raymer, 2006)

<table>
<thead>
<tr>
<th>Material Type</th>
<th>Materials Name</th>
<th>Density ($kg/m^3$)</th>
<th>Yield stress (MPa)</th>
<th>Shear Strength (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cloth</td>
<td>E grade fibreglass</td>
<td>1965</td>
<td>724</td>
<td>70.3</td>
</tr>
<tr>
<td></td>
<td>S grade fibreglass</td>
<td>1439</td>
<td>1509</td>
<td>51</td>
</tr>
<tr>
<td></td>
<td>Carbon fibre - Weave</td>
<td>1600*</td>
<td>345*</td>
<td>35*</td>
</tr>
<tr>
<td></td>
<td>Carbon fibre - Uni</td>
<td>1600</td>
<td>550</td>
<td>70</td>
</tr>
<tr>
<td>Resin</td>
<td>Epoxy resin</td>
<td>1550</td>
<td>96.5</td>
<td>N/A</td>
</tr>
<tr>
<td></td>
<td>Polyester resin</td>
<td>1810</td>
<td>185</td>
<td>N/A</td>
</tr>
<tr>
<td>Sandwich Core</td>
<td>Isolite expanded PS foam</td>
<td>24</td>
<td>0.14 (comp)</td>
<td>0.26</td>
</tr>
<tr>
<td></td>
<td>Extruded blue styrofoam</td>
<td>40</td>
<td>0.7</td>
<td>0.4</td>
</tr>
<tr>
<td></td>
<td>Honeycomb core</td>
<td>30</td>
<td>1.4</td>
<td>1.1</td>
</tr>
<tr>
<td></td>
<td>Closed cell foam core</td>
<td>41</td>
<td>0.5</td>
<td>0.6</td>
</tr>
<tr>
<td>Filler</td>
<td>Microsphere filler</td>
<td>250</td>
<td>N/A</td>
<td>N/A</td>
</tr>
<tr>
<td></td>
<td>Milled glass</td>
<td>1360</td>
<td>130**</td>
<td>N/A</td>
</tr>
<tr>
<td>Laminate</td>
<td>Ply wood</td>
<td>31</td>
<td>6.2 (edge)</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Balsa wood</td>
<td>160</td>
<td>73 (axial)</td>
<td>1.1</td>
</tr>
<tr>
<td>Metal</td>
<td>Aluminium alloy 2024-T3</td>
<td>2780</td>
<td>345</td>
<td>283</td>
</tr>
<tr>
<td></td>
<td>Mild steel threaded rod</td>
<td>7870</td>
<td>260</td>
<td></td>
</tr>
</tbody>
</table>

* denotes epoxy mix
** denotes ultimate stress

3.10 Results of the Platform Conceptual Designs

This process included the development of a conventional design concept with a low, externally mounted payload, low mounted camera, single tractor propeller, high mounted parachute bay, and mid to high mounted wing. In addition to sizing the aircraft, some materials were identified as being suitable for this application. The individual component material selection will be covered in the detailed design process. Figure 3.11 lists the design parameters that have been determined from the conceptual design phase as well as the final three view concept CAD model.
Figure 3.11: Simplified Three View of the final Design Concept
Chapter 4

Control System Concepts

The autopilot is the central component of an automated UAV system. It uses sensors to determine the UAVs position, orientation, airspeed and altitude. Given this sensory information, the autopilot is able to stabilise, control and navigate the UAV. A significant expenditure of project time was allocated to the selection of the controls systems and the approach taken in this regard. This element of the design would represent a major portion of the overall financial expenditure of the project. This section will examine the selection of an autopilot system.

4.1 Components

Initially the components of modern UAV autopilots will be considered. Commercially available autopilots are composed of some, or all of the following components:

- Central Processing Unit (CPU)
- GPS receiver
- Gyroscopes
- Accelerometers
- Pressure transducers
- Electronic compass
- Ultrasonic altimeter

In order to consider the desirable characteristics of the autopilot system, the function of these components will be discussed.

The CPU is the most significant component of the control system, as it performs the primary tasks of the autopilot. This processing unit is responsible for the manipulation of input data from the sensors, using this data to generate the necessary outputs to the control surfaces. Not only is data manipulation an essential task of the processor, it is responsible for maintaining
the stability and control of the aircraft, allowing the aircraft to perform the given navigational commands specified by the user. Hence, the processor speed and quality should be considered in the autopilot selection process to ensure quick and efficient communication between the autopilot and aircraft control surfaces.

The GPS unit provides positional information to the main processing unit via communication with satellites. The GPS is used primarily for navigational purposes, giving the aircraft’s position and heading, as well as being used to estimate the aircraft’s altitude and speed.

All autopilot systems incorporate a series of gyroscopes and accelerometers, sometimes called and internal measurement unit (IMU). The gyroscopes measure the pitch, roll and yaw of the UAV during flight, while the accelerometers measure the UAV’s acceleration in the x (longitudinal) and y (lateral) directions. This can be used to coordinate flight manoeuvres, maintain aircraft stability and as a form of dead reckoning in the event of GPS failure.

Two pressure transducers can be used in order to measure both static and dynamic pressure. The manipulation of data from these two sensors allows the calculation of both airspeed and altitude.

Autopilots may incorporate the use of an electronic compass in order to provide information relating to the aircraft’s heading. An electronic compass may be incorporated into the navigational software of the autopilot and used in conjunction with the GPS. This arrangement is particularly useful if the UAV is operating in strong winds, as it specifies the direction in which the aircraft is travelling. The electronic compass can also be used in conjunction with pressure transducers for dead reckoning in the event of GPS failure.

Some autopilots use an external Ultrasonic Altitude Sensor (UAS), which enables the measurement of altitude to an accuracy of a few centimetres whilst flying at low altitudes. Altitude may also be calculated through the use of a GPS and pressure transducer unit. However, the accuracy provided is less than that of the UAS, with an accuracy of a few metres. Automated landings require a high degree of accuracy in regards to the altitude measurement, an accuracy which is not attainable through the use of a GPS unit coupled with a pressure transducer. Hence, if automated landings and take-offs with conventional landing gear are required, the use of a UAS is required.

### 4.2 Control System Specifications

In order to determine a suitable autopilot for the UAV, two elements needed to be considered. The functionality of the autopilot needed to meet the requirements for automation and in-flight monitoring as set out in Feasibility Study (see Section 2.6.4). Given the required functionality, the selection approach was optimised by considering certain selection parameters, which are detailed below. Some criteria were due to project constraints, and others were performance based parameters. Due to the groups relative inexperience in this area it was difficult to determine the exact requirements of the system prior to purchase. This resulted in the performance based parameters of available systems being considered and the most suitable system within the budget.
was selected. The decisions made regarding the control system were then used in the final sizing process. Note that 'complete system' refers to everything required in order to implement the control systems and communication with ground station, but exclude the servos themselves. This holistic view had to be used, as some options considered combined communications equipment with the control systems themselves.

**Constraints**

- Budget for complete control system ≈ $10,000
- Development time or Lead time < 2 month
- Functional Requirements:
  - Provide for autonomous flight excluding take-off, landing, and payload deployment
  - Provide for in-flight monitoring of the aircraft

**Optimisation Criteria**

<table>
<thead>
<tr>
<th>Priority</th>
<th>Minimise</th>
<th>Maximise</th>
</tr>
</thead>
<tbody>
<tr>
<td>Highest</td>
<td>weight</td>
<td>functional flexibility</td>
</tr>
<tr>
<td>Lowest</td>
<td>power consumption</td>
<td>software adaptability</td>
</tr>
</tbody>
</table>

**4.2.1 Off-The-Shelf versus Custom Built**

The first major decision that required consideration was how to obtain an autopilot system. The two options were to utilise a complete off-the-shelf autopilot solution, or design and build a custom autopilot system for this aircraft. It should be noted that an open source autopilot project currently exists on the internet, which, in the case of a custom design, could have been used to form the basis of the autopilot system.

The use of a custom built autopilot has several advantages for this project. The primary advantage is the relatively low budget required, as purchasing all the required individual components can cost considerably less than that of an assembled system. In addition, the programming method, interface capabilities and in-flight communication abilities of the autopilot can be tailored to meet the requirements of the UAV. This flexibility of operation would be advantageous in terms of creating a UAV that could be used in many different civil applications, as well as having the potential to become a research platform within the University.

Despite the benefits of a custom built control system, the development of an autopilot involves many complications. A custom built solution will likely not have the reliability achieved with an off-the-shelf autopilot due to the time constraints of the project. In addition to the decrease in
reliability, the time required to design, build and program an autopilot is substantial and may prevent the completion of deliverables, in particular entrance to the ARCAA Outback Challenge (ARCAA, 2007). For a baseline comparison, research found that BAE Systems required an eight month period in their attempt at ‘rapid development’ of an autopilot for an Eagle 150 airframe (Liang, 2002).

These considerations were weighted according to their significance for this project and compiled into a decision matrix. This decision matrix helped quantify this decision and is shown Table 4.2.

Table 4.2: Decision matrix considering a custom built autopilot versus a commercially available unit

<table>
<thead>
<tr>
<th></th>
<th>Weighting (%)</th>
<th>Custom-built Autopilot</th>
<th>Commercial Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Development Time</td>
<td>30.3</td>
<td>9.1</td>
<td>30.3</td>
</tr>
<tr>
<td>Reliability</td>
<td>25.8</td>
<td>5.2</td>
<td>25.8</td>
</tr>
<tr>
<td>Design Complexity</td>
<td>18.2</td>
<td>7.3</td>
<td>18.2</td>
</tr>
<tr>
<td>Flexibility</td>
<td>13.6</td>
<td>13.6</td>
<td>8.2</td>
</tr>
<tr>
<td>Price</td>
<td>12.1</td>
<td>12.1</td>
<td>3.6</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>100</strong></td>
<td><strong>47.3</strong></td>
<td><strong>86.1</strong></td>
</tr>
</tbody>
</table>

Due to the aforementioned time constraints and reliability considerations, an off-the-shelf autopilot was considered the option most suited to the goals of this project. This decision was further reinforced with procurement of industrial sponsorship.

### 4.2.2 Autopilot Market Review

Many off-the-shelf autopilots were considered, with the autopilots and corresponding manufacturers listed below.

- 1028g, 2028g and 2128g by Micropilot (2007a)
- Kestrel 2.22 by Procerus (2007c)
- Microbiot AP by Microbotics (2007)
- IFCS-6000 by Integrated-Dynamics (2007)
- 3400 by U-NAV (2007)
- WePilot2000 by weControl (2007)
- Micro guidestar by Athena (2007)
Initially, specifications were obtained for all the above autopilots to allow comparison. By considering these specifications, particularly the aspects included in the criteria above, four autopilots were selected as suitable for the control system of this project. In-depth negotiations then occurred with these four manufacturers. The selected autopilots were the Piccolo Plus, Kestrel 2.22, 2028g and the AP04. These autopilots are summarised in Table 4.3 with the physical characteristics and the initial indicative prices of the autopilots. Prices listed are approximate, based on an exchange rate of $AU1 to $US0.8.

<table>
<thead>
<tr>
<th>Autopilot</th>
<th>Weight (g)</th>
<th>Size (mm)</th>
<th>Power Consumption (V.mA)</th>
<th>Country of Origin</th>
<th>Indicative Price (AUD)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Piccolo Plus</td>
<td>212</td>
<td>122x61x38</td>
<td>12, 400</td>
<td>USA</td>
<td>$10000</td>
</tr>
<tr>
<td>Kestrel 2.22</td>
<td>16.8</td>
<td>50x35x12</td>
<td>5, 500</td>
<td>USA</td>
<td>$6250</td>
</tr>
<tr>
<td>2028g</td>
<td>28</td>
<td>100x40x15</td>
<td>6.5, 140</td>
<td>Canada</td>
<td>$6875</td>
</tr>
<tr>
<td>AP04</td>
<td>300</td>
<td>74x68x60</td>
<td>10, 400</td>
<td>Spain</td>
<td>$12250</td>
</tr>
</tbody>
</table>

The functionality of all four autopilots is sufficient for use in this project. The autopilots are capable of autonomous take off and landing, which is an important attribute in terms of the long-term development of iSOAR. The autopilots also allow the programming of sufficient waypoints (greater than 100) both before and during flight. Waypoints must be configurable during flight for successful entry into the UAV outback challenge, as the UAV’s path needs to be adapted depending on the images obtained. The autopilots accept RC override and are capable of maintaining continuous data link with a ground station. RC override is a necessity for testing and safety purposes during the autopilots operation. In order to provide data link however, some of the autopilots require additional external units.

**Piccolo Plus**

The Piccolo Plus autopilot from Cloud Cap Technologies (Figure 4.1) is an exceptionally complete control system. The system includes an integrated UHF data link with 40kBaud additional bandwidth for user defined functions such as image downlink. Therefore this system requires fewer communication links simplifying the system and minimising interference possibilities. The control software is also configured to provide dead reckoning in the event of GPS loss.
Through communication with Cloud Cap Technologies the negotiation of a significant educational discount was initiated. This may have been a sufficient discount for the autopilot and additional components, making it affordable and within project budget. However, due to export laws in the US, the lead time for a Piccolo Plus autopilot was in the order of 6 months. Therefore, even though this system is highly suited to this application, it is not practicable to purchase and test the autopilot within the time constraints of this project.

**Kestrel**

The Kestrel 2.22 autopilot (Figure 4.2) produced by Procerus is a very compact and lightweight autopilot. In addition to the standard autopilot sensors, the Kestrel has a magnetometer and configurable input/output functionality. The magnetometer allows for dead reckoning in the event of GPS loss.
Unlike other packages, the Kestrel package does not include the software required to interface from a ground station. The additional cost of software is approximately $3000, causing this to be a less economical option than it initially appeared, as shown in Table 4.3. The Kestrel 2.22 also needed to be imported from the US and would have a significant lead time, as mentioned above.

**Micropilot 2028g**

The Micropilot 2028g (Figure 4.3) is a cost effective, compact autopilot. The autopilot is designed to minimise weight and size. Like the other autopilots considered, the 2028g is functionally sufficient for this project and includes all the necessary sensors. While the 2028g does not include a method of dead reckoning for GPS loss, an external electronic compass module is available which provides this functionality.
Micropilot offered a 40% educational discount for this project, which made this the most cost effective autopilot solution. Also, Micropilot is based in Canada, and quoted an export time of only 2-4 weeks, which is significantly lower than those of the US autopilots. The 2028g has been used by many reputable companies including NASA, INTA and BlueBird Aero systems (Micropilot, 2007).

**AP04 UAV**

UAV Navigation emphasise the redundancy and safety of the AP04 autopilot (Figure 4.4). The system includes a failsafe redundant processor and integrated sensor failure tolerance. This effectively means the autopilot can continue to operate after any singular component fails. The AP04 is a very complete system including a radio link with a specified range of 100km and a 3 axis magnetometer. The AP04 also includes 16 user definable servo or input/output lines.

![Figure 4.4: AP04 (UAVNavigation, 2007)](image)

UAV Navigation is a Spanish based company; however an Australian based distributor was located. There was discussion with the Australian distributor, Air Affairs, and UAV Navigation with regards to an educational discount or sponsorship. While the companies were very positive, no formal arrangement was able to be ascertained in time for the procurement of this autopilot to become a real possibility.

### 4.2.3 Autopilot Selection

The Micropilot 2028g was selected as the control system for this project. From the short-list of four, this was the only autopilot system capable of satisfying all the project constraints of cost, import time, and functionality. However, the autopilot did also rate favourably in terms of weight and size, although as it is not a complete system in itself, communications equipment were required to be purchased separately. Table 4.4 shows the selection of components that were purchased, and the additional units that were considered.
4.2. CONTROL SYSTEM SPECIFICATIONS

Table 4.4: Micropilot 2028g System

<table>
<thead>
<tr>
<th>Purchased Components</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>2028g</td>
<td>Contains CPU, 3 axis gyroscopes/accelerometers, GPS, static and</td>
</tr>
<tr>
<td></td>
<td>dynamic pressure transducers</td>
</tr>
<tr>
<td>Ground Station Software</td>
<td>GUI for flight programming and in-flight monitoring and control</td>
</tr>
<tr>
<td>Compass Module</td>
<td>Allows additional measurement of heading and provides for dead</td>
</tr>
<tr>
<td></td>
<td>reckoning in the case of GPS loss</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Additional Components Considered</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>UAS</td>
<td>Ultrasonic Altitude Sensor, required for autonomous conventional</td>
</tr>
<tr>
<td></td>
<td>take-off and landing</td>
</tr>
<tr>
<td>GPS upgrade</td>
<td>GPS upgrade. Allows for differential GPS, in addition to being a</td>
</tr>
<tr>
<td></td>
<td>more accurate GPS unit</td>
</tr>
</tbody>
</table>

Table 4.5 details the specifications of the autopilot in regard to the chosen selection criteria. It needs to be noted that with the Micropilot autopilot, communications equipment and batteries were not included, which needed to be purchased in addition to the value shown below. Also, the addition of importing and freight fees is included.

Table 4.5: Autopilot Selection

<table>
<thead>
<tr>
<th>Parameter</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Cost</td>
<td>$6260</td>
</tr>
<tr>
<td>Lead Time</td>
<td>2-4 weeks export time quoted, actual lead time was 7 weeks from</td>
</tr>
<tr>
<td></td>
<td>beginning of final negotiations</td>
</tr>
<tr>
<td>Weight</td>
<td>28g</td>
</tr>
<tr>
<td>Size</td>
<td>10 x 40 x 15 mm</td>
</tr>
<tr>
<td>Power Consumption</td>
<td>140mA @ 6.5V or equivalent</td>
</tr>
<tr>
<td>Functional Abilities</td>
<td>Capable of fully automating a flight path, including take-off and</td>
</tr>
<tr>
<td></td>
<td>landing. Can control additional outputs, but only using servo</td>
</tr>
<tr>
<td></td>
<td>signals. Internal calculations can be implemented based on sensor</td>
</tr>
<tr>
<td></td>
<td>readings, with up to 30 additional PID loops available for use.</td>
</tr>
<tr>
<td>In-flight Monitoring</td>
<td>Custom software is provided which allows for both in-flight</td>
</tr>
<tr>
<td></td>
<td>monitoring of the aircraft as well as in-flight modifications to</td>
</tr>
<tr>
<td></td>
<td>the flight plan.</td>
</tr>
</tbody>
</table>
Chapter 5

Imaging Systems Conceptual Design

The UAV requires an onboard imaging capability to provide a versatile platform for civil applications. In order to ascertain the requirements for this airborne surveillance system the ARCAA UAV Outback Challenge operational requirements were considered (ARCAA, 2007). Suitable reconnaissance systems were then investigated in order to select a system. This investigation comprised of an examination of analogue, digital and airborne cameras, and an effective means for video transmission to the ground station. Creating a system with some degree of autonomous searching capabilities was considered early in the design phase, but was disregarded due to the level of difficulty and hence long development time that would be associated. Hence, any automation was considered beyond the scope of this project. This section outlines the analysis carried out, the systems investigated, and the overall imaging system selection.

5.1 Operational Requirements

The operational requirements of the UAV Outback challenge were used during the feasibility study to determine a required cruise velocity of 90 km/h, this was specified with the intent of a search area width of 120m. These parameters were specified in order to cover the required search area in the UAV Outback Challenge within a time of one hour.

Field of View (FOV)

The field of view is the angle viewed through the horizontal plane. This angle represents the height and width of the image captured through the lens, for example, humans experience close to a 180 degree field of view (Denoyer & Johnson, 2001).

Since the flight altitude, h, and search area width are known, the required FOV, \( \beta \), can then be calculated using basic trigonometry. These calculations also generate the search area length, \( L \), which is used later in conjunction with the search area width in the analysis of camera resolutions. Consideration of the flight altitude allows an equation for the search area length in relation to the FOV to be obtained. The vertical FOV was first considered, as shown in Figure 5.1, and hence leads to the derivation of equation 5.1.
The derivation due to Figure 5.1:

\[ L = 2h \tan \left( \frac{\beta}{2} \right) \]  

(5.1)

As this equation contains two unknowns, the search area length and FOV, a second equation was derived to develop the required equation. The search area width was considered throughout the horizontal FOV, leading to the derivation of equation 5.2 for search area length with respect to FOV. The two equations were then solved simultaneously to find the required FOV and search area length. Figure 5.2 shows the trigonometric geometry of the horizontal FOV.

\[ L = \frac{120}{\tan \left( \frac{\beta}{2} \right)} \]  

(5.2)

Hence, solving simultaneously lead to the calculation of the FOV and search area length, with \( \beta \) being 68.4 degrees, and L being 176.64 metres. As was previously stated, the desired parameters included the FOV, search area width and flight altitude, which are now defined by the required FOV. These values were identified as the required specifications for the camera selection, in order to maximise the imaging systems potential.

### 5.2 Camera Class

A brief description of the main forms of imaging systems considered in this report follows; analogue imaging systems, digital imaging systems, and UAV and airborne imaging systems.
The camera arrangement on the UAV is one of the factors to be taken into account for the image processing. The UAV will possess a single onboard camera. Since autonomy of the imaging system was not considered, the chosen camera will provide live, continuous streaming of video or other imagery that will be sent to the ground station for manual analysis of the image. This image received will be used to identify the target and deliver a package to the target. A diagram of the imaging system is shown in Figure 5.3.

![Diagram of Imaging Component Layout](image)

Figure 5.3: Imaging Component Layout

### 5.2.1 Analogue Cameras

Analogue camera systems have been the common choice in previous years and still continue to be heavily used in present day applications. In particular, most industrial machinery cameras are analogue due to their price and ease of use (Zuech, 2007).

Analogue cameras operate by monitoring continuous waveforms and conditions, converting them into an electronic response (Robin, 2007). These electronic responses are stored as voltage fluctuations, which are later converted into electronic signals then transmitted to the receiving device. This receiver then interprets the electronic signals and converts them accordingly into audio or video (Robin, 2007).

The resolution of analogue imaging is in the form of TV lines and effective pixels. TV lines denote the number of horizontal black and white lines that may be distinguished in an image that is the same height and width (Gray, 2001). These TV lines may be resolved into lines of horizontal resolution (LoHR), and lines of vertical resolution (LoVR), where each resolution is used in order to be able to distinguish the other. Lines of resolution are not to be confused with pixels, where pixels are the picture elements and considered the smallest information element of an image (Gray, 2001).

### 5.2.2 Digital Cameras

Digital cameras are becoming more prevalent for industrial purposes. In applications where high definition images are required, the high-resolution imaging provided by digital cameras is
beneficial (Zuech, 2007).

Whilst analogue cameras have a resolution with relation to TV lines, digital cameras use pixels as their form of resolution. Each separate pixel holds a certain number of bits of information, dependent on whether the image is grey scale or colour. A grey scale image can hold between two to eight bits per pixel, whereas a colour image may hold between 2 to 24 bits. This allows for the calculation in the file size of the image, through the multiplication of pixels by the number of bits they contain.

Digital cameras convert the same waveform seen by analogue cameras into digital data using an analogue to digital converter (Robin, 2007). Instead of recording a continuous waveform like analogue cameras, digital cameras sample the waveform and record frames. Motion of the video is then viewed through rapid display of the frames recorded. This digital data consists of 1’s and 0’s, with each pixel in the image holding a data package of eight bits. The ninth bit contained in the package is a parity bit, which tells the receiving unit whether or not it has received the correct code from the transmitter (Robin, 2007). Whereas analogue video needs to be interpreted, digital video does not.

5.2.3 UAV and Airborne Cameras

Difficulties in air surveillance may be created due to airframe vibration. Vibration transferred to the camera through the fuselage can be determined through the use of basic trigonometry (Denoyer & Johnson, 2001). Since the UAV will be flying at an altitude of 130m, the vibration in the image due to an angle of 0.5 degrees, for example, is 1.13m. These problems however, may be alleviated through the use of specially designed airborne cameras which utilize the benefits of gimbaled systems (Space-Electronics, ND), image restoration systems (Likhterov & Kopeika, 2002), or vibration isolation systems (McHugh, ND).

5.3 Defining Characteristics

Frame Rate

The frame rate of the camera specifies the number of images captured within a period of time, generally one second. Motion blur could be created in the circumstances that the frame rate of the camera is unable to cope with the velocity of the UAV. Considering a cruise speed of 90 km/h, a camera with a frame rate of 25 frames per second causes a length in image blur of 1 m. The frame rate of the camera must be considered in order to minimize the amount of image blur produced during flight (McHugh, ND).

Electronic Shutter Speed

A shutter is a device used to allow a pre-determined amount of light to enter a camera lens during operation. The shutter speed is the time for which the shutter is held open, to allow the
light to enter the camera and reach the film or the image sensor. Controlled with the help of an electronic timer, the shutter speed can have an impact on imaging moving objects as an increase in exposure time will increase the amount of motion blur (Zettl, 2005). Since the images will be taken during flight, all captured images are of moving objects. Therefore, the shutter speed is to be taken into account.

**Flickerless Mode**

Flickerless mode is a feature that can improve the compression of useless video information. Flicker is the result of the differences between the frequencies (60 Hz) of the ionisation of the gas in a fluorescent light fixture with that of the vertical frequency (59.94 Hz) in a colour camera. The flickerless mode of the camera eliminates this flicker. The result is reduced file sizes and transfer bit rates of the compressed video image size.

**Backlight Compensation**

Backlight compensation is the video gain done either manually or automatically to correct the exposure of subjects that are in front of a bright light source. This feature is necessary to reduce the level of brightness when an object is exposed in front of a bright light source.

**Auto Gain Control**

The auto gain control of a typical video camera measures the brightness of the scene and computes an appropriate exposure. This is necessary in order to allow the necessary amount of brightness to be exposed when the video is taken. The gain limiter is also a feature set in order for the automatic gain control to limit itself to a particular gain value.

**Autowhite balance**

When a white object is illuminated with a low colour temperature light, the colour of the object appears reddish while with a high colour temperature light it appears bluish. Therefore it is necessary to compensate the colour difference caused by the light sources so that a white object appears as white under any lighting conditions. Thus, the Autowhite balance feature is used to compensate the colour difference.

**Signal to Noise Ratio**

Noise is an unwanted signal generated during the capture of an image and during transmission. In video systems this usually refers to random or thermal noise of a type that produces a moving speckled effect in the picture. Noise is quantified using the ratio between the maximum signal level and the noise level. During transmission of the images the noise produced in the background will increase and cause the image to blur (McHugh, ND).
5.3. DEFINING CHARACTERISTICS

Pixel Sizes

As the FOV of the camera and relative visible area are known, the pixel sizes for corresponding resolutions may be obtained. Tables 5.1 and 5.2 provide a list of resolutions available in the market with relation to the image size produced from the camera, and subsequent pixel sizes. Table 5.1 shows the effective pixel sizes produced via an analogue camera. Pixel sizes produced via a digital camera may be found in Table 5.2.

Table 5.1: Analogue Resolution to Corresponding Effective Pixel Sizes

<table>
<thead>
<tr>
<th>Horizontal Resolution (TV Lines)</th>
<th>Vertical Resolution (TV Lines)</th>
<th>Image Width (m)</th>
<th>Image Height (m)</th>
<th>Horizontal Pixel Size (m)</th>
<th>Vertical Pixel Size (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>240</td>
<td>320</td>
<td>120</td>
<td>176.64</td>
<td>0.500</td>
<td>0.552</td>
</tr>
<tr>
<td>380</td>
<td>380</td>
<td>120</td>
<td>176.64</td>
<td>0.316</td>
<td>0.465</td>
</tr>
<tr>
<td>420</td>
<td>420</td>
<td>120</td>
<td>176.64</td>
<td>0.286</td>
<td>0.421</td>
</tr>
<tr>
<td>628</td>
<td>582</td>
<td>120</td>
<td>176.64</td>
<td>0.191</td>
<td>0.304</td>
</tr>
<tr>
<td>640</td>
<td>480</td>
<td>120</td>
<td>176.64</td>
<td>0.188</td>
<td>0.368</td>
</tr>
<tr>
<td>1280</td>
<td>1024</td>
<td>120</td>
<td>176.64</td>
<td>0.094</td>
<td>0.173</td>
</tr>
</tbody>
</table>

Table 5.2: Digital Resolution to Corresponding Pixel Sizes Size(Mp)

<table>
<thead>
<tr>
<th>Size (Mp)</th>
<th>Horizontal Resolution</th>
<th>Vertical Resolution</th>
<th>Image width (m)</th>
<th>Image Height (m)</th>
<th>Horizontal Pixel Size (m)</th>
<th>Vertical Pixel Size (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.3</td>
<td>1280</td>
<td>1024</td>
<td>120</td>
<td>176.64</td>
<td>0.094</td>
<td>0.173</td>
</tr>
<tr>
<td>2</td>
<td>1600</td>
<td>1200</td>
<td>120</td>
<td>176.64</td>
<td>0.075</td>
<td>0.147</td>
</tr>
<tr>
<td>3</td>
<td>2048</td>
<td>1536</td>
<td>120</td>
<td>176.64</td>
<td>0.059</td>
<td>0.115</td>
</tr>
<tr>
<td>4</td>
<td>2304</td>
<td>1728</td>
<td>120</td>
<td>176.64</td>
<td>0.052</td>
<td>0.102</td>
</tr>
<tr>
<td>5</td>
<td>2560</td>
<td>2048</td>
<td>120</td>
<td>176.64</td>
<td>0.047</td>
<td>0.086</td>
</tr>
<tr>
<td>6</td>
<td>2848</td>
<td>2136</td>
<td>120</td>
<td>176.64</td>
<td>0.042</td>
<td>0.083</td>
</tr>
<tr>
<td>7</td>
<td>3072</td>
<td>2304</td>
<td>120</td>
<td>176.64</td>
<td>0.039</td>
<td>0.077</td>
</tr>
<tr>
<td>8</td>
<td>3264</td>
<td>2448</td>
<td>120</td>
<td>176.64</td>
<td>0.037</td>
<td>0.072</td>
</tr>
</tbody>
</table>

It should be noted that some variation occurs between different models of camera. However, since the variation is minor, the values used provide a good estimate of the pixel sizes in the image for any feasible camera.

In order to distinguish a human on the ground from the desired altitude, the pixel sizes of the image must be small enough to allow for a distinguishable, crisp image and minimal blurring. Under normal surveillance circumstances, the largest dimension of a human is the shoulder width, approximately 50cm. Hence the imaging system must be sensitive in order to differentiate between a human and the environmental surroundings. The pixel sizes of the image must therefore be relatively small. Using the aforementioned tables the minimal desired resolution is an 8 mega pixel camera with a 3264x2448 resolution. However, resolution of the imaging system is a specification that was highly dependent on project budget, availability of the desired cameras, and ease of image transmission.
5.4 Comparison and Selection

In order to select a suitable imaging system, considered separately were the criteria for the camera selection, the communications equipment, and for the overall imaging system. To select an adequate camera system, the setting of parameters such as the weight, field of view, and flight altitude were crucial. Similarly, the communications equipment required parameters such as the weight and overall transmission range of the streaming video. The range of transmission was required to be 10km as stated previously in Section 2.6.6. Selection of the overall imaging system was therefore largely dependent on the weight of the system and constrained to fit within the proposed budget. Limitations on the weight of the imaging system may be found in Section 2.6.5, however all other parameters including the altitude, search area width and FOV were to be determined.

5.4.1 Digital & Analogue Camera Comparison

As the main forms of imaging systems involve digital or analogue image processing, a comparison between the two forms was required, along with a comparison to infrared imaging. The main factors distinguishing analogue and digital systems apart is the difference in resolution and the form of data storage and transmission.

<table>
<thead>
<tr>
<th>FEATURE</th>
<th>DIGITAL CAMERA</th>
<th>ANALOGUE CAMERA</th>
<th>INFRARED CAMERAS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cost</td>
<td>High</td>
<td>Low to Medium</td>
<td>Very High</td>
</tr>
<tr>
<td>Resolution</td>
<td>High</td>
<td>Low to Medium</td>
<td>High</td>
</tr>
<tr>
<td>Required Baud rate</td>
<td>High</td>
<td>Low</td>
<td>High</td>
</tr>
<tr>
<td>Long Range Transmission</td>
<td>Hard</td>
<td>Possible</td>
<td>Hard</td>
</tr>
<tr>
<td>Image Enhancement</td>
<td>Possible</td>
<td>Difficult</td>
<td>Possible</td>
</tr>
<tr>
<td>Extraction of Details</td>
<td>Easier</td>
<td>Hard</td>
<td>Easier</td>
</tr>
<tr>
<td>Error Correction</td>
<td>Yes</td>
<td>No</td>
<td>Yes</td>
</tr>
<tr>
<td>Generation Loss</td>
<td>No</td>
<td>Yes</td>
<td>No</td>
</tr>
<tr>
<td>Day Lighting and Night Vision</td>
<td>Good</td>
<td>Poor</td>
<td>Excellent</td>
</tr>
<tr>
<td>Manipulation on Computer</td>
<td>Possible</td>
<td>Not Possible</td>
<td>Possible</td>
</tr>
<tr>
<td>Colour and Brightness</td>
<td>Adjustable</td>
<td>Not Adjustable</td>
<td>Adjustable</td>
</tr>
<tr>
<td>Digital to Analogue converter and vice versa</td>
<td>Required for transmission</td>
<td>Not Required</td>
<td>Required for transmission</td>
</tr>
<tr>
<td>Video Capture Software</td>
<td>Required</td>
<td>Not Required</td>
<td>Required</td>
</tr>
<tr>
<td>Degradation of Images on storage</td>
<td>No</td>
<td>Yes</td>
<td>No</td>
</tr>
<tr>
<td>Storage</td>
<td>Uses Bits</td>
<td>Electrical signals into a Physical Medium</td>
<td>Uses Bits</td>
</tr>
</tbody>
</table>

Table 5.3: Digital, Analogue and Infrared Camera Comparison
Advantages and Disadvantages of Analogue

An initial analysis was performed and generalised comparisons made between the three systems under consideration. From this analysis a comparative table was produced, as is found in Table 5.3.

Considering the Infra-red option, some substantially different capabilities of the system should also be noted. In particular, use of the infra-red system would reduce many of the environmental factors that need to be considered in terms of the digital and analogue systems. As the operation of the infra-red system is dependent on its higher thermal sensitivity, flare due to bright lighting, and distance has a lower impact on the quality of the image received. The used of a thermal system would also allow for a simpler automation of the system in future work especially if coupled with a secondary system for visual verification, although this has already been stated as beyond the scope of this design.

A point of note relating to the analogue system is the potential for interference to occur in the image due to the surrounding electronic componentry Lutha & Inglis (1999). Especially considering the limitations of on-board space, such interference would be hard to suppress if evident.

Result of Comparison

From Table 5.3, it can be concluded that analogue imaging is the most cost effective and user friendly system of the three alternatives, although it is not the most suited to the task. A digital camera provides a promising camera system, however, it does suffer from the difficulty of image transfer, which is discussed further in the following section. The major drawback of Infrared imaging is the expense of such a system which unfortunately rules it out as a viable option, even though it has some unique advantages for this application. Hence further testing of the two remaining systems was required in order to determine the most beneficial unit.

In order to analyse the effectiveness of each form of imaging system for the application of the UAV a comparison was carried out between a 4MP digital camera and a low-resolution analogue lipstick camera borrowed from the School of Computer Sciences at The University of Adelaide. To simulate the search for a human based at the given altitude of 130 metres, a person was photographed from a distance equal to the aircraft’s altitude. The digital image captured was then analysed and the clarity of the person at that distance evaluated. The images obtained can be found in Figures 5.4 (a), (b) and (c).
CHAPTER 5. IMAGING SYSTEMS CONCEPTUAL DESIGN

The testing showed that the digital image obtained was much clearer. However, the image was taken with the camera in a stationary position. From Figure 5.4 (a), it can be seen that the target was not recognizable without any zoom from the camera. Figure 5.4 (b) and Figure 5.4 (c) show the target focused at different resolutions. This testing provided a general idea of the distance and image of the target that has to be recognized.

Testing was also carried out with an analogue lipstick camera. The components shown in Figure 5.5 were borrowed for the tests, including the transmitter, receiver, lipstick camera and power
adapter. Figure 5.5 (b) shows the lipstick camera used. This camera was connected to a television in order to view the quality of the image. The image was clear for a short distance of 10 metres. However, it would not be possible to view a person from a height of 130 metres with the lipstick camera. Thus, a high-resolution camera would be required for imaging.

![Image of camera components](image1)

![Lipstick camera close up](image2)

Figure 5.5: Analogue Camera and its Components

**Image Transmission**

The other parameter for the imaging system is the transmission of streaming video to a ground station, selection of an imaging system was based primarily around the systems transmission capabilities. Real time video was to be streamed across a 10km range, and hence an appropriate transmission system was needed. However, the capabilities of such a system are directly related to the specifications of the chosen camera. Hence, a comparison was made between a 4MP digital camera and a low resolution analogue camera in order to determine the most suitable type of imaging system.

Using the data from the resolution to pixel sizes tables, found in Tables 5.1 and 5.2, Table 5.4 outlines the comparison between the digital and analogue camera.

<table>
<thead>
<tr>
<th>Camera Type</th>
<th>Horizontal Resolution</th>
<th>Vertical Resolution</th>
<th>Image width (m)</th>
<th>Image Height (m)</th>
<th>Horizontal Pixel Size (m)</th>
<th>Vertical Pixel Size (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Analogue</td>
<td>240</td>
<td>320</td>
<td>120</td>
<td>176.64</td>
<td>0.5</td>
<td>0.552</td>
</tr>
<tr>
<td>4MP Digital</td>
<td>2304</td>
<td>1728</td>
<td>120</td>
<td>176.64</td>
<td>0.052</td>
<td>0.102</td>
</tr>
</tbody>
</table>

Considering a 4 mega pixel digital camera, the file size of a single image could be found. In order to determine the approximate file size of this image the amount of information stored on each pixel must be known. For example, if the image were a grayscale image each pixel would contain 3 bits, whereas for a 24-bit colour image each pixel would contain 9 bits. Due to the application of the UAV colour imaging is highly desirable to resolve a person in the outback. Upon consideration of a single 24-bit colour image, and using the knowledge that each pixel hold 9 bits of data, the approximate file size was determined as 36Mb. Similarly, a grayscale image
would give an approximate file size of 12Mb. Both files are relatively large in size, however if grayscale imaging were utilized, the time for transmission of a single image could be found.

An available modem stocked by RF Innovations provides a transmission baud rate of 115 kilobits per second. Thus, dividing the image size through by the baud rate gives the required time for such an image to transmit. For this particular image, the theoretical time for transmission would be 14 minutes and 40 seconds. However, through the use of JPEG compression the files size can decrease to 10% of its original value (Bockaert, 1998). Utilizing this compression technique, the file size for a grayscale image then becomes 1.2 Mb, giving a theoretical time for transmission of 1 minute and 50 seconds. In order to evaluated whether this time for transmission was feasible, the velocity of the aircraft was taken into account. Since the UAV would be traveling at speeds above 65km/h, equating to approximately 18m/s, the imaging system would be capturing individual images 1986 metres apart and hence would not provide a complete coverage of the search area.

**Result of Comparison**

The comparison between digital and analogue technologies highlighted one of the major problems of streaming image optimisation, that is; the trade-off between image quality and transmission time. Although the higher quality of a digital system was clearly desirable in terms of the higher image resolution that could be obtained, the concept of a high-resolution digital camera was deemed unfeasible due to the limitations in image transmission. In contrast, an analogue system is capable of real time, long-range image transmission, however to achieve this, a reduction in image quality occurs. From this analysis it was therefore determined that a suitably high resolution analogue camera with a FOV of 68.4 degrees was suited to this application.

### 5.4.2 Available Options

Table 5.5 provides details of a number of commercially available analogue cameras, including information on some associated video downlink options and prices. It should be noted that all downlink systems considered are capable of transmission over 10km. As the budget limitations of the project allow funds of only $2200 for imaging systems, it is important to consider the system as a whole, in order to suitably distribute the funds. It can be seen from Table 5.5 that the downlink systems provided from RF Links are above the given budget. Hence, Wireless Video Cameras provides the most suitable downlink system. This system may either be used in conjunction with the camera provided with the system, or purchased minus the camera and supplemented with a camera from Future Hobbies or Hi Cam. While Future Hobbies supplies cameras of a higher resolution, lower current draw and at a lower cost to the camera supplied by Wireless Video Cameras, the Wireless Video camera was selected as it was already integrated with the supplied transmission system.
### Table 5.5: Camera and Downlink Options

<table>
<thead>
<tr>
<th>Company</th>
<th>Component</th>
<th>Price</th>
<th>Weight (g)</th>
<th>Voltage</th>
<th>Current</th>
<th>Resolution</th>
<th>FOV (deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wireless Video Cameras</td>
<td>Transmitter/Receiver</td>
<td>$1,407.8</td>
<td>113</td>
<td>12VDC</td>
<td>600mA</td>
<td>NA</td>
<td>NA</td>
</tr>
<tr>
<td>Hi Cam</td>
<td>Camera</td>
<td>$202.9</td>
<td>70</td>
<td>12VDC</td>
<td>130mA</td>
<td>450TVL</td>
<td>90</td>
</tr>
<tr>
<td>Hi Cam</td>
<td>Camera</td>
<td>$202.9</td>
<td>29.7</td>
<td>5.6VDC</td>
<td>160mA</td>
<td>380TVL</td>
<td>73</td>
</tr>
<tr>
<td>Procerus Technologies</td>
<td>Gimbaled Camera</td>
<td>$3381.45</td>
<td>76</td>
<td>5VDC</td>
<td>160mA</td>
<td>450TVL</td>
<td>360 pan, 80 tilt</td>
</tr>
<tr>
<td>Kappa</td>
<td>Camera CSS 90 WA</td>
<td>$12,295.84</td>
<td>350</td>
<td>9-36VDC</td>
<td>3W</td>
<td>480TVL</td>
<td>90</td>
</tr>
<tr>
<td>Future Hobbies</td>
<td>Camera: Uncased</td>
<td>$149.95</td>
<td>12</td>
<td>12VDC</td>
<td>120mA</td>
<td>520TVL</td>
<td>Unknown</td>
</tr>
<tr>
<td>Future Hobbies</td>
<td>Camera: Cased</td>
<td>$173.94</td>
<td>62.4</td>
<td>12VDC</td>
<td>120mA</td>
<td>520TVL</td>
<td>Unknown</td>
</tr>
<tr>
<td>RF Links</td>
<td>MX-6000 Transmitter</td>
<td>$2,808.65</td>
<td>5</td>
<td>6.5VDC</td>
<td>430mA</td>
<td>NA</td>
<td>NA</td>
</tr>
<tr>
<td>RF Links</td>
<td>VRX-24 L Receiver</td>
<td>NA</td>
<td>12VDC</td>
<td>300mA</td>
<td>NA</td>
<td>NA</td>
<td>NA</td>
</tr>
<tr>
<td>RF Links</td>
<td>AMP-18 Amplifier for Receiver</td>
<td>NA</td>
<td>12VDC</td>
<td>30mA</td>
<td>NA</td>
<td>NA</td>
<td>NA</td>
</tr>
</tbody>
</table>
### 5.5 Imaging Systems Selection

The image from the UAV was to be captured using a high resolution analogue camera from WirelessVideoCameras (2005). The camera selected weighs 70gms and is compact in size. The power supply required for the camera is 180mA at 12VDC. The high resolution camera is a colour CCD with 450 TV lines and is shown in Figure 5.6. The additional features of the camera comprise of various adjustment modes. These are the flickerless mode, back light compensation, electronic shutter speed, auto gain control, auto white balance and gain limiter.
Wireless Video Cameras also provided the most suitable video downlink system, as discussed in Section 6.3, which will be used for the transmission of imagery.
Chapter 6

Onboard and Ground Station Communications Hardware

Communication equipment plays a vital role in the transmission of data between the UAV and the ground station. There are three different communication devices installed on the aircraft that are utilized for various purposes. These are listed below.

1. The Autopilot RF Modem
2. The RC Controller and
3. The Video downlink.

6.1 Autopilot RF Modem

The autopilot modem effectively defines the operational radius of the UAV, as the plane is never intended to fly beyond its communication range. This identifies the need for a modem pair capable of transmission over a distance of at least 10 km, as stated in the Feasibility Study. RF Modem Microhard’s MHX-2400 (Figure 6.1) extended range 2.4GHz modems were chosen for the transmission of information relating to the autopilot, selected primarily because of their long range capabilities and for ease of integration with the autopilot. The modems have a specified range of 30 km for transmission over clear line of sight with elevated modems and were recommended by Micropilot as tried and tested systems, compatible with the 2028g autopilot.

Figure 6.1: Microhard MHX-2400 RF Modem (Microhard, 2006)
6.2 RC Controller

The MHX-2400 is a RF Modem transceiver with frequency hopping capability. This modem uses the Frequency hopping spread spectrum (FHSS) and is operating in the 2.400 to 2.4835 GHz ISM band. The modem 'hops' from one frequency to another after a predetermined time interval and the dedicated transceiver at the other end responds to the signals sent or received from the corresponding transceiver. The modem at the ground station is set up as the Master and the modem on-board the UAV is set up as the Slave which synchronizes with the Master modem. The transceiver has a baud rate of 115200 bps and has a power consumption of 550mA and 5 VDC and typically at 210mA and 4.9-5.5 VDC during Rx mode. The onboard modem is relatively lightweight, weighing about 75 grams.

The transceiver operates on the license free band which enables the general public to utilize it. The transceiver provides long range, reliability and sufficient data transfer rate. The modem also possess a built in Cyclic Redundancy Checking (CRC-16) error detection device which re-transmits data in order to achieve 100% accuracy and reliable data transfer. This prevents the data packets from being lost or corrupted during data transmission.

Effective communication between the master and the slave depends on three main factors which are System gain, Path loss and Interference. The transmitter power, transmitter gain, receiver gain and receiver sensitivity are the factors that contribute to the system gain. Path loss and Interference depends on several factors including the terrain profile and the height of the antenna from the ground. The antenna chosen is a unity gain Omni-directional antenna.

The Effective Isotropic Radiated Power (EIRP) from the transceiver is 1 Watt and is operated on a frequency hopping spectrum of the pattern number 44-46 operating on a frequency spectrum of 2.4492-2.4792 GHz is utilized for this application. According to the Radio communications (Low Interference Potential Devices) Class License 2000, Item 54, frequency hopping transmitters of 2400-2483.5 MHz are limited to a maximum EIRP of 4 Watts with a minimum of 75 hopping frequencies. The transceiver chosen thus falls within the legal bounds as specified.

6.2 RC Controller

The two main functional requirements for the RC controller were the need for at least a six-channel system, and full range capabilities, which implies that range is beyond visual recognition in normal conditions (Berg, 2002). A JR XP-6102 RC transmitter (Figure 6.2) and receiver set have been chosen for short range RC control. This is a six-channel system that emits radio waves at 36 MHz. This was selected as it satisfies the requirement of a full range transmitter with six channel functionality and was recommended as a reliable system by ModelFlight (2007).

The required six channels were Ailerons, Elevators, Rudder, Throttle, Parachute system and switching from RC to autopilot. The RC controller is set to airplane mode in the transmitter. The RC controller can transmit either by Pulse Code modulation (PCM) or Pulse Position Modulation (PPM). The modulation type chosen for this application is PCM. PCM uses digital technology whereas PPM utilizes analogue technology and has the ability of error checking and fail-safe procedures. PCM will cause the servos to hold their last position until the receiver can recognize the signal again when a transient interference is produced (i.e. loss of communication).
The PCM connection locks out when connection is failed. The PCM returns to normal function as soon as the radio signal is received again thus establishing the connection. PPM causes undesirable movement of servos when a bad signal is decoded (Werner, 2003). This controller met the required specification and was well within project budget purchased from ModelFlight (2007). The entire RC controller was thus setup using the manual provided by the manufacturer.

![RC Transmitter](image)

![RC Receiver](image)

**Figure 6.2:** JR XP-6102 RC equipment (Horizon-Hobby, 2006b)

### 6.3 Video Downlink

This section details the selection of an image transmission system for the imaging systems operations. Communication plays a vital role in the transmission of images from the UAV to the ground station. The quality of these images transmitted directly effects the capability of identifying the target. Various communication devices were investigated in order to provide the desired outcome. The transmitters/receivers were chosen based on certain specifications. The main specifications to be considered were the range, frequency, baud rate and cost. Other spec-
6.3. VIDEO DOWNLINK

Specifications are also outlined in Table 6.1. Analogue video requires a low amount of baud rate to be transmitted due to the low size of the image file as opposed to digital images which require a higher baud rate for transmission. Digital images cannot be transmitted directly, as a Digital to Analogue converter is required onboard the UAV to convert the digital signal to Analogue signal for effective transmission. Further, an analogue to digital converter is required at the base station to generate the digital image. There were a few options (Table 6.1) that were investigated in-order to perform the required objectives as mentioned above.

<table>
<thead>
<tr>
<th>FEATURE</th>
<th>RFI 9256</th>
<th>9XTend RS232 RF Modem</th>
<th>Wireless Video Downlink (AAR15 - B)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Range</td>
<td>30 km point to point</td>
<td>Up to 20 mile (32 km) with high gain antenna</td>
<td>25 miles (40 km)</td>
</tr>
<tr>
<td>Operational Frequency</td>
<td>900 MHz</td>
<td>900 MHz</td>
<td>2.4 GHz</td>
</tr>
<tr>
<td>Baud rate</td>
<td>115200 bps</td>
<td>125000 bps</td>
<td>Not specified</td>
</tr>
<tr>
<td>Power Supply</td>
<td>8 to 30 VDC</td>
<td>7 to 28 VDC</td>
<td>12 VDC</td>
</tr>
<tr>
<td>Power Output</td>
<td>1mW to 1 W</td>
<td>1mV to 1W</td>
<td>1W</td>
</tr>
<tr>
<td>Weight</td>
<td>260 grams</td>
<td>200 grams</td>
<td>113 grams</td>
</tr>
<tr>
<td>Connectors</td>
<td>BNC</td>
<td>RPSMA</td>
<td>RCA jack</td>
</tr>
<tr>
<td>Transmitter Dimension</td>
<td>19cm (L) x 8cm (W) x 3.5cm (H)</td>
<td>6.99cm x 13.97cm x 2.86cm</td>
<td>6.35cm x 7.62cm x 2.54cm</td>
</tr>
<tr>
<td>Operating Environment</td>
<td>-100C to +600C</td>
<td>-400C to +850C</td>
<td>Not Specified</td>
</tr>
<tr>
<td>Cost (AUD)</td>
<td>$1013</td>
<td>$260</td>
<td>$800</td>
</tr>
<tr>
<td>Connecting Interface</td>
<td>Dual RS232</td>
<td>RS232</td>
<td>Not specified</td>
</tr>
</tbody>
</table>

The wireless video downlink was purchased from WirelessVideoCameras (2005) as the camera system was also integrated with the system and enabled to reduce the setup time and other technical difficulties with the integration of other transmitting devices with the camera system.
The transmission of the live video was carried out with a transmitter. The transmitter (Figure 6.3) will be placed onboard the UAV and has a specified range of 25 miles. The transmitter has a high baud rate as it involves the transmission of video signals and hence the frequency type FM Analog. The transmitter has a set frequency of 2.411GHz and has a power supply of 360mA at 12VDC.

![Analogue Video Transmitter](image1)

The video signals is received at the ground station using a receiver. The receiver (Figure 6.4) consists of four different channels and is tuned to channel 1 to receive the video signals. The receiver is supplied power from an adapter which provides a 12VDC power supply.

![Analogue Video Receiver](image2)

The Effective Isotropic Radiated Power (EIRP) by the transmitter is 1 Watt and is operated on a set frequency of 2.411GHz. According to the Radio communications (Low Interference Potential Devices) Class License 2000, Item 26 transmitters operating between 2400-2450 MHz can have a maximum EIRP of 1 Watt when used for telemetry purposes (ACMA, 2006). The transceiver chosen thus falls within the legal bounds as specified.

Transmitters and receivers are to be connected to sufficient antennas for the transmission and reception of the signal. Two different types of antennas are available, directional and omni-directional antennas. These antennas consist of high gain and low gain antennas that will determine the range required for the transmission and reception. Directional antennas provide higher range in a given direction whereas omni-directional antennas are better for area coverage. The propagation of radio wave application is characterized by several factors (Sputnik-Inc, 2004).

- Geometric spreading of the wave front diminishes signal power and this phenomenon is commonly known as free space loss.
- Signal power is attenuated as the wave passes through solid objects such as trees, walls, window and the floors of buildings.
The signal is scattered and can interfere with itself if there are objects in the beam of the transmit antenna even if these objects are not on the direct path between the transmitter and the receiver.

Scattering and attenuation also affects signal transmission. Scattering occurs when RF signals are reflected and the direct signal combines with signals reflected off objects that are not in the direct path (Sputnik-Inc, 2004). Attenuation occurs when the RF signal passes though solid objects with part of the signal power being absorbed. Attenuation depends on the structure of the object through which the signal is passing through (Sputnik-Inc, 2004). These factors are minimal in this application as the UAV will be in the line of sight and there is no reflecting medium in the air.

A high gain directional grid antenna (Figure 6.5) of 24dB gain was utilized for the transmission/reception of video signals. This allows reliable image transmission over a long-range provided line of sight is maintained with the aircraft.

Hence the wireless video camera system was selected with the high gain antenna in order to provide the desired range of transmission and a quality image.
Chapter 7

Airframe Detailed Design

The conceptual design phase of the project specified the required size and geometry of the main wing and tail plane, mass of the aircraft and the type of launch and recovery systems.

The aerodynamic design aimed to ensure that suitable lift, drag, controllability and stability performance could be achieved by the main wing and tail plane. The design determined suitable airfoil profiles for the main wing and tail plane surfaces, as well as the installed angles of these devices. The design work also determined the required sizing and positioning of the control surfaces on the aircraft. A longitudinal stability analysis was also performed during the aerodynamic design work. This analysis allowed the neutral point of the aircraft to be determined. Knowing the location of the neutral point, the centre of gravity of the aircraft, in all configurations, could be positioned correctly relative to this point. This achieved an aircraft design that was inherently stable in the longitudinal direction.

The structural design of the wing and fuselage components required a significant amount of consideration. This process involved the prediction of maximum loading conditions and the analysis of stress in each of the structural components. All of the aforementioned aspects have been conducted with accordance to CASA regulations on structural integrity for a design load factor of 3.8 and a safety factor of 2.25 and 1.5 for composite and metallic materials respectively.

Launch and recovery systems were integrated into the airframe design that has been developed in this chapter. This involved the incorporation of proprietary subsystems to fabricate a suitable system for the car launch and parachute recovery system. All of the aforementioned developments are discussed in the following sections.

7.1 Main Wing Aerodynamic Design

7.1.1 Main Wing Specifications

The following specifications were required to be met during the aerodynamic design work of the main wing.

- Selection of a suitable airfoil to produce desired $C_{L_{\text{max}}}=1.2$ from conceptual design work
7.1. MAIN WING AERODYNAMIC DESIGN

- Determination of the required $\alpha_{\text{installed}}$ to allow aircraft to have horizontal attitude during cruise
- Sizing and positioning of ailerons on main wing to provide adequate roll control authority
- Sizing and positioning of ailerons so they could be easily integrated into the conceptual and detailed structural design of the wing
- Specification of suitable washout to promote main wing to stall at the root first, thus allowing roll control to be maintained during stall

7.1.2 Wing Airfoil

The design of the airfoil section for the wing is critical for ensuring the aircraft can achieve the required lift and drag performance. The shape of the airfoil affects the lift and performance of the aircraft in all flight regimes, including cruise, takeoff and descent (Raymer, 2006).

A custom airfoil could have been designed for the wing. However, given the large number of airfoils currently in use on small scale aircraft and the limited time frame of the project, this option was deemed an inefficient use of time and resources. Instead, an airfoil was selected from a number of profiles potentially suited to the requirements of the UAV.

Detailed Airfoil Specifications

The following requirements and issues had to be considered when selecting the airfoil for the main wing:

- Suitable for operation in flow with $1.0 \times 10^5 < Re < 6.0 \times 10^5$, (expected range for vehicle operation)
- Able to provide the desired maximum wing lift coefficient, $C_{L_{\text{max}}} = 1.2$, from conceptual design phase
- Have the highest possible $\frac{L}{D}_{\text{wing}}$ (from identified airfoils,) to allow aircraft to achieve highest possible $\frac{L}{D}_{\text{vehicle}}$
- Low, constant $C_m$ to reduce torsional loads and induced drag from trimming
- Able to be manufactured

Potential Airfoil Candidates

Six potentially suitable airfoil profiles were identified from a literature review. Four of these are used in commissioned UAVs similar in design and weight to the project vehicle. The other two airfoils are suitable for low flow. The airfoils, and their respective uses are presented in Table 7.1.
Table 7.1: Low Re Airfoils Candidates

<table>
<thead>
<tr>
<th>Airfoil</th>
<th>Designer</th>
<th>UAV in use on</th>
<th>Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Eppler 423</td>
<td>Eppler</td>
<td>Unknown if any</td>
<td>Eppler (1990)</td>
</tr>
<tr>
<td>S1210</td>
<td>Selig</td>
<td>Unknown if any</td>
<td>Selig (2006)</td>
</tr>
<tr>
<td>SD7032</td>
<td>Selig</td>
<td>'Sender' (NRL)</td>
<td>Selig (2006)</td>
</tr>
<tr>
<td>SD7037</td>
<td>Selig</td>
<td>'Aerosonde' (Aerosonde)</td>
<td>Ollie (2001)</td>
</tr>
<tr>
<td>SD7062</td>
<td>Selig</td>
<td>'Stallion' (Oaklahoma State University)</td>
<td>Selig (2006)</td>
</tr>
<tr>
<td>LA2573a</td>
<td>Liebeck</td>
<td>'Flyrt' (NRL)</td>
<td>Lednicer (2007)</td>
</tr>
</tbody>
</table>

Plots of these airfoil shapes can be found in Appendix A.1.

Airfoil Selection Process

The airfoil selection process involved comparing the two dimensional flow performance of the airfoil candidates over the angle of attack range in which the airfoils were not stalling, \((0 \text{deg} < \alpha < 12 \text{deg})\) for most of the candidate airfoils.) Although the two dimensional performance of the airfoils does differ when they are used in a three dimensional wing, an indication of \(\frac{L}{D_{\text{wing}}}\) can be obtained by finding \(\frac{L}{D_{\text{airfoil}}} = \frac{C_l}{C_d}\). Thus, for the purposes of the airfoil comparison, it was assumed that the airfoil with the greatest \(\frac{L}{D_{\text{airfoil}}}\) would produce the wing with the highest \(\frac{L}{D_{\text{wing}}}\). Similarly, the airfoil with the lowest, most constant section pitching moment coefficient, \(C_m\), would produce the wing with the lowest, most constant pitching moment coefficient, \(C_M\).

Three dimensional flow effects cause wings to have lower lift coefficients, than the two dimensional section airfoil lift coefficients which constitute the device. Consequently, since a specified value of \(C_{L_{\text{max}}}\) was given, it would have been a poor selection process to select an airfoil which produced \(C_{l_{\text{max}}}\) value approximately equal to \(C_{L_{\text{max}}}\), without correcting it for three dimensional flow. As a result, the airfoil selection process aimed to select an airfoil with \(C_{l_{\text{max}}}\) slightly larger than \(C_{L_{\text{max}}}\). The \(C_{l_{\text{max}}}\) value would then be corrected to \(C_{L_{\text{max}}}\) using the accepted methods presented by Abbot & VonDoenhoff (1959), and provided that the airfoil could produce \(C_{L_{\text{max}}} > 1.2\), then it would be suitable.

As a result, in addition to being manufacturable, the selected airfoil profile (for the specified range,) was to have:

- High \(\frac{L}{D_{\text{airfoil}}}\)
- Low, constant \(C_m\)
- \(C_{l_{\text{max}}} > 1.2\) such that \(C_{L_{\text{max}}} > 1.2\) after three dimensional correction

Two Dimensional Airfoil Analysis

To compare the performance and suitability of each of the selected airfoils, 'JavaFoil' airfoil analysis software was used. This method was selected for a number of reasons including:
• The ease of obtaining required data
• The ease of comparing different airfoils
• Efficiency compared to other such methods including wind tunnel testing.

The analysis was conducted by inputting the appropriate coordinates of each airfoil shape, which were retrieved from Selig (2006). Although this method relies on many mathematical assumptions and simplifications, it was hypothesized that these approximations in the mathematics would be no more inaccurate than the errors associated with wind tunnel testing, including measurement errors and leakage. As such, it was expected that analysis results obtained from JavaFoil would be approximately equal to more widely accepted wind tunnel test data. To verify this hypothesis, analysis results obtained from JavaFoil for the performance of an airfoil were compared with results from wind tunnel tests for the same airfoil. This comparison was done for the SD7032, SD 7062, and the Eppler 423 airfoils, at various values within the specified range for the main wing.

JavaFoil Validation  The validation of the JavaFoil results with wind tunnel data was performed by comparing the differences in the lift coefficient values of the airfoils at the maximum angle of attack of the respective airfoil, and also the differences in polar plot curve shapes of the airfoils.

The discrepancies in lift coefficient values at maximum airfoil angle of attack were found to fall between 2.7% and 7.8%, (Table 7.2) The polar plots were also found to be approximately equal in terms of shape and values, (see Appendix A.2) Consequently, the discrepancies between the JavaFoil results and wind tunnel data were deemed small enough to verify the validity of the use of JavaFoil for the comparison of the airfoils.

<table>
<thead>
<tr>
<th>Airfoil</th>
<th>Re (×10⁵)</th>
<th>( C_{l_{max}} ) (Wind Tunnel)</th>
<th>( C_{l_{max}} ) (JavaFoil)</th>
<th>% Difference</th>
</tr>
</thead>
<tbody>
<tr>
<td>SD7032</td>
<td>2.5</td>
<td>1.388</td>
<td>1.426</td>
<td>2.74</td>
</tr>
<tr>
<td>SD 7032</td>
<td>5</td>
<td>1.398</td>
<td>1.441</td>
<td>3.08</td>
</tr>
<tr>
<td>SD 7062</td>
<td>2.5</td>
<td>1.4319</td>
<td>1.554</td>
<td>8.53</td>
</tr>
<tr>
<td>SD 7062</td>
<td>5</td>
<td>1.4668</td>
<td>1.581</td>
<td>7.79</td>
</tr>
<tr>
<td>Eppler 423</td>
<td>NA</td>
<td>NA</td>
<td>NA</td>
<td>NA</td>
</tr>
</tbody>
</table>

Results of Two Dimensional Analysis  Polar plots and moment coefficient versus angle of attack plots were produced for each airfoil. These can be found in Appendix A.3.

The first analysis of the results involved the comparison of the average maximum lift coefficients of the airfoils over the Reynolds Number range, as well as the investigation of the magnitude and consistency of the moment coefficient values over the operational angle of attack range. The values of these parameters for the airfoils is presented in Table 7.3.
Table 7.3: Results of Average and Investigations

<table>
<thead>
<tr>
<th>Airfoil</th>
<th>Average $C_{l_{max}}$ (Over Re Range)</th>
<th>Is $C_m$ approximately constant over operational range of $\alpha$</th>
<th>Average Constant $C_m$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Eppler 423</td>
<td>2.52</td>
<td>Yes</td>
<td>-0.27</td>
</tr>
<tr>
<td>S1210</td>
<td>2.11</td>
<td>Yes</td>
<td>-0.25</td>
</tr>
<tr>
<td>SD 7032</td>
<td>1.43</td>
<td>Yes</td>
<td>-0.09</td>
</tr>
<tr>
<td>SD 7037</td>
<td>1.27</td>
<td>Yes</td>
<td>-0.08</td>
</tr>
<tr>
<td>SD 7062</td>
<td>1.81</td>
<td>Yes</td>
<td>-0.09</td>
</tr>
<tr>
<td>LA 2573a</td>
<td>1.44</td>
<td>No</td>
<td>0.005 to 0.02</td>
</tr>
</tbody>
</table>

The LA 2573a differs from the rest of the airfoils by having an inconsistent $C_m$ over the range of $\alpha$ values. Furthermore, this moment is positive, which is undesirable as it produces nose up pitch tendencies which can create vehicle instability. For this reason, the LA 2573a was disregarded.

The Eppler 423 and S 1210 airfoils were disregarded for two reasons. Both profiles produced lift coefficients much larger than 1.2. As such, if they were to be used with the given wing geometry, they would generate significant drag due to the trimming required to maintain level flight with the specified wing geometry. The wing geometry could be changed to accommodate the high lift coefficient values, however this would result in significant induced drag, since

\[ C_{D_{\text{induced}}} \propto C_L^2 \]

(Raymer, 2006)

which is highly undesirable. These profiles also have large moment coefficients, which again would create excessive drag due to the need to trim out the moments generated by these profiles.

After analyzing the polar plots of the SD 7032, SD 7037, and SD 7062, the SD 7032, the SD 7062 was found to have the worst $\frac{C_m}{B_{\text{airfoil}}}$ performance, as well as a much greater lift coefficient value than required. For this reason, the SD 7062 was disregarded.

The SD 7032 and SD 7037 airfoils were identified from the two dimensional analysis to be the most suitable. However, the SD 7032 had the best $\frac{C_m}{B_{\text{airfoil}}}$ performance, whereas the SD 7037 had the better moment coefficient performance, and also potentially less induced drag with its lower maximum lift coefficient. Since both could also be easily manufactured, it was decided to base the final airfoil selection on the results of the three dimensional corrections of the airfoil lift coefficients.

**Three Dimensional Lift Coefficient Correction**  The wing lift coefficient able to be produced by the SD 7032 and SD 7037 airfoils was determined by using the methods presented by Abbot & VonDoenhoff (1959). These methods relate the wing lift coefficient through the airfoil lift coefficient using Equation 7.1.

\[ C_L = \frac{C_l - C_{l_0}}{C_{l_{a1}}} \]  

(7.1)
For the purpose of this analysis, $C_{l_{a1}}$ and $C_{l_{b}}$ were calculated using the results of Abbot & VonDoenhoff (1959), for half-spanwise locations. The airfoil section lift coefficient, $C_l$ at the half-spanwise locations at stall conditions was calculated using the above equation, for different values of $C_L$, and then plotted. An additional curve with the maximum $C_{l_{max}}$ (at stall conditions) values at those half-spanwise locations obtained from JavaFoil, was also plotted. The approximate $C_{l_{max}}$ value for the wing for each airfoil was then determined to be the $C_L$ value, which produced a $C_l$ distribution equal to the $C_{l_{max}}$ plot in a spanwise location. This value of wing lift coefficient corresponds with the onset of stall within these regions, and hence is the theoretical maximum wing lift coefficient. These graphs are shown in Figure 7.1 and 7.2 for the SD 7032 and SD 7037 respectively.

![Determination of Wing Lift Coefficient (SD 7032)](image)

Figure 7.1: Determination of Wing Lift Coefficient (SD 7032)
From these plots, it can be seen that

\[ C_{L_{\text{max}}|SD7032} \approx 1.35 \]

\[ C_{L_{\text{max}}|SD7037} \approx 1.20 \]

It is expected that the actual wing installed on the plane will have a lower lift coefficient than the theoretical value, due to the imperfections in the geometry of the manufactured product. Consequently, the SD 7032 profile was selected, so as a small margin for lift coefficient loss could be tolerated in the final design, (Figure 7.3).
7.1.3 Wing Installed Incidence Angle

For cruise conditions, it is desirable for the aircraft to have a horizontal pitch attitude to minimize the drag of the vehicle. In addition, for the UAV, a horizontal cruise attitude provides optimum camera performance, and payload trajectory.

To achieve the horizontal attitude in cruise, the wing needs to be installed at the correct incidence angle to develop the required lift during cruise. The calculation of this required angle was made using Equation 7.2 given by Abbot & VonDoenhoff (1959).

\[
\alpha_{\text{installed}} = \frac{C_{L_{\text{cruise}}}}{a} + \alpha_{l_{0}} + J\varepsilon
\]  

(7.2)

The following values were used in this equation

\( C_{L_{\text{cruise}}} = 0.517 \)

(from results of conceptual design)

\( \alpha = 0.0846\text{deg}^{-1} \)

(from Abbot & VonDoenhoff (1959) for given wing geometry)

\( \alpha_{l_{0}} = -4\text{deg} \)

(from JavaFoil results)
\[ J = -0.4 \]

(from Abbot & VonDoenhoff (1959) for given wing geometry)

\[ \varepsilon = -2 \text{deg} \]

(wing twist to prevent wing-tip stall - this is discussed further in Section 7.1.4.)

and resulted in

\[ \alpha_{\text{installed}} = 1.72 \text{deg} \]

### 7.1.4 Roll Control Surface Design and Integration

Ailerons are used on the UAV to control the roll motion of the vehicle. The wing aerodynamic design was also responsible for the design and sizing of these control surfaces, as well as the design of the wing to ensure these devices work effectively.

#### Sizing of Ailerons

The sizing of the ailerons used the recommendations of Simons (2002), and Eger (1983). These sources provided recommended chord length values, and areas, relative to the wing itself.

According to Simons (2002), the length of the aileron should be between 20% and 30% of the chord length of the main wing.

\[ 0.20C \leq C_{\text{ailerons}} \leq 0.30C \]

For ease of manufacture and design, the ailerons were to be tapered linearly with the wing. This allowed the ailerons to be specified in terms of a constant, local wing chord length, as well as allowed the surfaces to mount between the lower and upper aft spars of the wing, (discussed further in wing structural design.)

To satisfy all of the requirements, recommendations, and considerations, the chord length of the ailerons was set at 21% of the local chord of the wing.

\[ C_{\text{ailerons}} = 0.21C \]

For ease of hinging, the ailerons were designed to be trapezoidal in plan form. As a result of the taper of the wing, the area of each aileron is given by Equation 7.3.

\[ S_{\text{ailerons}} = \frac{0.205(C_1 + C_2)}{2} b_{\text{ailerons}} = \frac{0.205(2C + \frac{dc}{d(\frac{z}{2})})}{2} b_{\text{ailerons}} \]  \hspace{1cm} (7.3)

where \( C_1 \) and \( C_2 \) are the local chord lengths at the root and tip sides of the aileron respectively, (Figure 7.4)
Eger (1983), suggests that the aileron should occupy approximately 7% of the area of each wing section, (starboard and port)

\[
\frac{2S_{\text{aileron}}}{S} = 0.07
\]

The above equations are difficult to solve analytically to find \( C_1 \) and \( b_{\text{aileron}} \). Instead an interactive approach was used to arrive at the following values.

\[
C_1 = 233\text{mm}
\]

\[
b_{\text{aileron}} = 0.40\left(\frac{b}{2}\right)
\]

and the aileron lies between 55% and 95% of the half span of the wing.

This results in

\[
S_{\text{aileron}} = 1.5 \times 10^{-2}m^2
\]

\[
\frac{2S_{\text{aileron}}}{S} = 0.065
\]

values which were deemed acceptable in terms of controllability as well as manufacturability.

**Aileron Half-spanwise Location Design and Wing Twist**

As mentioned previously, the ailerons are located at 55% and 95% of the half span of the wing. Such a location provides adequate leverage to control the aircraft, with minimal control surface deflection, and hence loading required.

In the event of wing stall, it is desirable to maintain roll control. Hence the ailerons should not be experiencing stall whilst the remainder of the wing is stalling, and consequently main wing stall should commence at the root first, and at the tip last. To achieve this, the wing is twisted such that the angle between the centrelines of the tip and root cross-sections, (the wing twist angle) \( \varepsilon = -2\text{deg} \) as recommended by Raymer (2006) for general aviation aircraft.
7.2 Tail Plane Aerodynamic Design

7.2.1 Tail Plane Specifications

The following specifications were required to be met for the aerodynamic design of the tail plane:

- Selection of a thin symmetrical airfoil profile, capable of operating in flow with $1.0 \times 10^5 < Re < 3.0 \times 10^5$

- Determination of $\alpha_{\text{installed}}$ to allow tail plane to provide effective pitch and yaw control as well as vehicle stall recovery capabilities

- Sizing and positioning of elevators on horizontal stabiliser to provide adequate pitch control authority

- Sizing and positioning of rudder on vertical stabiliser to provide adequate yaw control authority

- Sizing and positioning of elevators and rudder so as they could be easily integrated into the conceptual and detailed structural design of the wing

7.2.2 Tail Plane Airfoil

The primary function of the tail plane is to generate lift forces to create the required lateral and longitudinal moments for vehicle control. The tail plane is not required to contribute to the lift force which supports the weight of the aircraft during flight. Consequently, a symmetrical airfoil was specified so as the lift forces produced by the tail plane would be approximately equal in both directions. A thin airfoil was specified, as thin airfoils produce less form drag than equivalent thick profiles.

Due to project time constraints, an in depth JavaFoil comparison of suitable airfoils for the tail plane was not conducted. It was also deemed unnecessary, since the drag performance of different, thin, low Re, symmetrical airfoils was found to be very similar during the initial literature survey performed for this section.

From the investigation of literature on low drag, low Re, symmetrical airfoils from Selig (2006), and Simons, (1999) it was found that the SD 9032 airfoil, as shown in, Figure 7.5, was a suitable choice. Thus the SD 9032 airfoil was the selected profile for both the vertical and horizontal stabilisers.
7.2. TAIL PLANE AERODYNAMIC DESIGN

7.2.3 Stall Recovery

As specified, the airfoil used for the horizontal stabiliser is required to be able to recover the aircraft from main wing stall. To do this, the airfoil needs to stall later than the main wing, so that pitch control can be maintained and hence allow the pilot / autopilot to pitch the nose of the aircraft down and recover.

The angle of attack of the horizontal stabiliser is affected by the flow over the main wing. This flow has the effect of reducing the effective angle of attack of the horizontal stabiliser due to the downwash over the main wing, and also the installed angle for the device. The effective angle of attack of the horizontal stabiliser is this given by Equation 7.4 from Raymer (2006).

\[
\alpha_{H_{\text{main-wing-stalled}}} = \alpha_{\text{stall}}(1 - \frac{\partial \varepsilon}{\partial \alpha}) + (\alpha_{\text{installed}} - \alpha_{\text{installed}}) + \Delta \alpha_{OL} 
\]

\(\alpha_{\text{stall}} = 10\,\text{deg}\)

(from wing design analysis)

\(\frac{\partial \varepsilon}{\partial \alpha} = 0.585\)

(from Raymer (2006) for specified main wing and tail plane geometry)

\(\Delta \alpha_{OL} = 0\)

(since no flaps used on aircraft)

Figure 7.5: Cross Section of SD 9032 Airfoil as used in Horizontal Stabiliser (Selig, 2006)
Thus, for

\[ \alpha_{\text{installed}} = 0 \text{deg} \]

\[ \alpha_{H, \text{main-wing-stalled}} = 2.43 \text{deg} \]

when the main wing is stalled.

Since

\[ \alpha_{H, \text{stall}} = 8 \text{deg} \]

(from JavaFoil results for SD 9032 airfoil)

the horizontal tail can be installed at an angle of up to 5deg, and will still provide stall recovery benefits. However, if the installed angle is too large, unnecessary drag will be generated, due to the need to trim the control surfaces to maintain level flight.

### 7.2.4 Stabiliser Installed Incidence Angles

The installed incidence angles of both the horizontal and vertical stabilisers were chosen to both be 0 deg.

\[ \alpha_{\text{installed}} = 0 \text{deg} \]

\[ \alpha_{\text{installed}} = 0 \text{deg} \]

By installing the stabilisers with 0 deg incidence, stall recovery benefits could be attained. Furthermore, the surfaces could be installed with relative ease, and all required trimming can be performed by the pilot or autopilot, and adjusted depending on vehicle configuration. This trim method was deemed acceptable by both the engineers and test pilot, and as such the installation angles were also deemed to be acceptable.

For future work, the trim information from the pilot or autopilot should be used to install the stabilisers at angles which contribute positively to the trimming of the aircraft. This would reduce the drag, and improve the efficiency and effectiveness of the tail plane.

### 7.2.5 Pitch Control Surface Design and Integration

Elevators are used on the horizontal stabiliser to control the pitch motion of the aircraft. The sizing of these surfaces used the recommendations of Simons, (2002.)

According to Simons (2002) the chord length of the elevators should be between 20\% and 30\% of the chord length of the horizontal stabiliser.

\[ 0.20C_H \leq C_{\text{elevator}} \leq 0.30C_H \]

For ease of manufacture and design, the elevators were designed to be rectangular. This allowed the elevators to have a constant chord length, which was chosen to be 0.30C_H.

The span of the elevators was selected to be as large as possible. Consequently, the span of the elevators would ultimately be determined during the manufacture of the airframe.
7.2. TAIL PLANE AERODYNAMIC DESIGN

Thus the size of the elevators was selected as:

\[ C_{\text{elevator}} = 0.30C_H = 37.2 \text{mm} \]

\[ b_{\text{elevator}} = \text{as large as possible based on airframe manufacture.} \]

7.2.6 Yaw Control Surface Design and Integration

A rudder located on the vertical stabiliser, is used to control the pitch motion of the aircraft. The sizing of this surface used the recommendations of Simons (2002).

According to Simons (2002), the chord length of the rudder should be between 20% and 30% of the chord length of the vertical stabiliser.

\[ 0.20C_V \leq C_{\text{rudder}} \leq 0.30C_V \]

The rudder was designed to be rectangular for ease of manufacture. This allowed the rudder to have a constant chord length. Again, to ease manufacture, the rudder chord size was increased to be the same length as the elevator chord, \( 0.30C_H \). This was considered acceptable, since the main effect of this size increase was to increase yaw control authority, which was satisfactory.

Like the elevators, the span of the rudder was selected to be as large as possible. Consequently, the span of the rudder would also be ultimately determined during the manufacture of the airframe.

Thus the size of the rudder was selected as:

\[ C_{\text{rudder}} = 0.30C_H = 37.2 \text{mm} \]

\[ b_{\text{rudder}} = \text{as large as possible based on airframe manufacture.} \]

7.2.7 Longitudinal Stability Analysis

A simplified longitudinal stability analysis was performed so that the aircraft could be designed with inherent stability in the longitudinal direction in all flight configurations. The analysis involved the determination of an appropriate static margin for the aircraft, the calculation of the neutral point of the aircraft, and finally the calculation and positioning of the centre of gravity of the UAV for all flight configurations.

Specifications of Longitudinal Stability Analysis

The longitudinal stability analysis was required to satisfy the following specifications:

- Determine the location of the neutral point of the aircraft, \( Xac_A \)

- Configure the aircraft such that a static margin of approximately -15% is achieved for all flight configurations. This value was selected from recommendations from Raymer (2006), and would result in an aircraft which is inherently stable in the longitudinal direction, and adequately responsive and controllable.
Determination of the Neutral Point of the Aircraft

The neutral point of the aircraft is the position along the longitudinal axis of the aircraft, where the aerodynamic moment is constant. By knowing the approximate location of this point, the aircraft could be designed so its centre of gravity lies ahead of this point for all flight configurations, and thus achieve an inherently stable design in the longitudinal direction.

The position of this neutral point is measured in terms of %MAC, measured from the leading edge of the MAC. This point was estimated using the simplified method, (Equation 7.5) presented by Simons, (2002)

\[ X_{acA} = X_{acW} + \eta V_H \left( \frac{a_H}{a} \right) \left( 1 - \frac{\partial \varepsilon}{\partial \alpha} \right) \]  

(7.5)

To solve the above equation, the following approximations were made.

\[ X_{acW} = 25\% MAC \] (Raymer, 2006)

\[ a_H = 0.105 \] (from JavaFoil results for horizontal stabiliser airfoil section)

\[ \eta_s = 0.6 \] (Simons, 2002)

Consequently,

\[ X_{acA} \approx 40.9\% MAC \]

Configuration of Centre of Gravity

To satisfy the stability requirements of the aircraft, the centre of gravity of the vehicle needs to lie ahead of the neutral point. The distance between the centre of gravity and the neutral point is called the static margin (Equation 7.6) and like the neutral point, this parameter is measured in terms of %MAC.

\[ SM = X_{cg} - X_{acA} \]  

(7.6)

The UAV is intended to operate in four different configurations; RC Test, Autopilot Test, Camera Test, and Mission. The payload and battery power requirements for each differ, and consequently, the centre of gravity for each configuration had to be determined, and adjusted until the design SM of the aircraft was approximately -15% for all flight configurations as specified. This determination was performed by making accurate weight measurements of all components of the aircraft in each configuration, and positioning the components which could be moved, (eg payload, and RC receiver,) within the aircraft until the required static margin values were obtained.

The above analysis resulted in the development of the Design Centre of Gravity Envelope of the aircraft, (Figure 7.6) From this plot, the design SM for each configuration can be identified. These design SM values are presented in Table 7.4, at it can be seen that the static margin of the aircraft was designed to be approximately -15% for all flight configurations.
7.3 Control Surface Detailed Design

The detailed design of the control surfaces focused on the hinging and actuation of the devices. Resulting from this was the selection of suitable servo actuators, as well as the required power source for the actuation system.

7.3.1 Control Surface Hinging

When hinging the control surfaces consideration were given to minimising actuator loads and flutter tendencies. Both of these issues were found to be alleviated through well positioned hinge lines.

The masses of control surfaces can create moments about their hinge lines, which increase required actuation force. This can be alleviated through hinging the surfaces at their centres of gravity.
gravity. However, since the surfaces all weigh less than 50g, such moments were considered to be negligible, and hence actuation loads were not considered for positioning of the control surface hinge lines.

Flutter considerations drove the positioning of the hinges for the control surfaces. The mass moments generated by the control surfaces about their hinge lines create flutter tendencies, which causes loss of control surface effectiveness, (Raymer, 2006). These can be alleviated by hinging control surfaces along their centroidal axes. This could easily be done for the elevators and rudder, however unlike the elevators and rudder, the ailerons do not cover the majority of their respective lifting surface. This makes hinging the ailerons along their centroidal axis difficult, due to the need to manufacture hard points within the wing for hinging. However, Raymer (2006) claims that flutter tendencies can be significantly reduced by having little play in hinge shafts. In addition, Raymer (2006) claims that short actuator rods, which are less susceptible to buckling, also reduce flutter tendencies. Consequently, the ailerons were specified to be hinged at the leading edges of the devices, with piano hinges, which have little shaft play. In addition, the actuator rods were also specified to be as short as possible.

The resultant hinge lines for the control surfaces, as measured from the leading edges of the mean aerodynamics chords of their respective lifting surfaces, were specified as follows:

\[ C_{Hinge,El} = 0.80C_H \]

\[ C_{Hinge,Rud} = 0.80C_V \]

\[ C_{Hinge,Ail} = 0.79C \]

### 7.3.2 Control Surface Deflections

Using recommendations from Simons (2002) \( \theta_{\text{surface max}} \) was specified as 20deg for each control surface.

### 7.3.3 Actuators

Electro-mechanical servo motors are used to actuate the control surfaces. The sizing of these motors was performed by estimating the maximum torque expected on the motors, and suitably sized proprietary units were then selected to meet the required performance level. The required electrical power for these devices was determined through experimental means.

As is common practice with small scale aircraft, one servo is used to actuate the elevators, and one to actuate the rudder. Both of these use push rods to transfer the moment causing forces. Again in line with common practice, two servos, one in each wing, actuate the ailerons. Through the use of specified short push rods, aileron flutter tendencies were reduced.
Actuator Requirements and Design Torques

The resisting moments on the control surfaces are the result of the moments due to the masses of the control surfaces, and the aerodynamic loads. As mentioned, the mass moments acting on the control surfaces were considered negligible, meaning that the moments acting on the control surfaces were due to aerodynamic forces only.

Estimates of the aerodynamic loads acting in the control surfaces were obtained using Equation 7.7 (Eger, 1993)

\[ F_{surface} = K_F q S_{surface} \sin(\theta_{surface}) \]  

\( (7.7) \)

Using the information in Eger (2003) \( K_F \) was approximated to be 1.2.

The resultant moment acting on each control surface was thus found to be given by Equation 7.8.

\[ \tau_{surface} = 1.2 q S_{surface} \sin(\theta_{surface}) \]  

\( (7.8) \)

To calculate the maximum torques needed to actuate the surfaces, simple trigonometry was used to determine the relationship between the torque produced by the servo motor, and the moment acting on the control surface.

The use of proprietary length control horns for the servos and control surfaces resulted in the following lengths:

\[ l_{servo} = 10 \text{mm} \]

\[ l_{surface} = 20 \text{mm} \]

The maximum deflection of the servos was set at 45deg.

Consequently, for maximum servo and surface deflection, the following relationship (Equation 7.9) is valid.

\[ l_{surface} \approx l_{servo} \approx 7 \text{mm} \]  

\( (7.9) \)

Using Equation 7.9, the following relationship was derived (Equation 7.10).

\[ \tau_{servo_{\text{max}}} = \tau_{surface_{\text{max}}} \cos(\theta_{servo_{\text{max}}} - \theta_{surface_{\text{max}}}) \left( \frac{\sin(\theta_{surface_{\text{max}}})}{\sin(\theta_{servo_{\text{max}}})} \right) = 0.44 \tau_{surface_{\text{max}}} \]  

\( (7.10) \)

The results of the above analysis are presented in Table 7.5.
Table 7.5: Determined Control Surface Moments and Required Actuator Servo Torques.

<table>
<thead>
<tr>
<th>Control Surface</th>
<th>Area [sq. m]</th>
<th>$\theta_{\text{surface max}}$ [deg]</th>
<th>$\tau_{\text{surface max}}$ [kg.cm]</th>
<th>$\theta_{\text{servo max}}$ [deg]</th>
<th>$\tau_{\text{servo max}}$ [kg.cm]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aileron</td>
<td>0.015</td>
<td>20</td>
<td>0.70</td>
<td>45</td>
<td>0.31</td>
</tr>
<tr>
<td>Elevators</td>
<td>0.022</td>
<td>20</td>
<td>0.32</td>
<td>45</td>
<td>0.14</td>
</tr>
<tr>
<td>Rudder</td>
<td>0.0076</td>
<td>20</td>
<td>0.16</td>
<td>45</td>
<td>0.07</td>
</tr>
</tbody>
</table>

In addition to being able to provide the required torques, the servos were also required to:

- Be light weight
- Have high reliability
- Fit into available space
- Be of an analogue type; the recommended type for use with the MP2028g autopilot system

**Servo Selections**

To improve the reliability of the airframe, metal geared devices were specified for the rudder and elevator. Unfortunately metal geared servos, which could be located in the wings and used for aileron actuation, could not be sourced, and as such plastic gears were specified for these surface actuators. However, servos with maximum design torques much greater than the expected maximum required torques from the aileron servos, were specified to address this reliability issue.

Table 7.6 details the selected servos for the UAV. The information in this table demonstrates how each type satisfies the relevant requirements.

Table 7.6: Selected Servos

<table>
<thead>
<tr>
<th>Control Surface</th>
<th>Selected Servo</th>
<th>$\tau_{\text{servo max}}$ [kg.cm]</th>
<th>$\tau_{\text{servo design}}$ [kg.cm]</th>
<th>Safety Margin [%]</th>
<th>Weight [g]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aileron</td>
<td>JR NES 339</td>
<td>0.31</td>
<td>3.25</td>
<td>90.5</td>
<td>18</td>
</tr>
<tr>
<td>Elevators</td>
<td>JR ES 579</td>
<td>0.14</td>
<td>8.30</td>
<td>98.3</td>
<td>41</td>
</tr>
<tr>
<td>Rudder</td>
<td>JR ES 579</td>
<td>0.07</td>
<td>8.30</td>
<td>99.2</td>
<td>41</td>
</tr>
</tbody>
</table>

**Required Servo Battery Power**

The selected servos require a 4.8V power source, thus batteries capable of producing this were needed. However, the charge requirements of the batteries needed to be estimated via experimental means. To determine the required amount of charge simulated torques, as shown in Table 7.6, were placed on the servos and current measurements were recorded as the servos were actuated.
The results of this experiment revealed that the collective peak current draw of all of the servos is
1A. Thus, to provide sufficient battery power to complete the mission of the aircraft, a minimum
of 1.1Ah of battery charge was required. However, it was desirable to ensure control surface
operation was maintained even after the propulsion system has gone off line. Thus, 2.2Ah of
battery charge was specified to accommodate this.

The batteries selected for the actuation system were four, standard, Ni-MH, AA batteries con-
nected in series. These batteries provide 4.8V and 2.500Ah of charge, thus satisfying the above
requirements.

7.4 Wing Structural Design

7.4.1 Requirements

The wing structure was designed to meet several operational requirements. These were related to
both the structural properties of the wing during operation and the practicality of transporting
and handling the aircraft. These requirements were divided into the following categories and
addressed during the design process.

- Support a 9kg aircraft during flight with the appropriate load and safety factors as per
  CASR
- Removable wings for transport
- Withstand normal handling conditions
- Withstand impact in all flight regimes
- Withstand loads induced by parachute deployment

Some of these requirement were difficult to quantify and reasonable judgement, based on advice
from experienced persons, was used to generate a solution.

7.4.2 Wing Structural Layout

The following section refers to the wing structure, shown in Figure 7.7. This structure is com-
prised of five main elements; the foam core of the wing, carbon fibre wing spars, a fibreglass
skin, carbon fibre wing boxes and the aluminium alloy tongue that is mounted to the fuselage
and connects the two wings. The components are designed to transfer all loads generated from
aerodynamic forces from the wings to the internal structure of the fuselage while allowing for
the wings to be simply removed from the aircraft for transport.
7.4.3 Aerodynamic Loading

Design Factors and Assumptions

The following factors were used in all of the wing design calculations:

- A load factor of 3.8 in accordance with CASR
- A safety factor of 2.25 for composite materials in accordance with CASR101
- A safety factor of 1.5 for composite materials in accordance with CASR101

The following assumptions were made in all analysis:

- The aerodynamic loads from the fuselage were negligible, based on the non-lifting surface design of the fuselage shape. In reality some lift is created by the fuselage, meaning that the wings provide less lift than the models developed in this section suggest. Consequently, the presented load analyses provide slight overestimates of the actual bending and shear stresses in the wing structure, which was considered favourable to the aircraft’s design.

- The aerodynamic load due to drag is insignificant when compared to the aerodynamic lifting force and torque, and thus was not considered in the analyses.

- The model used to estimate the lift distribution over the wing did not consider wing twist or the reduction in lift near the wing tip due to vortex production. In reality, these two factors bias the load profile towards the wing root, (Figure 7.8). Consequently the model provided a favourable overestimate of the maximum bending stresses in the wing, as it estimated the loads to be further outboard from the wing.
Lifting Force Distribution

The aerodynamic loading profile over the wing was investigated so an evaluation of the stresses within the wing structure could be carried out. Published load approximations were used to estimate the loading on a wing, however it should be noted that the presence of tip vortices and different wing shapes make calculating this load precisely a difficult task.

The aerodynamic lift load distribution curve for an elliptical wing (as shown in Figure 7.9) is of an elliptical shape (Raymer, 2006). It can be seen in this curve that the aerodynamic load is a maximum at the root of the wing and zero at the tip. This is due to the maximum chord length at the root of the wing and a zero chord length at the tip. Although the aerodynamic load curve of an elliptical wing is an ellipse, it should be noted that it is not the same shape as the plan form of the wing. The area under the load curve must be equal to the lift required for the aircraft.

For a non-elliptical wing an approximation of the load distribution can be used to determine the aerodynamic loading. This spanwise load estimation is known as Schrenk’s approximation.
(Raymer, 2006). For a tapered wing a trapezoidal load distribution, with a total area equal to the required lift, provides a very rough load approximation. The Schrenk approximation improves this distribution, by averaging it with the ideal elliptical distribution. This approximation is shown in Figure 7.10.

Figure 7.10: Schrenk’s Load Approximation (Raymer, 2006)

To determine the trapezoidal and elliptical lift distributions over the wing, the following equations from Raymer (2006) were used:

**Trapezoidal Chord**

\[
C(y) = C_{\text{root}} \left[1 - \frac{2y}{b} (1 - \lambda)\right]
\]  
(7.11)

**Elliptical Chord**

\[
C(y) = \frac{4S}{\pi b} \sqrt{1 - \left(\frac{2y}{b}\right)^2}
\]  
(7.12)

where \( S = \frac{b}{2} C_{\text{root}} (1 + \lambda) \)

Equation 7.11 and 7.12 describe the chord of the wing itself. Hence, when integrations are performed over the span of the wing, these equations yield the wing surface area, not the aerodynamic load for each case. Hence, both Equation 7.11 and Equation 7.12 were multiplied by a constant such that each integration gave the total lifting force required. Using equations 7.11 and 7.12, the sizing parameters of the wing and Schrenk’s approximation the equation for the load distribution on the wing was obtained. It should be pointed out this analysis was performed to one half of the wing, and applied to both halves using the fact that the wing is symmetrical. Consequently, all of the following figures and values are for one half of the wing only.

**Accounting for Wing Weight**

Analysis of operational wing loads requires the combination of aerodynamic lift distribution and wing weight distribution. The summation of these distributions more accurately represents the
forces experienced by the wing structure. Using a linear approximation of the wing weight with respect to spanwise chord length as recommended by Raymer (2006), an equation for the load profile over the wing was determined, (Equation 7.13).

\[ P = 0.336 \sqrt{\frac{998001 - 1000000y^2}{\pi}} + 121.006 - 74.28 \]  

(7.13)

**Modifications to Raymer’s Method**

Raymer’s method places the root of the wing at the centreline of the fuselage, as a result the wing reference area includes an ‘imaginary’ area concealed by the fuselage. It was calculated that this area accounts for approximately 7% of the total lifting force of the wing. However, the lift due to the fuselage was assumed to be negligible, and hence this force needed to be redistributed over the actual wing area. This was performed by distributing the virtual fuselage lifting force evenly over the actual wing area, which resulted in the addition of a constant force of 13.06N to Equation 7.13, such that the total area under the curve between \( y = 0.055 \text{m} \) and \( y = 0.999 \text{m} \) was equal to the lift required for the aircraft. The resultant load distribution (shown in Figure 7.11) developed from this method was used for the structural design.

![Wing Load Distribution](image)

**Figure 7.11: Aerodynamic Load Distribution over Wing (Root to Tip)**

**Determining Shear Force and Bending Moments**

The integral of the load distribution equation from \( y = 0.055 \) to 0.999 along with the appropriate boundary conditions resulted in the determination of the shear force distribution over the wing, as shown in Figure 7.12. Similarly the bending moment distribution, as shown in Figure 7.13, was determined through integration of the shear force distribution, again with the use of the appropriate boundary conditions determined from basic stress analysis. Given the complexity of the load distribution equation, these integrations were performed with the assistance of Matlab 7.0 software.
To verify the shear force and bending moment distributions generated, the shear force and bending moment at the root of the wing was calculated using the derived distributions. The values obtained were then checked with hand calculations for a simple cantilever beam analysis of the wing. The correlation of the two sets of results suggested that the derived load, shear force and bending moment distributions were valid, and suitably accurate.

### 7.4.4 Materials

The materials selected for the wing design were uni directional carbon fibre, pain weave carbon fibre cloth, an epoxy resin and isolite. Table 7.7 shows where these materials were used in the wing.

<table>
<thead>
<tr>
<th>Material</th>
<th>Where used</th>
</tr>
</thead>
<tbody>
<tr>
<td>Uni directional carbon fibre</td>
<td>Wing spar, wing box</td>
</tr>
<tr>
<td>Plain weave carbon fibre cloth</td>
<td>Leading edge</td>
</tr>
<tr>
<td>Isolite foam</td>
<td>Core</td>
</tr>
<tr>
<td>Epoxy resin</td>
<td>Used in conjunction with carbon fibre</td>
</tr>
</tbody>
</table>
7.4.5 Structural Analysis

The structural analysis of the wing was based around the four individual components of the structure. These were the wing spars, foam core, the sleeve or wing box within the wing and the tongue within the fuselage. The wing spars were designed to carry both shear due to the lifting force and the bending moment created by the aerodynamic lifting force. The foam was analysed to resist a torsional force only, while the wing box and tongue were analysed to transfer both the shear force and bending moments from the lifting force into the fuselage.

Wing Spars

Spars were located in the spanwise direction of the wing and provided the primary component of bending and shear stress resistance for the lifting surface. The spars were located based on statistical research. The locations of the front and rear spar are at 15% and 78.6% of the chord length respectively (Raymer, 2006). 1.25mm thick, uni directional carbon fibre was selected for these spars. The fibres in this material are cured under tension, and thus result in a stiffer material. A stress analysis was performed to determine the required width of the spars.

The structure itself was analysed by finding the centroid of all four spars, and considering them as a single structure. This was done using Equation 7.14 and 7.15.

\[
x_{\text{centroid}} = \frac{\sum A_i x_i}{\sum A_i} \quad (7.14)
\]

\[
z_{\text{centroid}} = \frac{\sum A_i z_i}{\sum A_i} \quad (7.15)
\]

The data describing the airfoil profile was used in a cartesian co-ordinate system to locate the centroid of the spars in both the x (longitudinal) and z (vertical) directions. Once the centroid of the profile was known, the moment of inertia of the structure was calculated assuming that inertia due to the cross sectional area of each spar was negligible and that only each spar’s distance from the centroid contributed to the moment of inertia of the structure. Thus, the moment of inertia about the x (longitudinal) axis was found using Equation 7.16.

\[
I_x = \sum A_id_{zi}^2 = A\sum d_{zi}^2 \quad (7.16)
\]

Where \(d\) is the distance between the centroid and the centre line of the spar.

The stress in each spar was then determined using Equation 7.17.

\[
\sigma = \frac{Mz}{I} \quad (7.17)
\]

For the initial analysis the spar structure was considered as a beam with four elements. Using the above equations, it was found that the point of greatest bending stress is always on the upper edge of the spar on the top surface of the wing, at 15% of the chord length and at the
root of the wing. Hence by substituting the appropriate z value it was possible to analyse this stress and compare it with the maximum yield stress for a carbon fibre, epoxy resin structure. An initial analysis showed that a 12mm wide (x-direction) spar provides a reserve load factor greater than 3.7, over the entire length of the spar. Although over engineered, these dimensions were used to ensure structural integrity was maintained during parachute operation, where the loads experienced by the aircraft were somewhat unknown.

Figure 7.14 shows the spanwise bending stress in the wing, and also shows the yield stress of the carbon. In this figure, it can be seen that the reserve factor increases considerably towards the tip of the spar and the root is the critical point. It should also be noted that the fuselage ends at the point 0.055m from the root of the wing, hence for 0<y<0.055m the spar has been analysed but will not actually exist, this section will be supported by the tongue connecting the wings. The spar will begin at the root of the wing, a distance of 0.055m from the fuselage centreline. In addition, for 0.055<y<0.195m there exists wing boxes, constructed from a matrix of resin and chopped carbon fibre and fixed between the upper and lower spars. These boxes will considerably increase the strength and in turn the reserve factor in this region.

![Figure 7.14: Stress in Upper 15% Spar due to Bending and Yield Stress of Carbon Fibre](image)

The shear stress in the spars was also calculated. An initial analysis based on the assumption that a single spar carries all of the bending and shear stress was used. It is known from the plot of the shear distribution, shown in Figure 7.12, that the shear force in the wing decreases from the root to the tip of the wing. A brief, conservative analysis was done using the standard safety factor of 2.25, a shear load force of 170N, and the minimum cross sectional area of the spars, which is 1.25mm x 12mm. The standard equation for shear stress in a beam, Equation 7.18 was used to determine the shear stress in a single spar at the critical point.

\[
\tau_{\text{max}} = \frac{3V}{2A}
\]  

(7.18)

It was found that the maximum shear stress was 17MPa. Given the yield stress of the carbon is 70MPa this gives a reserve factor of around 4. Since the shear force will not be more than 170N for y>0.055, there will be four spars rather than just the one used in this analysis and the spars are not thinner than 1.25mm it was concluded that the shear stress was not critical for
this structure.

For further confidence in the analysis another method was used to calculate the maximum bending stress in the spars. This method used the geometry of the wing and the position at which the aerodynamic force is applied. It was assumed that the centre of pressure is at approximately 25% of the chord length. The front spar is placed at 15% and the rear at 78.6%. Using the ratios below in Equations 7.19 and 7.20 the bending experienced by the front and rear spars was determined.

\[
\%load = \left( \frac{78.6 - 25}{78.6 - 15} \right) = 84\% \tag{7.19}
\]

\[
\%load = \left( \frac{25 - 15}{78.6 - 15} \right) = 16\% \tag{7.20}
\]

Using this load separation between the front and rear spars, the spar pairs were now considered separately. The moment of inertia of each spar pair, that is the upper and lower element, was calculated using a similar method to the one presented previously. Evaluating each spar based on the appropriate loading values, depending on distance from the root and the spar, enabled the shear stresses to be plotted against the span of the wing and compared to the yield stress of the carbon fibre. As before, Equation 7.17 was used in this analysis.

Both of the methods used to determine the stresses in the spars produced values that are considerably lower than the yield stress of the material being used.

**Torsion**

The aerodynamic load on all wings creates a torque about the y-axis of the aircraft. This torque is caused by the difference in the location of the centre of pressure of the wing and the neutral point or point at which there are no torsional stresses in the wing. The neutral point was calculated using the method in Beer et al. (2006). This neutral point was a function of the foam in the central section of the wing between the spars and the spars themselves. The location of the centroid of the spars and the foam was calculated using Equation 7.14 and 7.15 presented above. A weighted average of these two centroids was then used to calculate the neutral point. The average was weighted according to the material’s modulus of elasticity. The distance between the neutral point and the centre of pressure is the lever arm that creates the torsion in the wing.

The centre of pressure was found for various angles of attack and Reynolds numbers. Figure 7.15 shows that Reynolds number does not effect the location of the centre of pressure below the stall angle (10deg) however the location does change with various angles of attack while the neutral point is static. As a result of this changing lever arm it was necessary to find the angle of attack that would give the greatest torsion on the wing. The lever arm was calculated for angles of attack between 0 degrees and 20 degrees. This lever arm was then multiplied by the lifting force associated with that angle of attack, this gave the torque on the wing. The results of the torque at different angles of attack can be seen in Figure 7.16. It should be noted that a dip in the torque curve appears during the onset of stall, due to the spoon effect.
The wing was designed such that only the foam would resist all of the torsion generated by the wing. The shape of the foam was approximated to be a trapezoid between the spars. Following a method in Beer et al. (2006) used to calculate torsional stresses in a prismatic bar, some assumptions were made in order to perform the analysis. These assumptions were as follows:

- The foam was approximated to be of rectangular cross section
- The height of the foam was equal to the average vertical distance between the spars

This method produced a torsional stress of 321kPa in the worst case scenario with a safety factor of 2.25. The cross breaking strength of the isolite foam being used for the wing core is 260kPa, hence the foam was found to be incapable of withstanding the torsional forces of the aerodynamic loads.
To overcome this problem, advice was sought from experienced composite designers and builders. Using this advice, an 85gsm fibreglass mat / epoxy resin composite skin was used for the wings. It was determined that these three layers, in conjunction with the spars, would be capable of resisting the torsion unable to be withstood by the isolite. In addition the spars will resist some torsion. Further analysis in this area is required to fully quantify the wings ability to resist torsion however due to time constraints the advice received was deemed sufficient, given the safety factors used.

**Wing Box**

The wing box forms the connection between the top and bottom spars within each wing halve. The dimensions of the wing box are determined by the size of the tongue that is mounted in the box and the distance between the spars. The forward tongue has a thickness (x) of 6mm and a height (z) of 20mm tapering down to 6mm x 18mm. This resulted in the forward wing box having a thickness of approximately 12mm and a height of 25mm. The rear tongue has a thickness of 12mm and a height of 6mm tapering down to 10mm x 5mm. The boxes have been designed using 200gsm 0/90° plain weave carbon fibre, epoxy resin and a mixture of chopped mat, filler and resin. The manufacturing of this component is discussed later in Section 8.2. The length of the wing box has been designed to be 135mm in length, based on the predicted tolerance of the wing box.

**Tongue Design**

Figure 7.17 shows the design and assembly of the tongue to be fixed in the fuselage between the wings. It is constructed from two components, the rear tongue and web and the front tongue. The two components fit together with a press fit and are further held in place by the ring frames in the fuselage. A structural analysis similar to that of the wing spars was conducted on the tongue, the results of which are presented in Table 7.8. The required dimensions of the tongue were determined from this analysis, (Table 7.9). Both tongues are tapered to allow for easy removal and connection with the wings while maintaining a tight fit. The front tongue is tapered on the top and bottom surfaces only, while the rear tongue is tapered on all surfaces due to additional space restrictions. Both front and rear tongues extend 135mm into the wing and as a result are tapered for the outboard 135mm.
The structural analysis was performed based on an aluminium alloy, namely 2024-T3, 6.3mm plate with a yield stress of 345 MPa. Sourcing this material proved to be difficult and subsequently influenced the design. The webbing between the spars, front spar and rear spar were all designed to be manufactured from 6mm plate, as a result only one piece of material was required.

The front spar is connected to the web and rear spar using a slot and tongue locating method. A press fit locates and holds the tongue within the slot. In addition each component is to be individually bolted to the ring frames in the fuselage, to ensure they are fixed in location during vehicle operation.

The tongue was designed to be connected to the ring frames in the fuselage using 4mm and 3mm stainless steel, allen head bolts with nylon locking nuts for security. The location of these bolts and more specifically the holes was carefully considered in order to not reduce the structural
integrity of the tongue. The standard guideline restricting all holes to be at least two hole
diameters from the edge of the material was followed and the holes were specified to be as small
as possible. The front bolts were required to pass through 20mm of ring frame and 6mm tongue,
therefore the minimum length required was 35mm to allow for a nyloc nut and washer. In addition,
due to access issues an allen head type was used. The rear bolts were more restrictive as there
was less material in the tongue where a hole could be made. Due to the radii in the web of the
tongue it was still possible to maintain the guideline mentioned above using a 3mm bolt. Due
to the reduced loads in the rear spars and tongue the ring frame was also thinner and hence
a shorter bolt, 25mm, was found to be suitable. The loads on the front and rear bolts were
identified as shear and tension respectively due to different configurations. 304 stainless steel
was available for all fasteners, for increased stress tolerance. It was calculated that the front
and rear bolts have reserve factors of 30 and 322 respectively and hence were deemed suitable.
Bolts with smaller diameters were not available in lengths required and hence the previously
mentioned bolts were used despite the large safety factors.

7.4.6 Summary of Wing Structure

The wing structure included the wing and tongue connecting them to the fuselage. This design
allowed for the simple removal of the wings for transport while maintaining structural integrity
during operation. The carbon fibre spars in the wing transfer all aerodynamic loads to the
tongue which in turn transfers the loads to the fuselage. The three layers of 85gsm fibreglass on
the wing not only resist torsional moments but also resist damage during handling and impact.
This structure was deemed manufacturable, however some modifications were made during this
process.

7.5 Fuselage Structural Design

This section details the design of the fuselage from the development of the requirements for the
fuselage through to the tooling required for production. The design of the fuselage was driven by
the internal volume requirements for the housing of the mechanical and electronic subsystems.

7.5.1 Fuselage Design Specifications

The fuselage was required to meet the following design specifications:

- House all mechanical and electronic components
- Be lightweight
- Be able to resist the aerodynamic, static, inertial and axial loadings acting on the airframe
- Produce low aerodynamic drag
- Have easy access to interior, for easy installation of batteries and electronic componentry
- Be simple to manufacture
### 7.5.2 Structural Layout

The fuselage structural layout is driven by the requirements of internal components, the loading that is generated during operation and the requirements for operational performance and efficiency. This section details the layout of the fuselage structure, the determination of loads required to be carried, the selection of suitable materials and the analysis of stress in the structure. The resultant structure is essentially a combination of an internal skeleton with a load bearing skin, similar to a semi-monocoque structure.

#### Internal volume requirements

To determine the internal volume requirements of the design, a database of all internal components and their dimensions was compiled, (Table 7.10). From this database, minimum physical dimensions could be determined in order to house the components with adequate room for installation and wiring. The physical integration of these internal components was modelled using CAD as shown in Figure 7.18, with consideration being given to location, access for wires and plugs, weight and size, to minimise the fuselage cross section.

<table>
<thead>
<tr>
<th>Component</th>
<th>Quantity</th>
<th>Height [mm]</th>
<th>Width [mm]</th>
<th>Length [mm]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Motor Batteries</td>
<td>8</td>
<td>25</td>
<td>44</td>
<td>142</td>
</tr>
<tr>
<td>ESC</td>
<td>1</td>
<td>25</td>
<td>75</td>
<td>50</td>
</tr>
<tr>
<td>Motor</td>
<td>1</td>
<td>60</td>
<td>60</td>
<td>85</td>
</tr>
<tr>
<td>Autopilot</td>
<td>1</td>
<td>50</td>
<td>38</td>
<td>101</td>
</tr>
<tr>
<td>RF Modem (AP)</td>
<td>1</td>
<td>15</td>
<td>54</td>
<td>89</td>
</tr>
<tr>
<td>Transmitter (Cam)</td>
<td>1</td>
<td>40</td>
<td>70</td>
<td>90</td>
</tr>
<tr>
<td>Cameras</td>
<td>2</td>
<td>45</td>
<td>50</td>
<td>60</td>
</tr>
<tr>
<td>RC Receiver</td>
<td>1</td>
<td>10</td>
<td>30</td>
<td>60</td>
</tr>
<tr>
<td>AP Batteries</td>
<td>1</td>
<td>10</td>
<td>30</td>
<td>55</td>
</tr>
<tr>
<td>Parachute</td>
<td>1</td>
<td>50</td>
<td>90</td>
<td>600</td>
</tr>
<tr>
<td>Aux Servos</td>
<td>2</td>
<td>24</td>
<td>13</td>
<td>22</td>
</tr>
<tr>
<td>Control Servos</td>
<td>2</td>
<td>40</td>
<td>19</td>
<td>38</td>
</tr>
<tr>
<td>RC Receiver Batteries</td>
<td>1</td>
<td>15</td>
<td>75</td>
<td>50</td>
</tr>
<tr>
<td>Camera Batteries</td>
<td>1</td>
<td>31</td>
<td>30</td>
<td>106</td>
</tr>
<tr>
<td>GPS ground plate</td>
<td>1</td>
<td>1</td>
<td>100</td>
<td>100</td>
</tr>
<tr>
<td>Servo Board</td>
<td>2</td>
<td>31</td>
<td>48</td>
<td>13</td>
</tr>
<tr>
<td>Deployable Payload</td>
<td>1</td>
<td>65</td>
<td>65</td>
<td>200</td>
</tr>
</tbody>
</table>
To provide adequate accessibility for installation and maintenance of components, removable hatches needed to be designed into the fuselage. A hatch was also needed, which could contain the parachute recovery system during flight, and open during parachute deployment. These removable hatches affected the load paths within the fuselage, and hence the locations and sizes of these hatches had to be carefully considered.

**Internal Structure**

The internal structure was designed around three main requirements. The connection of the wings to fuselage, the mounting of components and the mounting of the motor. The design of the wing structure requires two tongues, and hence two wing boxes in each wing. The tongue is required to transfer the loads from the wings into the fuselage, ideally through the internal skeleton. The connection between wings and fuselage also needs to be designed for removal.

The autopilot is required to be mounted near the centre of gravity of the aircraft and co-planar to the longitudinal and lateral axis of the aircraft. These requirements were primary in the design process of the mounting technique. The number and weight of the motor batteries determined their position in the aircraft in order to ensure inherent stability, this also drove the design of the mounting technique.

Motor weight was the main consideration with the design of an motor mount. This is not to say the torsional or axial loads of the motor were ignored, just that they were secondary to the weight
CHAPTER 7. AIRFRAME DETAILED DESIGN

of the motor and the associated inertial loads. The method of mounting the motor determined the materials and layout of the nose cone.

These considerations culminated in the development of the resultant internal fuselage structure as shown in Figure 7.19.

![Figure 7.19: CAD sketch of the developed internal fuselage structure](image)

Skin

To minimise the weight of the total airframe a load bearing skin was selected to reduce the amount of load bearing internal structure. Consequently, the skin was required to transfer the loads from empennage and the motor into the internal skeleton without deformation or significant buckling. These loads consisted of bending, shear, torsion and buckling.

Aerodynamic Considerations

To reduce form drag from the fuselage a minimum cross section is required, however skin friction drag is the major contributor to fuselage drag. To minimise drag from skin friction the exposed, or wetted, area of the fuselage needed to be minimised. A circular fuselage provides high strength, but at the cost of less usable internal volume for a given wetted surface area. Reducing the wetted area will reduce drag, however minimum width and height dimensions were determined previously. A trade off between strength and wetted area ultimately determined the shape of the fuselage. The resultant structural design can be seen in Figure 7.20.
The minimum size of the empennage of the aircraft is typically determined by the requirements of the mounting of the tail section. Sufficient structure is required to mount the horizontal and vertical stabilisers, from this minimum area the empennage can be tapered out to the main fuselage section. It is recommended not to exceed an angle of 15 degrees for the taper for purposes of separation (Raymer, 2006), and this recommendation was adhered to in the design of the fuselage.

**Deployable Payload Positioning**

The position and mounting of the payload determined the final design of the fuselage. An internally mounted payload would provide aerodynamic efficiency for the duration of the flight at the cost of complicating the design and manufacturing process and a reduction in available internal volume. Simplifying the design to mount the payload externally allowed the simpler fuselage design, with separately designed, modular payload system, capable of providing more mission flexibility. For these reasons, the second design option was adopted.

The positioning of the payload system had to be such that the location of the centre of gravity of the aircraft is not significantly altered once deployed. This was achieved by incorporating the ability to shift the payload within its system, so as the centre of gravity of the payload remains beneath the centre of gravity of the aircraft for all flight configurations.
7.5.3 Selection and Justification

Layout

Component layout resulted in a minimum width of 110mm and a minimum height of 180mm for the main section of the fuselage. This layout was driven primarily by the location and orientation of the auto pilot system, and the volume of the propulsion system batteries.

The layout of the internal components consists of two separate compartments aligned with the longitudinal and lateral axis of the aircraft, one lying above the battery board, the other below. For the purposes of lateral stability the motor batteries were designed to be located as low as possible in the aircraft. The camera was located in the bottom compartment so as effective ground surveillance could be carried out by the device.

As alluded to previously, the mounting of the motor dictated the diameter of the nose of the aircraft, which was designed to be 80mm. The length of the nose cone was determined by the length of the motor and the requirement for electronic components to be housed in this section, and this was determined to be 180mm.

Accessibility

In order to access all components in the aircraft each compartment required a separate access panel or hatch. Structural considerations required the size of these hatches to be as small as possible. The size of the batteries meant only a portion of the battery needed to be exposed to allow for their removal. This allowed for a hatch that was not required to cover the length of the entire battery bay. A small window was also required for the camera. A bottom hatch 480mm in length and the width of the aircraft was used.

A removable hatch for the top compartment needed to allow access for all electronic and control components, but more importantly needed to allow for the deployment of a parachute. The physical dimensions of the folded parachute dictated the size of this top hatch. The length of this hatch is 600mm with the width being dictated by the width of the fuselage (110mm).

To access the motor it was been decided to design a removable nose section as it reduced the complexity of construction and maintains structural integrity of the nose cone when compared to removable hatches.

Structure

The weight and size of the batteries determined the mounting platform for the components. Aircraft grade 6mm plywood has been used for the mounting platform for the motor batteries and all internal fuselage components. The number of laminates in this ply provides sufficient strength for attaching components, both on top and bottom. This board was the width of the internal dimensions and attached to a ring frame at the rear made from the same material. The front ring frame is made from two sheets of 4.5mm ply held together with bolts and nylon locking
nuts. The board is 600mm in length. To ensure the structural integrity of these frames and board a laminate of fibreglass epoxy resin has been adhered to both side.

Transferring the load from the wings to this internal structure was achieved using two central ring frames similar to those previously mentioned. Stainless steel bolts mount the wing tongue to these thicker ring frames attached to the mounting board. The entire structure of four ring frames and mounting board carries the inertial and gravity loads of the internal components.

The motor will be mounted to the front of the nose cone which is removable by unbolting the two plates making up the front ring frame. This is due to the rotational nature of the main motor component and the length of the propeller shaft.

The skin will consist of a honeycomb sandwich nose cone to carry the loads of the motor and a closed cell foam sandwich on the sides of the main fuselage to carry the loads from the empennage through area of the top and bottom hatches. The empennage will taper from the main fuselage section to the tail section and incorporate two small ring frames to resist buckling. All ring frames will be joined by an external longeron extending the entire length of the aircraft.

**Shape**

To optimise the aerodynamic efficiency of the aircraft a shape resulting in the least amount of wetted area was selected for the main section of the fuselage. For aesthetics and volume requirements this shape consists of parallel sides, a flat bottom and a curved top. The nose section tapers to the minimum required area for motor mounting in line with the practise of minimizing wetted surface area.

**Deployable payload**

The payload system developed, was developed in-situ on the airframe and can be seen in Figure 7.21. This first prototype was developed within a short timeframe, and the design focussed on operational reliability, rather than aerodynamic efficiency, purely for the purposes of the UAV Outback Challenge. The system uses a support bar, which is located to the airframe using two servo motors, and a simple lock mechanism. The bottle is attached to the main bar, and can be shifted for and aft depending on the location of the aircraft’s centre of gravity. To deploy the payload, the servos rotate through 90 deg, causing the support bar and bottle to fall from the heads of the servos.
7.5.4 Load Determination

The loading of the aircraft was required in order to perform a detailed stress analysis and design a suitable structure. A classical approach has been taken to predict these loads, as shown in Figure 7.22, with detailed hand calculations. A static analysis was conducted on the aircraft under a maximum pull up manoeuvre. This analysis comprised of aerodynamic, weight, inertial and thrust loading resulting in a load distribution throughout the fuselage for bending, shear and torsion loads. From these plots a suitable structure was able to be designed.

Figure 7.22: FBD of Applied Loads to the Fuselage

This analysis has been conducted at the maximum design speed of 120 km/h and with a 3.8 load factor and a 2.25 safety factor as per CASA regulations. This process is detailed in Appendix C. The result of this analysis is summarised in the Figure 7.23.
7.5. FUSELAGE STRUCTURAL DESIGN

7.5.5 Material Selection

The specification of materials was required in order to analyse stresses in the fuselage structure. The following details the selection of appropriate materials for structural components. The two critical components analysed were the aircraft’s centre board and the empennage skin.

Centre board material

Aircraft grade 6mm plywood has been used for the mounting platform for the motor batteries and all internal fuselage components. The number of laminates in this ply provides sufficient strength for attaching components, both on top and bottom. Although the exact material specifications could not be determined the expected stresses with in this centre board were calculated to ensure that they would fall within reasonable limits. The incorporation of reserve factors was deemed sufficient to ensure functionality of this component.

Estimation of Skin Thickness

A solid estimate of the skin thickness was required in order to estimate the stresses in the structural members. A known structural member made from three layers of 200 gsm fibre glass cloth and resin matrix has a thickness of 1.5 mm. This was used to estimate the thickness of a structure using three layers of 85 gsm fibreglass cloth. It was estimated that this skin thickness would be of the order of 0.75 mm. This value was used for all stress analysis calculations.
7.5.6 Stress Analysis

A preliminary stress analysis was made on the fuselage concept using elementary stress analysis procedures. The fuselage, empennage and nose cone were analysed separately. The results of this stress analysis was used in order to set the material requirements.

Fuselage Stress Analysis

A simplified stress analysis has been completed on the fuselage section. A schematic of the fuselage structure and its loading is shown in Figure E.2.

The analysis has been conducted assuming that shear, bending and torsional loads are carried through the open skin sections. It is assumed that the 6mm ply board assists significantly in the transfer of these loads however this has been neglected for the purposes of this calculation. A classical hand calculation approach has been used to evaluate the stresses in each of the sections. The method presented in Niu (1999) has been utilised using thin wall open section analysis for the shear and torsion calculations and simple beam theory for evaluating bending. Following this analysis each of the resultant stresses were multiplied by a safety factor of 2.25 as per CASA regulations. The results of this analysis are shown in Table 7.11.

Shear Analysis

\[ \tau = \frac{VQ}{Ib} \]

Bending Analysis

\[ \sigma = \frac{My}{I} \]

Torsional Analysis

\[ \tau = \frac{Tt}{J} \]

Where \( J = \frac{1}{3} \sum b_i t_i \) for skins
7.5. FUSELAGE STRUCTURAL DESIGN

Table 7.11: Stress Analysis Results on the Fuselage with a Safety Factor of 2.25 and a Skin Thickness of 0.75mm

<table>
<thead>
<tr>
<th>Fuselage Station (m)</th>
<th>Shear Force (N)</th>
<th>Shear Stress (MPa)</th>
<th>Bending Moment (Nm)</th>
<th>Bending Stress (Nm)</th>
<th>Torsional Load (Nm)</th>
<th>Torsional Shear Stress (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.308</td>
<td>-18.7</td>
<td>-2.63</td>
<td>2.0</td>
<td>0.28</td>
<td>-0.96</td>
<td>-0.011</td>
</tr>
<tr>
<td>0.462</td>
<td>-38.6</td>
<td>-5.43</td>
<td>44.4</td>
<td>6.27</td>
<td>-0.96</td>
<td>-0.011</td>
</tr>
<tr>
<td>0.616</td>
<td>61.2</td>
<td>-8.60</td>
<td>37.5</td>
<td>5.29</td>
<td>2.63</td>
<td>0.031</td>
</tr>
<tr>
<td>0.770</td>
<td>50.8</td>
<td>7.14</td>
<td>28.9</td>
<td>4.07</td>
<td>2.63</td>
<td>0.031</td>
</tr>
</tbody>
</table>

Empennage Stress Analysis

A similar stress analysis was conducted on the empennage section of the aircraft. This analysis varied slightly from the previous analysis in that the cross sectional geometry changed throughout the section. A schematic of the loading of the empennage can be seen in Figure 7.25.

![Free Body Diagram of the Empennage Loading](image)

Similar equations to those stated previously were used, however thin wall closed section analysis was used and thus the above mentioned equation for shear stress due to torsion was replaced with the Equation 7.21 below.

$$\tau_T = \frac{T}{2At} \tag{7.21}$$

The results of the analysis are presented in Table 7.12 below.

Table 7.12: Stress Analysis Results on the Fuselage with Safety Factor

<table>
<thead>
<tr>
<th>Fuselage Station (m)</th>
<th>Shear Force (N)</th>
<th>Shear Stress (MPa)</th>
<th>Bending Moment (Nm)</th>
<th>Bending Stress (Nm)</th>
<th>Torsional Load (Nm)</th>
<th>Torsional Shear Stress (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.924</td>
<td>49.2</td>
<td>0.46</td>
<td>23.1</td>
<td>9.11</td>
<td>2.45</td>
<td>0.45</td>
</tr>
<tr>
<td>1.078</td>
<td>46.9</td>
<td>0.56</td>
<td>16.4</td>
<td>8.44</td>
<td>2.27</td>
<td>0.54</td>
</tr>
<tr>
<td>1.232</td>
<td>46.5</td>
<td>0.74</td>
<td>9.74</td>
<td>7.19</td>
<td>2.08</td>
<td>0.72</td>
</tr>
<tr>
<td>1.386</td>
<td>46.1</td>
<td>1.21</td>
<td>3.05</td>
<td>4.00</td>
<td>1.89</td>
<td>1.17</td>
</tr>
</tbody>
</table>
Empennage Buckling Stress Analysis

Due to the hollow, thin structure of the empennage a buckling analysis was required to ensure that the structure would not fail due to the failure mode. An analysis presented in Niu (1999) was used to analyse the buckling stresses in the skin, assuming a thickness of 0.75mm. The details of this analysis are presented in Appendix C. Table 7.13 shows the results of this analysis.

<table>
<thead>
<tr>
<th>X(m)</th>
<th>(\sigma_{\text{Bending}}) (Pa)</th>
<th>(\sigma_{\text{Torsion}}) (Pa)</th>
<th>MS (approx.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.9</td>
<td>18313462.6</td>
<td>106799.5</td>
<td>4.273841</td>
</tr>
<tr>
<td>1.1</td>
<td>16948680.9</td>
<td>128908.1</td>
<td>5.025204</td>
</tr>
<tr>
<td>1.2</td>
<td>14409912.7</td>
<td>170497.5</td>
<td>6.811458</td>
</tr>
<tr>
<td>1.4</td>
<td>7983674.9</td>
<td>277593.7</td>
<td>15.92973</td>
</tr>
<tr>
<td>1.5</td>
<td>0.0</td>
<td>0.0</td>
<td>N/A</td>
</tr>
</tbody>
</table>

From the results of the aforementioned buckling load analysis it is safe to say that the tail section will not fail under aerodynamic loading for the given skin thickness and safety margins allowed. Thus this structure does not require any additional reinforcement to counteract buckling.

7.5.7 Results

The above analysis predicts that, for the required internal layout, a fuselage skin structure of 3 layers of 85 gsm fibreglass should be sufficient to withstand the applied aerodynamic loading, providing a skin thickness of approximately 0.75mm. This analysis has been conducted using traditional stress equations and hand calculations with considerations given to buckling of the structure.

7.6 Launch and Recovery Systems

This section of the report details the design of the modular undercarriage; as well the detailed design of the parachute recovery system and the preliminary design of a car launch mechanism. For preliminary testing and prototyping this UAV has been configured to be launched and recovered using a tricycle undercarriage. A parachute recovery system has designed, tested and integrated into the airframe for either primary or emergency recovery. This system increases the operational flexibility, safety and autonomy of the UAV system as a traditional runway and experienced pilot is not required for recovery. An alternative car launch method was also considered for the launch of this UAV. The preliminary design of such a system is presented however this system was not completed due to the project timeframe and the risk involved in testing such a system.
7.6. LAUNCH AND RECOVERY SYSTEMS

7.6.1 Landing Gear System

The UAV requires a launch system for flight. The predicted take off velocity is of the order of 70 km/h. The system must be capable of launching from grass or tarmac without damage and support the aircraft in the maximum take off weight condition at the required velocity. This system was aimed to be a simple design for ease of integration and reliability for initial flight tests.

Landing Gear Requirements

The landing gear to be integrated into the airframe has specific design requirements as stated below:

- Stability, both dynamic and static during ground roll operation
- Sufficient strength to carry loads during take off and landing
- Modular in design for integration and alternative launch method

Gear Configuration

A literature review and critical analysis of conventional landing gear configurations indicated a tricycle undercarriage, as seen in Figure 7.26 (a), was most suitable for ease of integration and modularity. An alternative tail dragger configuration, as shown in Figure 7.26 (b) was considered for the increased angle of attack during take off and associated reduction in take off length, however it was decided that the empennage structure would not be sufficient to take the loads of the tail wheel making the configuration unfeasible. To simplify the operation of the system and allow for the conversion to a car launch system the undercarriage was developed to be a single piece modular system installed as an alternative bottom hatch.

![Figure 7.26: Landing Gear Configuration Schematics (Currey, 1988)](image)
CHAPTER 7. AIRFRAME DETAILED DESIGN

Gear Selection

The landing gear was to utilise off-the-shelf landing gear components as much as possible to minimise development time. The following details the selection of the chosen landing gear products.

Main gear A survey of current local markets showed one local manufacture, Bolly products, to have a suitable main landing gear for aircraft of this size and weight. The selected system was chosen as it was at the top of the structural weight range whilst having reasonable geometry to facilitate simple integration with the existing designed airframe. The selected product is the ‘F3A - wide’ model and it is rated for design weights up to 7 kg. For use at the full design weight of 9 kg a structural analysis would be required however initial testing have a take of weight below this figure and so the system should be sufficient to take all landing loads. This system was well within budget and easy to obtain. A summary the main gear dimensions is in Table 7.14.

Table 7.14: Specifications of the Main Landing Gear

<table>
<thead>
<tr>
<th>Name</th>
<th>F3A Wide</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage width</td>
<td>110mm</td>
</tr>
<tr>
<td>Track</td>
<td>440mm</td>
</tr>
<tr>
<td>Height</td>
<td>180mm</td>
</tr>
<tr>
<td>Width @ top</td>
<td>42mm</td>
</tr>
<tr>
<td>Width @ foot</td>
<td>22mm</td>
</tr>
<tr>
<td>Sweepback</td>
<td>60 degree back</td>
</tr>
<tr>
<td>Weight</td>
<td>125g</td>
</tr>
</tbody>
</table>

Nose landing gear system has been selected and purchased due to its ease of integration. This system is simply a piece of 5/32 steel bar that has a torsion spring arrangement. This system is directionally adjustable, and has been implemented such that the nose landing gear is fixed in position.

Wheels were chosen to be of the largest available size for this application as the aircraft is expected to take off with speeds up to at the limit of this application. Wheels with diameters of 3.5 inch were chosen for this reason.

Gear location

The landing gear positions can be determined using the conceptual methodology presented in ‘Aircraft Landing Gear Design: Principles and Practices’ by Currey (1988). This method is generally used for commercial large aircraft however the principles are still applicable to this airframe.

This process involves the location of the main gear with respect to the aircraft’s centre of gravity, and uses the aircraft’s geometry to ensure functionality. The details of this analysis can be found in D. The results of this analysis has produced the locations as specified in Table 7.15.
Table 7.15: Landing Gear Locations

<table>
<thead>
<tr>
<th>Gear</th>
<th>Location from nose (m)</th>
<th>location from centreline (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nose</td>
<td>0.1</td>
<td>0</td>
</tr>
<tr>
<td>Main</td>
<td>0.539</td>
<td>±0.22</td>
</tr>
</tbody>
</table>

Landing Gear Integration

The proprietary landing gear sub systems were required to integrated into the airframe through the use of a modular structure. The opening in the bottom of the aircraft extends from fuselage station 0.180m to 0.600m. The landing gear is to be placed in the previously mentioned locations and supported by a fully removable modular structure. This structure must transfer all loads from the wheels to the ring frames of the fuselage. There are three forces to consider, the vertical force, side force and longitudinal force. Under normal landing conditions the vertical force is expected to be the largest and based on the descent or sink rate of the aircraft, for general aircraft the upper design limit is set by CASA at 3.6 m/s, the side force is expected to be the smallest while the longitudinal force is still expected to be significant. Considerations to these loading conditions were made when designing the undercarriage structure. A detailed analysis of these loadings was neither feasible or required for integration of these systems, due to the timeframe of the project.

The design of the modular system involves a modified bottom hatch. Rebates in the fuselage allow the bottom hatch to be attached and recessed in order to achieve a flush fit. Carbon fibre was used to construct the hatch to achieve the required structural integrity. The additional stiffness gained from the use of 3 layers of 200 gsm plain weave carbon fibre with an epoxy resin enabled the load from landing to be distributed into the ring frames. In addition to the hatch a tapered central beam was laminated onto the underside of the hatch. This beam was constructed from hand shaped balsa and then laminated onto the hatch with 2 additional layers of carbon fibre. This beam added rigidity to the hatch and allowed the main landing gear to be located between the two main ring frames. This tapered beam also provided an inclined mounting point which enabled the nose gear to be projected forwards of the front of the hatch by 80mm. This was necessary based on the required location of the nose gear and the need for the nose gear and main gear to be attached to a single modular structure that was completely removable. Inside the hatch small physical stops were used to transfer any longitudinal loads while the shape of the hatch transferred the lateral loads to the airframe. The undercarriage was attached to the half ring frames through the use of screws, which merely located the landing gear on the airframe. The design of the tricycle undercarriage can be seen in Figure 7.27.


Results

The landing gear has been operated on both grass and tarmac surfaces. A close inspection of the undercarriage after each flight was carried out to ensure no damage occurred. Of particular interest were the holes for the locating screws, nose gear mounting points and the internal stops to withstand the longitudinal force.

The first flight tests showed a significant deflection of the nose gear, up to 20° or 30° was observed during ground roll and take off. This deflection was also evident upon landing and resulted in the aircraft rolling over its nose on roll out. Although the nose gear design contributed to this, consultation with the pilot, inspection of the landing area and review of the landing footage indicated that soft uneven ground surface and pilot technique were two major contributing factors to the incident and it was not deemed necessary to redesign or replace the nose gear. Subsequent landings did not have the same problem and no damage was observed in the undercarriage from any subsequent landings.

Thus the landing gear has been successfully designed, integrated and tested.

7.6.2 Parachute Recovery System

A parachute recovery system has been integrated into the design of the UAV. The UAV is being designed as a multi-purpose platform with flexible operational considerations, and parachute
recovery is a method in which the operational flexibility and operational safety could be substantially improved. The coupling of this system with a proposed car launch system would increase the operational flexibility of the overall system as it would not require a traditional runway.

A proprietary system was pursued during the early stages of the design. Due to the number of commercial UAVs with parachute recovery systems it was considered a reasonable assumption that this would be a feasible option. Despite the popularity of UAV parachute recovery, a broad market survey of many UAV parachute manufacturers resulted in only one quotation. Butler parachutes, who have developed parachute systems for predator amongst many other UAV systems, quoted the project group approximately USD$18,000 for a single parachute system to meet the needs of this UAV. This expense was obviously not feasible for this UAV system and the design of a custom parachute system was investigated further.

Due to the pursuit of a proprietary parachute system the project group was forced to develop their parachute system in a short time frame. The literature review and design were conducted in a period of two days and the mechanism was developed in a period of two weeks. The whole timeframe for the development and ground testing of this system was completed in a period of three weeks, without sacrificing the result of the development.

**Literature Review**

A brief literature review was conducted on parachute design and methodologies. Despite parachutes being a commonly used system in everyday civilian and military applications there are few standardised texts and references on their design, and these are rarely available in Australia. A broad internet search uncovered a multitude of amateur rocket parachute design guide websites which use design data from credible parachute design guide textbooks. Fortunately the internet search uncovered a copy of ‘Recovery systems design guide’ by Ewing et al. (1978) from Irvin Industries inc. and commissioned by the United States Air Force, in 1978. This text was used as the principle working text in the design of this system, and a small number of amateur rocket recovery systems websites were used as supportive resources.

**Physical Components**

Parachute systems can vary is size and complexity greatly. A number of distinctive terms are used to describe the various components of such a system. Below is a variety of terms that are used in the subsequent analysis. Each of these definitions have been adapted from Ewing et al. (1978). A schematic of the constituent components is shown in Figure 7.28.
Primary parachute is the main provider of drag and is the largest component of the parachute system.

Pilot parachute, or pilot, is the parachute that is responsible for ensuring that the main parachute is deployed in a reliable manner.

Canopy is the section of material that is shaped to produce the drag inducing section of the parachute.

Suspension lines are the lines that join the canopy to the riser, or main line from the payload.

Swivel is a mechanical linkage between the suspension lines and the riser that allows the parachute to spin independently of payload.

Reefing ring is a device which is placed around the parachute suspension lines to extend the time of opening to reduce the opening loads.

Figure 7.28: Schematic of Parachute Components (adapted from Ewing et al, 1978)
Performance

As suggested by Ewing et al. (1978) the operations of a parachute is defined by the five following stages.

- Deployment
- Inflation
- Deceleration
- Descent
- Termination

Due to the limited timeframe of this project the critical area to be analysed for their performance were the safe and reliable performance during deployment, including kinematics of opening and the load generated, and the speed of the vehicle during descent and termination.

Loadings

The loads generated by the parachute are significant during deployment. Ewing, et al, present a simplified analysis method to predict the loads during operations. The major loads to be investigated are defined below.

**Descent loads** are the load generated during steady state descent and are analysed to estimate the descent speed of the vehicle during termination. This analysis is performed using traditional steady aerodynamics principles.

**Shock loading** is the load generated by the sudden impulse on the parachute during the deployment. This load is generated purely by the change in momentum of the mass of the parachute at full extension.

**Opening loading** is the load generated by the parachute during opening. This is significant as the free stream is at its highest velocity during the analysis. This loading, and the shock loading, can be seen graphically in Figure 7.29.
Parachute System Design Requirements

The design of a parachute system, as with many aerospace engineering systems, has multiple design trade offs in which the designer must seek to meet all of these requirements in order to reach a successful design. The proposed parachute recovery system must have the following design characteristics:

- Reliable deployment
- Fit inside the designated parachute volume of approximately 600mm by 100mm by 60 mm
- Suitable descent speed of the order of 5 m/s

Design of a Parachute System

The parachute bay is located in the top section of the fuselage. The entire system is required to fit within a volume 600mm long, 100mm wide and 60mm deep. In addition to this the central wing tongue is located in this volume. Currently available commercial systems were investigated however due to a lack of available systems and the cost of potentially suitable systems these were not considered a viable option. A custom designed system was investigated and deemed feasible. The limitations on the system would come from the available volume in the fuselage (600mm x 100mm x 60mm). The idea was to develop a deployment mechanism and then use the large parachute possible in order to minimise the descent rate.
The deployment system for the parachute utilises a spring-loaded system to deploy the pilot chute. This system is released by a servo that can be controlled manually from the remote control transmitter, from the auto pilot ground station or from the auto pilot itself should it lose communications with the ground station. Once activated the lock on the top hatch is released and two springs extend, removing the top hatch and deploying the pilot chute. The pilot chute then briefly slows the UAV will dragging the main parachute from the storage compartment. The main parachute is attached at four locations to the wing tongue using 200 lbs kite line, which is suitable for all operation conditions of the parachute system.

**Parachute Design**

The parachute design consisted of the design of two main systems; the primary and pilot parachute. The preliminary sizing of the primary parachute was conducted using elementary steady aerodynamics. Following this an analysis of the expected shock and opening loads was made through methods presented in Ewing et al. (1978), and is present in Appendix E. The preliminary sizing of the pilot parachute was made during this process and the results of this analysis is presented below.

**Main Parachute**

A detailed sizing process of the main parachute has been conducted using steady state aerodynamics, and presented in detail in Appendix E. The selection of an 8ft diameter parachute has been made as this was seen as a feasible trade off between internal volume limitations, low descent speed and the available parachute sizes. The results of this sizing process and the loading estimates is presented in Table 7.16 which would result in a design descent speed of 5.5 m/s.

**Pilot Parachute**

The pilot parachute has been sized using an analysis method presented in Ewing et al. (1978). With concerns of internal volume limitations the parachute size has been keep as small as possible. The resultant design has been determined to be 1 ft in diameter with a simple hexagonal planform and six parachute lines to provide attachment.

<table>
<thead>
<tr>
<th>Parachute</th>
<th>Diameter (ft)</th>
<th>Shock Load @ 120 km/h</th>
<th>Opening Load @ 120 km/h</th>
<th>C_D estimate</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main</td>
<td>8 ft</td>
<td>2.25 g</td>
<td>9 g</td>
<td>1.0</td>
</tr>
<tr>
<td>Pilot</td>
<td>1 ft</td>
<td>-</td>
<td>-</td>
<td>0.6</td>
</tr>
</tbody>
</table>

**Deployment Mechanism**

The deployment mechanism has been designed to maximise the chances of parachute deployment. This consisted of a latch actuator to release the top hatch and a front mounted spring loaded
mechanism, which ejected a small platform, that contained the pilot parachute, and jettisoned the top hatch. This system was actuated by two JS E331 servo motors, which ran in series to ensure actuation timing of both the top hatch release and the spring mechanism. Four vertical upright angle sections are used to location the drogue ejection platform and acts a physical stops such that the platform does not rotate. A vertically mounted ply board is used to separate the pilot chute from the main parachute bay, and to ensure the location of the pilot chute during flight. Several ground tests were conducted to ensure the reliability of the system which was deemed suitable. Figure 7.30 shows the CAD conceptual sketch of the parachute mechanism. This mechanism was constructed using aluminium angle and ply wood and the final product, along with the integration into the airframe, can be seen in Figure 7.31.

Figure 7.30: CAD Sketch of the Parachute Deployment Mechanism
7.6. LAUNCH AND RECOVERY SYSTEMS

(a) Deployment mechanism installation

(b) Deployment platform

(c) Top view - mechanism locked

(d) Top view - mechanism unlocked

Figure 7.31: Parachute Deployment Mechanism Components as Integrated into the Airframe

Summary

After initial investigation into potential recovery systems it was clear that a parachute system was most suitable for this UAV due to its ability to operate in tandem with a landing gear and be used in remote locations. Commercially available systems were not a viable option due to expense and availability and as a result a customised system was designed and developed. This system utilised a pilot chute, main parachute and deployment mechanism to ensure reliable operation. The system can be activated via remote control, human operator via the ground station or automatically through way point integration. Should the parachute be deployed for recovery or in an emergency it is expected that the descent rate will be approximately 5.75 m/s. In addition it is not recommended that the parachute be deployed below 200ft altitude and for preliminary testing it deployment should be conducted at relatively low speed.

7.6.3 Car Launch System

The selected car launch system is based on a tail dragger landing gear arrangement. A carriage, placed atop of a car, supports the vehicle as the UAV is driven up to takeoff velocity. The plane is pinned through the second ring frame, and the rear of the fuselage is supported in a cradle. The vehicle is free to pivot about the pins, and the height of the cradle allows the angle of attack of the vehicle to change from 0deg to 9deg, as required during takeoff. The limit of the angular movement is dictated by $\alpha_{stall_{max}} = 10\deg$.

When the aircraft reaches takeoff velocity, linear actuators retract the pins holding the plane, and the plane flies up and away from the vehicle.
The carriage places the aircraft above the ground vehicle. This feature helps reduce the buffeting experienced by the aircraft due to the disturbed flow around the car.

A sketch of the concept is shown in Figure 7.32.

![Figure 7.32: Car Launch System Concept Sketch](image)

**Specifications of Car Launch System**

The launch system was required to meet the following requirements:

- Minimal impact on flow over aircraft at takeoff
- Have significant structural integrity
- Provide scope for autonomous take off
- Simple and easy to manufacture
- Be constructed of readily available materials

**Car Launch System - Preliminary Design**

The conceptual design of the launch system resulted in the determination of suitable mounting locations for the UAV, the preliminary sizing of the launch carriage, as well as the selection of suitable release actuators.
Airframe Mounting Locations  The mounts for the UAV effectively replace the main wheels of a conventional tail dragger landing gear undercarriage. Consequently, these mounting locations needed to be positioned using accepted design practices. For tail dragger landing gear, the gear, relative to the centre of gravity of the aircraft, need to be positioned in such a way as to prevent nosing over, ground looping, and overturning. The gear must also allow the aircraft to rotate to the correct pitch attitude for takeoff. Ground looping and overturning are not an issue for the car launch system, as the carriage constrains the movement of aircraft in roll and yaw directions. Consequently, only nose over and takeoff attitude considerations had to be made.

To prevent nose over, whilst allowing an aircraft to move to correct takeoff pitch, Raymer (2006), suggests that the longitudinal position of the gear with respect to the centre of gravity of the vehicle satisfies $16\,\text{deg} \leq \zeta_{\text{main}} \leq 25\,\text{deg}$, (Figure 7.33).

![Figure 7.33: Recommended Longitudinal Position of Gear with respect to Vehicle Centre of Gravity, (Adapted from Raymer, 2006)](image)

To prevent the need to add additional structure to the airframe, the mounting points for the aircraft were positioned 20mm above the floor of the aircraft, in the second ring frame of the vehicle. This arrangement, relative to the designed centre of gravity location of the aircraft is shown in Figure 7.34.

![Figure 7.34: Position of Airframe Mounting Points to Vehicle Centre of Gravity](image)

To check the suitability of the mounting positions, $\zeta_{\text{main}}$, was calculated for the aircraft in each of its four configurations, (Table 7.17).
Table 7.17: Angle between Airframe Mounts and Centre of Gravity for Different Flight Configurations

<table>
<thead>
<tr>
<th>Configuration</th>
<th>$\zeta_{\text{main}}$ [deg]</th>
</tr>
</thead>
<tbody>
<tr>
<td>RC flight test</td>
<td>42.3</td>
</tr>
<tr>
<td>Autopilot test</td>
<td>43.7</td>
</tr>
<tr>
<td>Camera test</td>
<td>43.7</td>
</tr>
<tr>
<td>Mission</td>
<td>45.0</td>
</tr>
</tbody>
</table>

The value of $\zeta_{\text{main}}$ for all configurations was found to exceed the maximum recommendations. This indicated that if this launch system was a conventional tail dragger undercarriage, the aircraft would have difficulty in rotating to the correct pitch attitude during ground roll. However, unlike a conventional tail dragger, the design of the car launch frame ensures that the main wing is always at an angle of attack less than $\alpha_{\text{stall, max}}$. Thus, it was believed that the values of $\zeta_{\text{main}}$ were not large enough to prevent the aircraft from rotating to the correct takeoff attitude during ground roll, for all flight configurations. Thus the positions of the mounts was deemed acceptable, and validation of the positions would be obtained through an extensive testing program utilising the aircraft and car launch mechanism.

**Preliminary Sizing of Launch Carriage** The preliminary sizing of the launch carriage can be seen in a three-view drawing in Appendix K. The planform sizing of the carriage was based on the dimensions of the airframe, and the size of a typical car in which it is to be used on. The height at which the airframe is mounted within the carriage was based on recommendations from Ahmed & Ramm (1984). For a medium sized family car, the flow over the car should be sufficiently settled at a height of 500mm above the roofline. Consequently, the launch carriage places the UAV at this distance above the roof of the car. This would be verified through extensive ground testing of the system.

The launch carriage has been designed using 25mm x 25mm x 3mm mild steel tubing. This choice was based on the availability of the material, the ease of its manufacture, and the strength of the material, three reasons, which satisfy the specifications of the airframe.

**Actuator Selection** Linear actuators were selected as the launch mechanism for the system. One linear actuator is used to pull each of the two pins used to mount the airframe to the car launch system. These actuators were required to:

- Provide 50N of pull force, (determined from simple static analysis of the pin/mount hole system).
- Provide high speed actuation

Based on these recommendations, two LINAK LA121000-10501210 linear actuators were selected. These are capable of

- A maximum pulling force of 200N
• An actuation speed of 28mm/s under full load, (LINAK, 2007).

Hence these actuators have specifications, which were deemed sufficient for the task.

### 7.7 Summary of Airframe Detailed Design

The result of the airframe detailed design was an airframe capable of meeting all design requirements. A CAD image of the final design is included in Figure 7.35 below. The detailed drawings of the airframe and its structural components can be found in Appendix K.

![Figure 7.35: CAD of the Final Airframe Design](image-url)
Chapter 8

Manufacture

Constructed primarily from composite materials, the UAV utilized several different methods throughout the manufacturing process. The availability of these methods, as well as the skills and tooling available, dictated to an extent the design of the wings, fuselage, and other critical components. In addition to the several techniques used during construction, three main areas were considered throughout the manufacturing process. These three main areas comprised of the wing construction, the fuselage construction, and the internal structure construction. Upon manufacture of each of the aforementioned areas, the sections were then integrated through an assembly process.

8.1 Available Manufacturing Methods

The method of manufacture for all components of the aircraft was driven by the availability of tools, skills and experience. The manufacturing techniques used for each stage of the manufacturing process are discussed within the relevant sections. However, presented below are details of the methods, which could have been used to manufacture the plane.

The most automated tool available for the manufacturing process was a CNC milling machine with up to 6 degrees of freedom. This machine has the ability to machine shapes generated by a solid modelling CAD package, with restrictions placed on maximum dimensions. The tooling used for this machine does not allow for tight radius corners that may be required for certain components. A laser cutting machine was also available that has the ability to cut tight radii and give a smooth finish. The laser cutter only operates in 2 degrees of freedom.

Manufacturing the fuselage from composite parts can be done in two ways, both requiring the manufacturing of a plug or ‘dummy’ of the shape. Female moulds can be produced from the plug, leaving both the plug and moulds for reuse, or the plug can be skun with composite material and then ‘burnt out’ with a liquid solvent, leaving the skin intact. Laying the composite material can be done by either a wet lay up method or with the use of preimpregnated cloth. The later of these two consists of cloth preimpregnated with the required resin, curing of this cloth is done in a ‘hot box’ where the temperature is raised and held constant for a period of time. A wet lay
8.2 Wing Construction

As discussed in Section 7.4.4, the wings are comprised of an isolite foam core and covered with a fibreglass skin. Contained in the foam core are four unidirectional carbon fibre spars, extending from root to tip at 15% and 78% of the chord length from the leading edge of the wing. These spars, connected to wing boxes at the root, allow the wings to connect with the fuselage and internal frame structure, as can be seen in the schematic in Figure 8.1.

Figure 8.1: Wing Attachment CAD Schematic (Top View)

**Tongue Construction** Manufactured from a 6mm sheet of 2024-T3, the component was designed to allow for both sections to be laser cut from one sheet of material. Subsequent to the sections being laser cut, the components were hand finished in order to ensure a smooth surface with an absence of sharp corners.
Wing Box Construction  The concept of the wing box was to create a sleeve that could be set into the wing, and allow the tongue to be easily inserted or removed from the wing whilst retaining a tight fit. To achieve this, the boxes were manufactured by wrapping 200gsm 0/90° carbon fibre around the outboard 135mm of each tongue spar in a wet lay up fashion. The manufactured wing boxes were then installed into the wing cores, (see below.)

Wing Construction  The foam core of the wing was produced using a hot wire cutting method. Foam blocks were cut to the appropriate size for the wing, and then cut with the assistance of root and tip airfoil profiles, cut from laminate using a CNC controlled milling process. To achieve the desired 2° of washout in the wing, the tip template was rotated down by 2°.

The hot wire cutting method required the speed of the wire at the root to be faster than the speed at the tip. Incorrect wire speeds created drag in the wire, resulting in a poor finish on several wing prototypes. However, this problem was resolved through practice, enabling a good quality finish on the final, used set of wings.

The sleeves, wing boxes and spars were installed in the foam in a single process. In order to achieve the precise placement of all components, a jig was constructed from medium density fibreboard. This jig permitted the wings to be fixed into position with a zero sweep angle and zero angle of attack while the resin set, in addition to allowing the tongue to be positioned at the correct height. Prior to the assembly of all components, channels were carved out of the wing foam to allow for the wing spars to be recessed below the surface. In addition, the foam was removed from the root of each wing for receive of the wing boxes. The spars were glued into these recesses using epoxy resin, and the wing boxes were bonded to the wing spars and core using mixture of short length carbon fibre mat and epoxy resin to fill the void from the removed core foam. Figure 8.2 highlights the manufacturing of the wing. After curing, the wings were separated, and the wing boxes around the wing roots were hand finished to remove resin, which had seeped out during the manufacturing process.

Figure 8.2: Wing Attachment Construction Method
Once the final shape of the wings had been hand finished, the aileron was cut from the foam trailing edge and additional plywood ring frames installed between the four spars to accommodate the aileron servo. Subsequent to this, the wings were laminated with 3 layers of 0/90º 85gsm fibreglass cloth, with the middle layer of cloth offset by 45º to increase the torsional resistance of the material. Peel ply was used on the surface of the fibreglass during the curing process, to improve the resin to fibre ratio of the parts, and hence the finish and integrity of the wing structure.

8.3 Fuselage Construction

The shape of the fuselage was determined due to the volume requirements necessary to carry the control system, power plant and payload. Designed through the aid of a CAD package, the fuselage incorporates the main body of the aircraft and empennage section, along with the vertical stabiliser and the nose cone. Construction started with a foam plug, used to produce female moulds from which parts could be generated. These parts were then assembled with the structural components through the use of jigs and framework. The use of female moulds was elected as the best manufacturing process for the UAV, as it enabled parts to be reproduced both time and cost effectively in the event of an accident or manufacturing problem. In addition to efficiency, the female moulds enabled the production of multiple top and bottom hatches for the aircraft.

The profile for the mould plug was milled out of styrofoam. Due to restrictions on size and the number of degrees of freedom of the milling machine, the profile was divided into 6 parts. The 6 milled parts were then bonded together, and laminated with lightweight fibreglass, to provide the desired smoothness for accurate mould development.

Using the recommendations of the composite technician involved with the construction of the aircraft, a four part mould was developed for the fuselage. This mould system, (Figure 8.3), was selected primarily due to the ease of the removal of parts, comprised of:

1. The plug

2. A bottom mould

3. A top mould

4. A two-part vertical stabiliser mould

5. (The wings are also shown in Figure 8.3, but are not part of the mould system).
The bottom mould, running the full length of the aircraft, was designed to expand across the fuselage to the point where the battery board was to be located. This arrangement allowed the battery board to assist the joining of the top and bottom halves of the fuselage. Similarly, the top mould was designed to extend from the nose of the aircraft to the vertical stabiliser, and expand across the fuselage to the location of the battery board.

The vertical stabiliser was designed to be manufactured in two halves, and joined to the other two sections of the fuselage. This was done to ensure the shape of the vertical aerofoil was preserved during manufacture.

During the mould manufacturing process care was taken to ensure that the fibreglass did not ‘relax’ and distort the shape of the moulds. This was done by using sufficient quantities of material in the moulds, and also by installing flanges and framework to support the shape as the moulds were set.

The top and bottom parts of the fuselage were constructed using a wet lay up technique. Both parts were constructed of three layers of $0/90^\circ$ 85gsm cloth with peel ply, which helped control the resin to fibre ratio of the parts, and hence their finish and integrity. Once cured additions such as a honeycomb core in the nose section and a foam sandwich in the walls of the top part were added to improve the strength and integrity of the component. In addition, one layer thick, carbon fibre ring frames and longerons were added to the empennage and fuselage for structural integrity, particularly in buckling. Details of these additions can be found in Table 8.1.
8.3. FUSELAGE CONSTRUCTION

Table 8.1: Load Bearing Structural Additions to Fuselage Design

<table>
<thead>
<tr>
<th>Material</th>
<th>Where Used</th>
<th>Details of Use</th>
</tr>
</thead>
<tbody>
<tr>
<td>Honeycomb core</td>
<td>Nosecone</td>
<td>2mm foam sandwiched between 3 external layers and 2 internal layers of 85gsm fibreglass</td>
</tr>
<tr>
<td>Closed cell foam</td>
<td>Fuselage sidewalls</td>
<td>3mm cell, sandwiched between 3 external layers and 2 internal layers of 85gsm fibreglass</td>
</tr>
<tr>
<td>Carbon fibre</td>
<td>Empennage</td>
<td>20mm wide, single layer carbon fibre used for two ring frames in empennage of aircraft</td>
</tr>
<tr>
<td>Carbon fibre</td>
<td>Fuselage/Empennage</td>
<td>30mm wide, single layer carbon fibre used for longeron extending entire length of aircraft</td>
</tr>
</tbody>
</table>

The vertical tail section was manufactured using the same wet lay up as the other components of the empennage. However, the port and starboard components were joined while still in the mould using an epoxy filler and fillets on two vertical ring frames, (Figure 8.4). These frames were manufactured from 2mm plywood with a layer of carbon fibre laminated on each side. Once joined the part was assembled to the rest of the fuselage.

![Figure 8.4: Tail Construction](image-url)
8.4 Internal Structure Construction

The internal structure of the UAV is comprised of the ring frames, the centre board, and the nose cone and tongue. The structure has four ring frames, two of which are connected to the tongue and all four of which are connected to the centre board.

The ring frames vary in size and material, depending on location. Details of the design of the ring frames is detailed below:

- Front ring frame: Two single layer 4.5mm plywood frames bolted together using 4mm mild steel bolts and nylock nuts. One frame half is attached to the fuselage and battery board, the other is attached to the nose cone. This ring frame provides the necessary support for the nose cone and its associated loads. The frame halves are designed as bulkheads, which limit heat transfer from engine to the on board electronic systems, and also provide additional structural integrity at the front of the aircraft. Two small holes in the bulkhead provide passage for battery wires to motor.

- Second ring frame - 3 layers of 6mm plywood, bonded with epoxy resin. Used to transfer the loads from the front wing tongue, the second ring frame is laminated with carbon fibre to improve structural integrity.

- Third ring frame - 2 layers of 6mm plywood, bonded with epoxy resin, and laminated with carbon fibre to improve structural integrity. Smaller thickness than second ring frame due to reduced load carrying requirements.

- Rear ring frame - Single layer of 6mm plywood. Required to reinforce fuselage structure.

The centre board is a solid piece of plywood laminated with 85gsm fibreglass on each side. All ring frames are installed on the battery board using a notch arrangement, (Figure 8.5), allowing load to be distributed over a compression connection, in addition to fasteners and composite fillets also used to connect the internal structure to the fuselage.

Figure 8.5: Internal Structure Layout
8.5 Horizontal Stabiliser Construction

The horizontal stabiliser was constructed in a similar manner to the wing structure. The port and starboard sides of the horizontal stabiliser were hot-wire-cut from styrofoam, connected at the root, and covered in a single layer of 200gsm $0/90^\circ$ carbon fibre, (Figure 8.6).

Prior to laminating the horizontal tail, the elevators were cut from the foam and hinged using a simple axle and sleeve arrangement.

![Figure 8.6: Horizontal Tail Construction](image)

8.6 Assembly of Structural Components

The next phase of construction involved the assembly of the airframe structural components. The order of assembly followed is presented chronologically in this section.

**Installation of Ring Frames and Centre Board into Bottom Fuselage Section**  The pre manufactured plywood structure consisting of the ring frames and centre board was attached to the lower fuselage using composite fillets.

**Installation and mounting of wings**  A custom built jig, was used to install and mount the wings, (Figure 8.7). The jig ensured that the wings were mounted symmetrically, with zero yaw and roll angles. The correct angle of incidence was achieved through trigonometry and linear measurement. The wings, as mentioned, are attached to he fuselage via the aluminium tongue structure, which is mounted to the second and third ring frames using 4mm diameter mild steel bolts and nylock nuts.
CHAPTER 8. MANUFACTURE

Installation of Top Fuselage Section  After removal from the mould, the top hatch was cut from the part to allow access to the inside of the aircraft. This hatch, required for the parachute system, also allowed for improved quality control, as it provided access to the ring frames, allowing inspection and repair the composite fillets used to attach the component to the ring frames, (Figure 8.8).

In the empennage section the top and bottom parts were joined with the aid of closed cell foam. This foam allowed the top and bottom parts to have a larger join area, and hence improve structural integrity through this region.

Separation of Nose Cone from Moulded Fuselage Components  After joining of the top and bottom fuselage sections, the nose cone was removed. The nose cone was separated to allow access to the inside of the nose cone for easy mounting and servicing of the motor components. To provide a strong point to mount the motor to, the front of the nosecone was reinforced with eight layers of 200gsm $0/90^\circ$ plain weave carbon fibre.

The nose cone is attached through the front ring frame with four mild steel, hex head, 4mm bolts.
Installation of Horizontal and Vertical Stabilisers  A custom built jig was used to install the horizontal and vertical stabilisers. The purpose of the jig was to ensure that the stabilisers were installed with zero yaw, roll, and pitch angles, as required. The stabilisers were attached to the empennage using epoxy fillets and reinforced with composite cloth.

Final Structural Assembly  Once all components were fixed in place, a final, external structural connection was made between the top and bottom fuselage parts. During the moulding process a rebate was made in each part to allow for this. In this rebate, a strip of plain weave carbon fibre was placed. This carbon fibre strip extends the full length of the fuselage, and was used to improve the integrity of the join of the top and bottom fuselage sections.

During the moulding process, a rebate was also made into the upper fuselage section and the vertical stabiliser. Similar to the joining of the fuselage, a strip of carbon fibre was used in this rebate to improve the connection between the empennage and the vertical stabiliser.

8.7 Aircraft Finishing and Fitting

Once fully assembled the aircraft was painted with a two-pac finish, for aesthetic reasons and also to protect the carbon fibre from ultra-violet radiation.

After painting, the final stage of the manufacture of the aircraft was to install the sub-systems of the plane, including the control, communication, propulsion, and imaging systems. The installation of these systems was performed in stages. The timing of these stages was based on the pending flight test schedule.

8.7.1 Control System Installation

The physical installation of the control system required careful design, to ensure the effectiveness of the components. The following considerations had to be made when selecting the position for the control system components:

- The placement of the autopilot gyroscopes and accelerometers near the centre of gravity of the airframe, and central to the airframe
- The ease of access required to various components, in particular the servo board
- The need for an easily accessible a ‘kill switch’ on both the autopilot and the servos.

Taking into account the above considerations, an easily accessible area of approximately 360 x 110 mm on the centre board, beneath the wing tongue structure, was thus allocated to control systems electronics (including its batteries). Figure 8.9 illustrates the installed positions of the control systems equipment within the fuselage. of the components in use, which is followed by a picture of this configuration.
It should be noted that during flight, the batteries are covered by the ejection mechanism for the pilot parachute, and the main parachute covers the rest of the electronics bay.

Some of the sensors on board the autopilot required additional installation, which is discussed in the following sections.

**Pressure Transducers**

The two pressure transducers on board the autopilot are used for obtaining static and dynamic pressure readings, as required by the autopilot for accurate navigation. As the airframe is not airtight, the pressure transducer for the static measurement needed no additional pressure sensing componentry, such as a static port. A 1/8inch brass pitot tube located on the leading edge of the port wing is connected to the other pressure transducer, via silicon hose running through the interior of the wing, and obtains dynamic pressure readings during flight. The pitot tubing is located 1ft away from the fuselage, and extends 4inches ahead of the wing, so as to minimise the effects of prop wash and the flow over the aircraft on the pressure measurements.
obtained. These positions were determined from recommendations from (UAV-Flight-Systems, 2006). The final configuration is shown in Figure 8.10.

![Figure 8.10: Pitot Tube Installation](image)

**GPS Antenna**

The GPS antenna is mounted on top of the carbon fibre hatch with two screws. The carbon fibre acts as a ground plane for the antenna, and helps reduce the multi-path effects, from the reflection of satellite signals off of surfaces such as the ground, which can inhibit the performance and accuracy of the GPS system, (Krantz et al., 2001).

The use of carbon fibre as a ground plane is not commonly recommended by GPS manufacturers as the electrical resistance is of order one hundred times greater than that of copper plating. However, utilising the carbon fibre hatch which is already present on the airframe creates a much larger ground plane than is normally used, and thus it was decided to test the effectiveness of the carbon fibre ground plane experimentally. The performance increase in the GPS system obtained through the use of the carbon fibre ground plate was checked experimentally. The results of the test, shown in Table 8.2, revealed two things.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Time to lock</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>No ground plate</td>
<td>X</td>
<td>Could not receive a lock</td>
</tr>
<tr>
<td>Carbon fibre hatch ground plate</td>
<td>40</td>
<td>Was not able to break lock with large lateral and rotational movements of the receiver</td>
</tr>
<tr>
<td></td>
<td>Did Not Test (DNT)</td>
<td></td>
</tr>
</tbody>
</table>

The first is that a ground plate needed to be used so as GPS lock could be obtained. The second was that the carbon fibre ground plate was effective in obtaining a GPS lock, and could obtain one within 40 seconds, which was sufficiently quick. In addition, the GPS lock, once obtained, was not lost under large linear and rotational movements of the GPS antenna and ground plate.
Hence it was found that the designed GPS antenna and carbon fibre ground plane system, (Figure 8.11), was adequate for the navigation of the aircraft.

![Carbon fibre hatch with GPS antenna installed](image)

![Close up of installation](image)

![Curvature of the carbon fibre hatch](image)

Figure 8.11: GPS Antenna Integration

**Servos Installation**

The servos used to control the ailerons were, as specified, were placed in inside the wing structure, near the control surfaces. After the skinning of the wings, a cavity was made for the mounting of the aileron servos within the wing core. The servos were mounted to the installed servo ring frames, and a small plastic cover was then constructed to cover each servo.

A heated rod, was used bore holes in the wing, so as electrical connections could be made between the autopilot system and the wing mounted servos. In the port wing, this hole also provided the passage for the hose connecting the pitot to the dynamic pressure transducer.

The servos for the rudder and elevator were located on the center board. Long pushrods constructed of mild steel and balsa wood, along with proprietary control horns, were used to actuate these control surfaces.
8.7.2 Communications Equipment

Three sets of communication gear needed to be installed in the airframe. These were the remote control gear, the autopilot gear and the imaging system gear. These systems were mounted in the remaining available space on the centre board.

In addition, all three sets of communication equipment required mountings for antennas. A hole just aft of the port wing provides a passage for the RC aerial to exit the inside of the airframe. This is then free to trail behind the plane, in accordance with common remote control aircraft practice. The other two antennas were mounted externally on the underside of the empennage using rubber grommets.

8.7.3 Propulsion System

The motor was mounted in the front of the nose cone, through the carbon fibre reinforced front. Due to the reaction torque generated by the motor, the motor was mounted with a 1.75° offset to starboard. This offset also assisted in minimising the negative effects propeller wash creates to the effectiveness of the control surfaces of the aircraft.

The electronic speed controller and the diode junction used to connect the batteries together, were mounted in the nose cone. These were positioned close to the motor, as recommended by ModelMotors (2006) and Hacker-Brushless-Motors (2006).

The propulsion system batteries were mounted on the underside of the centre board. The batteries were arranged in two bays, each carrying four batteries. The batteries were fixed to the battery board using disposable zip ties.

8.7.4 Camera Mount

The camera is mounted to the rear, underside of the battery board. At this location a 50mm x 50mm window has been cut in the fibreglass fuselage. The camera is mounted on a removable platform containing reusable aluminium brackets that mount the camera such that it is tilted in the forward direction by 30°, as specified by the design of the imaging system. This mount and its location in the aircraft is shown in Figure 8.12.
The camera was mounted with foam, to help isolate the camera from airframe vibrations. It was decided that further work on the isolation of the device from vibrations would only be done if flight tests revealed that it was required. As will be discussed in Section 11.9, no additional vibration isolation was found to be required.
Chapter 9

Electronic Systems Operation

This section of the report will detail the operation of the control and imagery systems, detail the software based integration of the control system into the airframe of iSOAR, and detail a concept of operations and how it is applied for a typical civil UAV application such as the ARCAA competition requirements (ARCAA, 2007).

The control system is intended to provide for close range manual control of the UAV, as well as providing for long range autonomous flight between set waypoints. Autonomous take-off and landing are considered beyond the scope of this system. The system described in this section has been successfully configured for autonomous straight and level flight, but due to the project's time constraints and problems experienced with communications interference (refer to Section 11.10), has not shown waypoint following in practice. However, it is noted that provision for this has occurred, and additional flight tuning of the autopilot is all that is required to reach this further goal.

The imagery system is intended to provide for streaming video imagery from the UAV to a user display at the base station. Autonomy of the searching routine in any regard is considered beyond the scope of this system.

9.1 On-board Components

Figure 9.1 details a schematic of the control systems on-board components. The on-board imaging system consists simply of a battery, camera and transmitter. Onboard the UAV the camera and transmitter are connected to each other and to the power supply. The power supply is a 1500mA 11.8VDC lithium polymer battery.


9.2 **Ground Station**

In terms of the control system, the ground station is the user interface for in-flight control and/or monitoring of the aircraft. Three methods are used for communication and control of the autopilot. These are; utilising long range RF modems and the in-flight software package ‘Horizon’, implementing RC control, or directly connecting through a hardline. In terms of imagery, streaming footage is provided to a laptop screen for viewing.

The control systems long range RF modem is connected directly to a laptop, via a USB to serial cable. The layout of the control system’s in the ground station is shown in Figure 9.2. The Micropilot 2028g utilises a software package called ‘Horizon’ for in-flight communications and data transfer (Micropilot, 2004). This program manages all communication between the autopilot and the computer on the ground. Developed exclusively for the Micropilot autopilots, this software is not able to be modified and as such it is required that the operators work within the limitations of the program. This is one of the disadvantages of purchasing a complete autopilot system, however the program itself is fairly comprehensive and thus there are not too many limitations for most operations.
Coupled with this program, the RC transmitter allows for close range control of the aircraft. One of the auxiliary channels on the transmitter is used to switch between RC and autopilot control. When the UAV flies beyond the range of the RC transmitter, the autopilot will respond in the manner programmed by the autopilot RC dropout failsafe routine. This is usually set to return control to the autopilot.

The third method available to communicate to the autopilot is through the serial-to-JR cable which connects directly to the autopilot in place of the on board modem. This is for file transfer and autopilot monitoring on ground when power for the RF modems is not available, or if transmission errors are occurring between the RF modems.

For all in-flight operations, the combination of the laptop/RF modem connection, and the use of RC control is implemented. The 'Horizon' ground station interface software, shown in Figure 9.3, contains four main sections, a left bar, right bar, main screen, and sensor displays.
The left bar controls the communication between the ground station and the autopilot. The right bar can display detailed information about individual waypoints, and be used in setting new waypoints in-flight. The map display is user configurable. That is, user maps can be uploaded to be used as a background on which to display mission waypoints. This map can be configured to either remain in the same position or follow the aircraft as it moves. Size of area shown at one time is also configurable. The standard instrument/sensor display contains an altimeter, speed and pitch/roll display, as well as a throttle, readout (% of full throttle) and battery readings for both the servo and 2028 battery supplies.

For viewing of the streaming imagery, a the receiver provided by WirelessVideoCameras (2005) is coupled with a directional grid antenna mounted at a height of 4m. This mounting was achieved using a modification of the parachute test rig described in Section 6.3. This was considered sufficient for the purposes of the specified mission profile of the Outback Challenge, as all of the streaming imagery to be considered would be coming from a large distance away and hence occur over a small angle of operation. This allows for the use of a static stand with the directional antenna, however, for mission profiles occurring with surveillance occurring closer to the base station and hence over a larger angle a stand capable of rotation in both the horizontal and vertical planes would be desired. To suit the versatility intended of iSOAR, this stand would also need to be in small enough sections to allow ease of portability. Initial efforts occurred in this area, however a design was not finalised.

A high speed Belkin USB 2.0 DVD creator cable is then used to allow the video stream to be
viewed on a laptop. This system is flexible to NTSC/PAL format users and is compatible with VideoStudio software which allows live viewing and capturing of the live stream. This receiver and laptop are powered off of the control systems inverter.

9.3 Operation

9.3.1 Control Systems

Two methods of controlling iSOAR during flight are available; through the use of the short range RC unit, or via the autopilot. Control is switched between each mechanism by an auxiliary servo channel on the RC transmitter, or through a software override occurring on the autopilot. Whilst in RC operation, flight is as per normal, with the exception that the signals are routed through the autopilot. When in autopilot operation, the plane is flown by the consideration of two separate onboard files; the vrs file and the flight file. Of these, the flight file can be one of either a .fly file, or a .wpt file.

VRS Files

The *.vrs file is the file which the autopilot utilises in order to define the parameters of the aircraft and the desired parameters of normal flight. This file is in effect the configuration file, which the autopilot references in order to understand:

- what the permanent physical configuration of the aircraft is, and
- what default flight parameters to use when no other is specified.

This file is set-up during the initial configuration of the autopilot to the airframe, although it is possible to modify at a later time either directly or through an explicit line in the flight file.

The physical parameters of the aircraft define settings such as the servo home positions, throws, and expected pulse widths, which limit the actuation of the control surfaces and auxiliary channels to suitable values. The gain settings on the PID loops which implements the flight paths are also set in the VRS file, which will be discussed further in Section 9.4.

The VRS file also contains a series of default parameters, such as the cruise speed, climb speed, descent rate and default level flight mode. These parameters define the way that the autopilot will attempt to fly the UAV when no overriding command is used. The values chosen for the initial flights of the autopilot are effectively arbitrary, as all relevant details to the initial tuning flights are specified in the code. Telemetry from an RC flight is then used to quantify real values.

FLY Files

The *.fly files are a form of flight file which allow the programming of the flight path via a custom programming language developed specifically for the Micropilot autopilots. Figure 9.4
shows the usual layout of a flight file. An example of this is shown in Appendix F which is a flight file set to fly a simple course defined by 4 waypoints at the altitude that autopilot control was engaged. The file contains three examples of defining custom payload code, and one example of implementing an in-flight failure pattern.
9.3. OPERATION

Figure 9.4: Flight File Structural Layout
WPT Files

A *.wpt file fulfills the same purpose as the .fly files, but allows for a quicker and simpler programming of less complicated flight paths. A waypoint following method can simply be used where the operator simply needs to define a series of waypoints (three dimensional location in space) which the autopilot will then attempt to fly through using the default parameters as defined in the vrs file. It is also possible to ‘couple’ a waypoint file with the header and footer from a .fly file, enabling the re-use of payload definitions and error handlers for multiple files. It is this method of programming that would be most suitable for many civil applications as it allows a simply and quick method of defining flights, along with the capabilities of containing standard payloads.

9.3.2 Imagery Systems

The imagery systems have been set-up as effectively a static system, with only a single mode of operation. The connections of the imagery systems is shown in Figure 9.5. Once power has been applied to the on-board and ground station components, streaming video footage can viewed and recorded on a laptop. This is provided that the image transmission occurs from within the receiving angle and range of the directional antenna. Due to the limited testing phase this parameter was not determined as it is a function of both angle and range. Further testing of this parameter is recommended.

![Figure 9.5: Imaging System Setup](image)

9.4 Control Systems Implementation

The autopilot implements the control, stabilisation, and navigation of the UAV using 12 separate PID loops, which alternate taking control of the aircraft depending on the situation; climbing, banking, level flight etc (Micropilot, 2007b). The layout of these loops and their relationship to control surfaces for a conventional airframe is shown in Figure 9.6, which displays the cascaded nature of some of the loops. These cascaded loops are used to satisfy two coincident criteria.
The first criteria is related to the position or orientation of the UAV in relation to where it wants to be. This PID loop effectively plans a path as to how it is to move to reach the defined goal. The second criteria is that related to the actual operation of the control surface and how it is to achieve the desired path. This PID considers an instantaneous position on the planned path and modifies the actuation of the control surface in order to achieve this. In summary, where cascaded loops are present, the first considers the current ‘long-term’ goal of the UAV (set altitude, position etc.) whereas the second loop uses that path to define the control surface position in that instant of time.

![PID Loop Operation for a Simple Level Flight Scenario with No Target Heading](Micropilot, 2007b)

It is also important to note that feed forward terms are present in some of the loops. This term effectively anticipates the need for a control input before the need is evident in terms of direct sensory input. For instance, this term is used coordinates the efforts of the rudder and ailerons to ensure smooth banking.

Descriptions of the individual PID loops are given in Appendix G, an extract from the Micropilot 2028g autopilot manual (2007).

### 9.4.1 Tuning

In order to configure the autopilot for a particular airframe, a tuning procedure is required in order to adjust the gains of the PID loop. Research showed the most commonly used method for commercial autopilots is that of manual tuning (UAV-Flight-Systems, 2006) (Procerus, 2007b) (Micropilot, 2007b), whereas the other method sometimes employed involves the generation of an accurate simulation of the airframes flight dynamics. The two main advantages of the manual
procedure over the simulation approach relate to the development time and the accuracy of the simulator required. The Micropilot autopilot is supplied with data for a 40 sized RC trainer, which, while not perfect, is intended to provide a starting point for manual tuning and cut down on tuning time (Micropilot, 2007b). Their claim of a possible tuning time of around a week was supported by the aerobotics group at Monash University (Egan et al., 2003) in their use of the 2028g autopilot. However, it must be noted that the use of an electric motor does limit the possible flying time of the UAV in one day, due to the need to recharge batteries. Also noted is the comment by Egan et al. (2003) that “considerable knowledge and experience in flying model aircraft” is required during the tuning process. With this noted though, an expected tuning time of less than two weeks seemed attainable. In contrast, Egan et al. (2003) noted that tuning PID gains from a computer based model, while a successful method for larger aircraft, proved quite difficult for aircraft around 7kg. As the iSOAR airframe was designed to be around 9kg, it was noted that similar issues would most likely occur. This would not prohibit the tuning of the autopilot from the simulator, however it may degrade the accuracy of the results such that the gains produced are only marginally better than those supplied with the autopilot. Taking these factors into consideration, it was decided that a manual tuning procedure would be used for the integration of the Micropilot 2028g autopilot into the airframe. Unfortunately due to unforeseeable delays, the tuning of the autopilot was delayed such that the use of a simulator may have been a more viable approach, but the issues noted above would have still posed problems.

9.4.2 Process

The manual tuning process used to begin tuning the autopilots control, stabilisation and navigation loops involves the setting of flight path in order that a minimal number of untuned PID loops are in operation at any one time. Control of the UAV is then switched from RC to autopilot mid-flight, and the behaviour of the UAV monitored and used to adjust gains as necessary. A simplified tuning method is used for the inner stability loops (those directly affecting the actuation of the control surfaces), where gains of the proportional and differential loops are reduced by 25% when oscillations of frequency > 1 Hz are occurring, and the integral gain reduced by 25% for oscillations of frequency < 1 Hz. Although simple, this procedure does allow basic stability of the loops to be obtained relatively quickly. Although the gain levels achieved will not be optimum, it is important to consider what the purpose of the control loops are, and what impact this has on the desired gains.

It has been noted in the mechanical design of the plane that a statically and dynamically stable configuration has been achieved. That is; that the system will create forces in order to reject any small disturbances of its own accord when in a steady state flight condition, and that the amplitudes of motion due to these disturbances will decrease over time (Roskam, 2003). Hence in setting the gains of the PID loops, the intent is not to create stability an unstable system, but rather to enhance the stability of the system, and allow for control of the system. With this intent it can be noted that the gains themselves are not required to produce critical damping of the system, but rather simply need to be suitably large in order to ensure that a reasonably fast level of control can be achieved. Gains that are set low will slow the systems response to commands,
9.5. CONTROL SYSTEMS SAFETY

but should not create any other undesirable effects upon the system. In contrast, gains that remain too high may cause the system to become dynamically unstable, amplifying disturbances that act upon the system. Hence it is evident that using a simplified tuning procedure that will produce gains that are lower than critical is indeed valid approach in this case.

In tuning the outer loops, that is; those used to consider the current set-point of the UAVs position in space, various methods are used which are all variations on the process detailed above. The plane is flown in RC mode, control switched to autopilot, and the behaviour monitored and used to adjust gains. The process of tuning all PID loops is detailed in Appendix H.

9.4.3 Elevator, Rudder & Aileron Inner Stability Loops

The three inner stability loops of the aircraft controlling elevator from pitch, rudder from Y accelerometer (yaw) and ailerons from roll have been tuned. This has resulted in the gains represented graphically in Figure 9.7, for the unit disturbances described in Table 9.1.

![Figure 9.7: Tuned Inner Stability Loop Gains](image)

Table 9.1: Unit of Disturbance Considered for each Control Loop

<table>
<thead>
<tr>
<th></th>
<th>Proportional</th>
<th>Integral</th>
<th>Derivative</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aileron / Elevator</td>
<td>deg</td>
<td>deg-s</td>
<td>deg/s</td>
</tr>
<tr>
<td>Rudder</td>
<td>m/s²</td>
<td>m/s</td>
<td>m/s³</td>
</tr>
</tbody>
</table>

9.5 Control Systems Safety

With the development of control systems for any moving vehicle, safety is always a high priority. This is even more so with an autonomous UAV, where the implications of an airborne vehicle coupled with the lack of opportunity for useful human intervention create a large safety risk. In
the long term development of an autonomous UAV such as this an independent safety system should be developed, with the ability to take control of the control surfaces and implement termination of flight in a safe manner in the occasion of failure of the primary system. However, due to the time constraints on this project it was decided to implement instead a software based system. The operation and limitations of this system, along with its limitations, are discussed in the following subsections.

9.5.1 Significance

A civil capability assessment conducted by NASA in 2006 (Cox et al., 2006), preceded by a conference on unmanned vehicles 'UV Europe' in 2005 (Kochan, 2005), highlighted the need for proven safety features onboard a UAV as one of the major barriers to civil UAV development. Both occasions cited that one of the major items currently constricting the development of civil UAV use was the restriction and/or lack of current airspace regulations relating directly to UAVs, and that this situation would likely not improve until reliable safety systems were developed, including systems to:

- share airspace with other aircraft, and
- avoid imminent collisions with other aircraft

Study into this area highlighted the most probable solution a three phase effort, illustrated in Figure 9.8.
The safety system detailed in this report attends to the first phase. A similar methodology however, could also be applied to the second phase. Software to implement collision avoidance and sharing of airspace is currently under development at DSTO (2007) and the University of California (Kim et al., 2003), both of which who have recent success in this area. The implementation of the third phase could be achieved using a relay on the throttle and parachute servo control line, which activates if a heartbeat signal is not received from the autopilot, indicating normal operation. Although a preliminary design was created in order to implement this third phase, it was decided that this would not be implemented in iSOAR in order to conserve on-board space, weight, and development time.

9.5.2 Operation Safety

The safety system implemented on the UAV consists of a number of in-flight failure patterns which activate in specific circumstances, coupled with a safety routine which effectively terminates the flight. The response to the individual failure patterns has been determined in order to maximise the possibility of rectifying the error, however it is noted that the requirements of the ARCAA Outback Challenge had to be taken into account and that in some cases the system would be changed for general civil aviation use. Also, two buttons have been programmed into the ground station GUI to enable a manual 'panic button'. One of these buttons labelled 'Terminate' immediately ends the flight using the sequence described in section 9.5.3, the other, labeled 'Home', causes the aircraft to return to its original launching location at an altitude of 1000 feet, and then ends the flight using the afore mentioned sequence.

The failure patterns are effectively software interrupts, which activate in specific cases which cause significant risk to arise. These error handlers cover seven areas:

- Loss of GPS signal
- Loss of motor power
- Low servo or autopilot battery voltage
- RC link failure
- Long range loss of communication
- Fatal error\(^1\)
- Control error\(^2\)

When an error handler is activated, a small section of custom code is run, which implements the sequences discussed in Section 9.5.4.

\(^1\)the term 'fatal error' covers a broad number of possible errors related to sensor failures that may occur.

\(^2\)the term 'control error' refers to an instance whereas the autopilot decides that the control loops are no longer maintaining any significant control over the airframe, and therefore can no longer effectively pilot the aircraft. For instance; if servo rods become disconnected from the control surfaces the autopilot will eventually sense this as a lack of response to a significant manoeuvre.
9.5.3 Flight Termination

In the event that an error cannot be rectified, the safety system will implement a procedure which will end the flight in what has been deemed to be the safest method possible given the circumstances. The word ‘terminate’ will furthermore be used in this document to represent the immediate cessation of the flight in the following manner:

- Motor switched off and all control surfaces set to neutral
- Elevators set to a 5% incline to induce upwards pitched attitude
- Parachute deployed

All three commands will be performed in quick succession, but it must be noted that the parachute takes around two seconds to set completely. The use of elevator incline is intended to both slow the plane down a small amount as well as ensure that the parachute does not foul when it is released. This is a beneficial manoeuvre; however it is not critical to the reliability of the system.

9.5.4 Implementation

The response to the error handlers has been heavily influenced by the ARCAA UAV Outback Challenge competition requirements, and hence the competition specified events have been separated in the section below. The competition specified occurrences requiring termination are as follows (ARCAA, 2007):

- If the UAV goes outside the mission boundary
- If the communications link is lost for more than five seconds
- If the UAV is deemed to be out of control by the judges
- If the on-board systems lock-up

Table 9.2 indicates the mode of termination for each specific occurrence, as well as any additional notes on termination implementation.
## 9.6 Concept of Operations

The control systems operational concept for the Outback Challenge competition (ARCAA, 2007) has been developed and is outlined in the subsections below. This concept is intended to be suitable for many civil applications, consisting simply of a launch and fly out to a search area, search, loitering and payload deployment around a target of interest, and then return to base. Further flight testing of the aircraft would determine the overall abilities and restrictions of the control and navigation system.

### Table 9.2: Safety System: Termination Modes

<table>
<thead>
<tr>
<th>Event</th>
<th>Manual / Automatic</th>
<th>Implementation</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mission Boundary</td>
<td>M</td>
<td>Operator clicks on the 'Terminate' button for immediate termination</td>
</tr>
<tr>
<td>Loss of Communication Link</td>
<td>A</td>
<td>In-Flight Failure Pattern Six: Loss of Communication Signal</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Count 2 seconds</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Immediately turn aircraft toward ground station in an attempt to regain link</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Count 8 seconds</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- If comm. signal has not been regained, terminate flight</td>
</tr>
<tr>
<td>UAV deemed out of control</td>
<td>M</td>
<td>Operator clicks on the 'Terminate' button for immediate termination</td>
</tr>
<tr>
<td>On-board loss of control</td>
<td>A</td>
<td>In-Flight Failure Pattern Zero: Control Failure</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Termination mode immediately activated</td>
</tr>
<tr>
<td></td>
<td></td>
<td>OR</td>
</tr>
<tr>
<td></td>
<td></td>
<td>In-Flight Failure Pattern One: Fatal Error</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Termination mode immediately activated</td>
</tr>
<tr>
<td>Judges Request</td>
<td>M</td>
<td>Operator clicks on the 'Terminate' button for immediate termination</td>
</tr>
<tr>
<td>Loss of Motor Power</td>
<td>A</td>
<td>In-Flight Failure Pattern Three: Loss of Motor Power</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Termination mode immediately activated</td>
</tr>
<tr>
<td></td>
<td></td>
<td>(note that throttle is still set to zero in case control is still active and</td>
</tr>
<tr>
<td></td>
<td></td>
<td>motor sensor is reporting inaccurately</td>
</tr>
<tr>
<td>Low Battery Voltage</td>
<td>A</td>
<td>In-Flight Failure Pattern Zero: Low Battery Voltage</td>
</tr>
<tr>
<td></td>
<td></td>
<td>- Termination mode immediately activated</td>
</tr>
</tbody>
</table>

Discarding the mission boundary activation, these safety routines are suitable for an civil application. However, in the case of communications link loss, it is important to note that the communications link to ground is only present in order to enable monitoring and additional features of the autopilot, and is not required for aircraft control. Thus in a generic application the two and eight second breaks would be extended to much higher values, to allow more time for problems to be rectified. If the mission area is restricted, calculations such as completed in Appendix I for the competition can be modified in order to determine for a worst case scenario what possible drift from original location the UAV would have, given set time periods.
Launch will be conducted using RC control will be used for the take off procedure, followed by a single localised circuit during which the pilot will obtain a reasonable level of altitude before switching to autopilot control.

Flight path control will be managed by the onboard autopilot which conducts all subsequent flight alterations. The initial flight leg, searching routine and final homeward leg is programmed before the flight, along with a number of failsafe routines. As the majority of the flight is beyond RC range, the most likely outcome in the result of a control failure of some kind is the activation of the parachute via the safety system discussed in Section 9.5.

Searching and deployment will be conducted in two phases. The first phase consists of manually sighting the target using the video stream provided on the laptop. Activation of a pre-programmed routine will allow the plane to loiter in the area of interest, passing directly over the point of activation in the middle of a figure of eight pattern. This allows continued monitoring of the site of interest, whilst the target is being confirmed.

Payload deployment will be conducted on the end of an extended, direct approach to the target mid-flight, after identification of the target is confirmed. This occurs whilst the autopilot is loitering and is the simple manipulation of waypoints via the on-screen display. Payload deployment can occur manually through the autopilot interface. An physical overlay is used on the imaging screen, which creates a targeting boundary; that is, a boundary for a specific speed and altitude within which deployment should occur. The operator watches this until the approach vector causes the target to appear within this targeting boundary which is the condition for payload deployment. Upon this condition, a selection of the payload deployment button on the autopilot interface causes the release of the payload.

'Head for home’ A pre-programmed retrieval routine is programmed into the autopilot. Upon activation of this routine the autopilot will direct the UAV to a chosen altitude and location at which RC control will be taken and the plane landed under manual control. If the re-establishment of an RC link is not possible, a separate routine is activated, which causes the UAV to a separate location and activates the parachute.

9.7 Summary

The control and imagery systems have been tested on the ground to verify the operation of each component followed by the integration into the airframe. A ground station has been detailed and implemented such that the UAV can be controlled manually or autonomously while in communication with the ground station as well as provide for manual viewing of an imagery stream. The three basic inner stability loops of the autopilot have been tuned to values appropriate for the iSOAR airframe, and straight, level, autonomous flight has been achieved. A complete...
safety system has been detailed and implemented into the control systems. Finally the concept of operations has been developed for entry into the ARCAA Outback Challenge.
Chapter 10

Component Testing

Component testing was conducted in order to verify design specifications and determine the performance of components. The components were tested individually to be consistent with the modular design of the UAV. This section will discuss the testing of the propulsion system, wings, the parachute, autopilot, camera and communications equipment.

10.1 Propulsion System Testing

To ensure the validity of the data used to select the motor, a small, static motor thrust test was performed. This provided a justification of the selection, by verifying the the power output of the motor with the published data from the manufacturer.

When using a propeller, the manufacturer claims the data in Table 10.1 is accurate (ModelMotors, 2006).

<table>
<thead>
<tr>
<th>$V_{motor}$ [V]</th>
<th>$I_{motor}$ [A]</th>
<th>$P_{shaft}$ [W]</th>
<th>$P_{battery}$ [W]</th>
<th>$\eta_{motor}$ [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>26.2</td>
<td>27.5</td>
<td>612</td>
<td>721</td>
<td>85</td>
</tr>
</tbody>
</table>

The above data was used to estimate the static thrust that the manufacturer would have obtained. This calculation was made using Equation 10.1 (McCormick, 1994).

$$T_{static} = \sqrt[3]{2\rho A_{disc}(\eta_{propeller} P_{shaft})^2} = \sqrt[3]{2\rho A_{disc} P_{thrust}^2}$$

(10.1)

Based on information from Roskam (2004b), it was assumed that propeller efficiency, at static conditions.

The purchased motor was fitted with a propeller of the same size as the one relevant to the manufacturer’s data, and operated under similar voltage and current conditions to the published data. The produced thrust was measured using a load cell, and the results of the test, as well as the calculated static thrust value from the manufacturer’s data, is presented in Table 10.2.
As this data shows, the difference in total efficiency, \( \eta_{total} = \frac{P_{\text{battery}} - P_{\text{thrust}}}{P_{\text{battery}}} \times 100\% \) between the two results is approximately 6.3%. This result suggested that the manufacturer’s data was trustworthy, and thus the motor would provide the desired performance.

<table>
<thead>
<tr>
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<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Test</td>
<td>26.2</td>
<td>28.4</td>
<td>742.8</td>
<td>0.31</td>
<td>32.7</td>
<td>331.7</td>
<td>552.8</td>
</tr>
<tr>
<td>Manufacturer</td>
<td>26.2</td>
<td>27.5</td>
<td>720.5</td>
<td>NA</td>
<td>35.0</td>
<td>367.2</td>
<td>612.0</td>
</tr>
</tbody>
</table>

In addition to the above motor test, a field test was conducted with the propulsion system to estimate the endurance of the system. With 50% of the battery supply on board the plane, the motor was run at static conditions, at 70% to 90% throttle settings. Operating under these conditions, the propulsion system ran for 40 minutes. This result suggested that the propulsion system would indeed be capable of providing enough energy to provide the vehicle with the specified endurance of 1 hour.

### 10.2 Wing Structural Test

A load test was undertaken in order to examine the structural integrity of the wing. The test was conducted using sand bags of known mass, a common method for carrying out a load test on aircraft. By dividing the load distribution plot, shown in Figure 7.8, into five segments, it was possible to determine the required weight for each section of the wing. Under the maximum loading configuration for the wings, that is at the maximum take-off weight (MTOW) with a load factor of 3.8, the wings were loaded with the necessary sandbags distributed across the wing. Shown in Figure 10.1, the load was distributed both span wise and chord wise, in order to reflect the aerodynamic loading on the wing. The deflection of the wing structure was then measured and compared with the calculated value. The deflection experienced was considerably less than the calculated value, this was expected as the calculated value is only based on the spars. However in the test the foam and skin also resisted some of the bending force.

![Figure 10.1: Wing Structural Test Setup](image)
10.3 Parachute System Testing

The parachute system designed and developed by the project group was tested to ensure the validity of the theoretical analysis and to ensure the reliability of deployment. Testing was conducted in two phases; parachute drag tests and deployment system tests. Parachute drag tests consisted of the

10.3.1 Parachute Drag Tests

Testing of the primary parachute was conducted in order to determine the steady state descent rate and predict the opening loads. This was conducted with an experimental set up on a utility vehicle with a large lever arm that was attached to a spring gauge. By driving at different velocities the drag coefficient could be estimated for. This developed a large range of experimental data that could be used for accurate estimate of the descent speed. Opening loading was generated through manual deployment of the parachute during forward travel. Active monitoring of the spring gauge was used to obtain the peak loading experienced which was recorded for later analysis. The results of these tests are presented in Figure 10.2.

![Parachute Drag Test Results](image)

(a) Drag coefficient during steady state testing

![Parachute Opening Load Tests Results](image)

(b) Opening load experimental results

Figure 10.2: Experimental Parachute Test Results
10.3 PARACHUTE SYSTEM TESTING

Static analysis

The results presented in Figure 10.2 (a) have been analysed with the application in mind. The estimate of a steady state drag coefficient of approximately 0.94 implies that the actual descent speed would be approximately 5.75 m/s, which is marginally increased from the predicted 5.5 m/s. This additional descent speed has been deemed feasible for this design.

\[ C_{D_{\text{experimental}}} = 0.94 \]

Opening load analysis

The analysis of the experimental opening load data, as presented in Figure 10.2 (b), has been analysed with the operational requirements of that parachute system in mind. The experimental line of best fit has been assumed to be a second order polynomial with an equation as stated in Equation 10.2 below.

\[
\frac{F_{\text{opening}}}{mg} = 0.0412v^2 - 0.0874v + 1.2202
\]  (10.2)

The experimental data predicts a far higher opening load than expected form the analytical results, as described in Appendix E. Reasons for this inconsistency could result from experimental set up where a small amount of slack allowed the test rig to gain some momentum during opening loading and the fact that the test rig was forced, by the inertia of the vehicle, during opening. During real deployment of the parachute system the deceleration of the UAV will reduce the opening loads significantly, as the parachute is not so strongly forced into opening conditions due to the coupled dynamics of the aircraft-parachute system. Despite this explanation it is not recommended to deploy the parachute at design cruise conditions until preliminary tests have been conducted at lower speeds, and confidence gained in the system and the associated loads during opening.

10.3.2 Deployment Mechanism Testing

Extensive testing was conducted to ensure desirable kinematics of the deployment mechanism thus reliable operation. A representative box with internal dimensions equivalent to that of the UAV was attached to the 3 metre arm above a vehicle. This box consisted of the parachute bay, actual deployment mechanism, and a representative tail plane. The tail of the aircraft was included in the test to ensure that the pilot or main chute did not become tangled or restricted during deployment. Initially the system was configured to have a hinged top hatch that was not jettisoned when the parachute was launched. Tests showed that this decreased the reliability of the system, hence the hinge system was removed and the hatch was jettisoned for each parachute launch. Once the hinges were removed the complete system was tested 5 times at velocities up to 55km/hr with a 100% success of deployment. A person standing separate to the vehicle activated each of these tests from the remote control transmitter.
Another characteristic of the parachute system is the opening time of the parachute, testing showed that from when the command is given to deploy the pilot chute it takes just over four seconds for the main parachute to set. In addition to the opening time for the parachute it is expected that there will be some time required for the UAV to enter steady state descent. Based on this opening time and the velocity of the UAV the parachute recovery system is not expected to be a viable recovery option for altitudes less than 200ft.

10.4 Autopilot Testing

A series of tests were completed on-ground to ensure that the autopilot was responding as expected, and that all on-board sensors were in working order. These included comparing expected accelerometer and gyroscope outputs to actual outputs for various angles, checking airspeed readings from the dynamic pressure sensor, and verifying correct GPS readings at several locations around Adelaide.

Of these tests, points of note were the sensitivity of the dynamic pressure sensor, which showed a high sensitivity to wind gusts. This highlights the need for careful choice of controlling parameters during days with gusty conditions to ensure that control surface outputs aren’t overly effected by wind gusts. Also of note was a jitter in the response of the GPS receiver, (Figure 10.3). This jitter was in the order of 2 metres at maximum, excepting outliers, and had a tendency to move within a CEP (circular error probable) of around 10m when left in the same position for a long period of time (>10min). This is within the manufacturers specifications of a horizontal error < 5m (50% of the time) and < 8m (90% of the time) (Trimble, 2007). Although the jitter occurred much quicker than expected, significant effects were only observed while the receiver was stationary, and so no modifications to the autopilot were considered. These effects can be seen in Figure 10.3, during which the aircraft was within 2 meters of position 1 (top right) for 27 minutes, followed by a movement of approximately 30m over 6 seconds, and followed by remaining at position 2 for around 10 seconds. Note that the division of 0.0001 in longitude is equal to around 9.2m, and 0.0001 in latitude equal to around 11.1m, assuming the earth is a perfect sphere with a radius of 6378km (Michels, 1997).

Figure 10.3: Plot of GPS Data over a Period of 30 Minutes
10.5 Camera Calibration

Calibration of the camera was carried out by adjusting the modes mentioned in Section 5.5. The modes were adjusted by using 8 dip switch controls as seen in Figure 10.4.

![Camera Circuit](image)

Figure 10.4: Camera Circuit

The camera’s flickerless mode was tested and the results are shown in Figure 10.5. The flickerless mode was turned off and the image produced had continuous flickers and a darker image was also attained. The image obtained was blur and consisted of changing pattern of TV lines. The flickerless mode was turned on and a clear image with no flickers and no changing pattern of TV lines was observed. Thus, the flickerless mode turned on was a desirable factor.

![Flickerless Mode Adjustment](image)

Figure 10.5: Flickerless Mode Adjustment

The backlight compensation mode was also tested, as shown in Figure 10.6. The backlight compensation was turned off and an image with bright light was obtained due to the glare from the sunlight. The image produced was not desirable to view any object due to the amount of high brightness present in the image. The backlight compensation was then turned on and the high brightness was reduced and a desirable image to view the object was attained. Thus, the backlight compensation mode was turned on in order to view the object and eliminate glare due to sunlight.
Further testing of the camera was conducted to determine an appropriate shutter speed. The shutter speed was set to various levels in order to determine the desirable shutter speed. The levels set for the shutter speed were 1/60, 1/100, 1/250, 1/500, 1/1000, 1/2000, 1/4000 and 1/10000 secs. A 1/125 setting meant that the shutter curtain could open and close within one hundred and twenty five of a second. The lower shutter speed allowed immense amount of light to enter the camera and was exposed on to the film causing a blur in the image and low quality when exposed to the sunlight. A high shutter speed of 1/10000 was selected in order to allow less amount of light to enter the camera when exposed to bright sunlight and the result can be seen in Figure 10.7.

In addition to these factors, the camera was calibrated with the following configuration. The auto gain control was set to 12dB and the gain limiter was set to 26dB. The auto white balance was turned on in order to regulate the amount of white light entering the camera and the auto white settings was set to 'outdoor' from the other options available. This helped provide a clear image by eliminating the glare during sunlight and improving the resolution of the camera.

10.5.1 Pre-Flight Testing

Pre-flight testing was undertaken in order to determine the quality of the camera and the range of the video downlink. The camera was calibrated to its best quality initially without the utilization
of wireless transmission it was then connected to the transmitter that was directly connected to
the laptop using the Belkin USB DVD creator. The camera was tested indoors and outdoors
however the indoor testing did not account for the image obtained during sunlight. The camera
was recalibrated during outdoor testing by adjusting the modes present on the camera eliminated
glare from sunlight. A clear image was obtained during the testing process as seen in Figure
10.8.

![Calibrated Image](image.png)

Figure 10.8: Calibrated Image

10.6 Communication Range Testing

10.6.1 Autopilot RF Modem

Ground testing for the range of the autopilot modem was conducted. The signal strength was
observed at different distances, using the LED display on the RF modem. An indicative signal
strength was recorded as a value between 0 and 3, where 3 is a strong signal and 0 is no signal.
The results of this test are graphed in Figure 10.9, and it can be seen that a maximum range of 6
km was achieved. The modems were elevated to a height of 2 metres in an attempt to maintain
line of sight. Despite this elevation trees and variations in terrain deteriorated the signal. It is
believed that a greater distance will be achieved when one modem is mounted on the UAV in
the air.
10.6.2 RC Controller

A test was conducted to determine the range capabilities of the RC controller. The test was conducted in an open field in an attempt to maintain line of sight. Two methods of testing were employed to fully test the range of the RC equipment purchased. The first method was used to verify the manufacturer’s standards that a range of 75 feet could be achieved with the antenna retracted. This setup will suppress the output from the transmitter, reducing its range. The second test was performed to determine the functional range of the system with the antenna extended.

The procedure of the two tests was almost identical. The RC receiver was placed on the ground with antenna horizontal along the ground. A servo was connected to the receiver and controlled by the transmitter, held at arms length. The distance between transmitter and receiver was increased until the transmission was unsuccessful.

Test Results

The range achieved with the antenna down was approximately 150 metres. This surpasses the required distance, verifying the manufacturers specifications.

The range achieved with the antenna up was 800 metres. The image below (Figure 10.10) shows the location of the receiver (bottom right) and the transmitter at the final position. This range is sufficient for RC operation of the UAV. The signal passed through some urban obstructions and a number of trees. Therefore, the actual maximum range of the system may be greater than this distance. The transmitter was also on a slope, slightly elevated when compared to the receiver, which also may have affected the signal.
10.6. COMMUNICATION RANGE TESTING

10.6.3 Video Downlink

A test was conducted in order to determine the range of the video downlink. The antenna was mounted on a 4 meter stand and directed towards the transmitter, as shown in Figure 10.11. A clear image with no breakouts or noise disruption was observed up to a range of 6km. The location of this test is shown in Figure 10.12. Line of sight could not be maintained beyond this distance, so the transmission was lost. However, it is believed that greater distances can be achieved if line of sight is maintained.
Figure 10.12: Image of Test Location (Google, 2007)
Chapter 11

Flight Testing

Testing of the complete UAV system was broken into phases. Each flight aimed at testing the operation of different systems. Components were added gradually over several flights in order to build up to a complete system. In total six flight test days were carried out which included a total of eight flights. The order in which systems were tested was: Airframe / RC control system, imaging systems, payload deployment system and Autopilot system. The final flight took place at Kingaroy at the ARCAA UAV Outback challenge to demonstrate the capabilities of the aircraft, this is discussed in Chapter 12. The following section details the aims of each flight test day, the outcomes and modifications that were made. This chapter also includes an evaluation of the different aspects of the flight tests.

11.1 Flight Test Procedures

The development of detailed flight test procedures was initialised well before flight testing phase was undertaken. The flight test procedures can be found in Appendix J. The purpose of these documents was to:

- Develop of plan for the flight tests
- Ensure that all required components were taken to each flight test
- Detail exactly what the objectives were
- Develop methods and practices that was both safe and efficient
- Develop the required documentation for the recording of data in the field

From the development of these documents it was identified that each flight test consisted of three phases; pre-flight, flight test and post flight. The pre-flight phase consisted of the charging of all batteries, fitting of batteries, ensuring the full functionality of the RC equipment and control surfaces and ensuring the aircraft, and all required equipment, was packed for transportation. Flight test phase consisted of the unpacking and set up of the aircraft, ensuring the operation
of all equipment and servos, an RC range test and final safety checks. The post test phase consisted of the disassembly of the aircraft for transportation, the removal of all batteries ready for charging, and the checking of the aircraft structure and control surfaces for damage.

11.2 Proof of Aerodynamic and Mechanical Design

Objectives

The purpose of the initial flight tests was to provide information about the aerodynamics of the design in ground roll, take off, climb, flight and landing. The controllability and the reliability of the RC system were also investigated. Additionally the test aimed to confirm the expected endurance of the motor batteries. The test also allowed the pilot to become familiar with the aircraft for all manoeuvres including take off, climb, banking, descent and landing.

Aircraft Configuration

Due to the inherent risks of initial flight tests the aircraft was configured to contain only essential items for flight. This configuration includes four of a possible eight batteries and RC system but did not include imaging equipment or the autopilot. However the batteries for the autopilot and imaging equipment were onboard to improve the aircraft’s longitudinal static stability. The configuration of the aircraft for this flight is summarised in Table 11.1.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Status</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of power plant batteries</td>
<td>4</td>
</tr>
<tr>
<td>Autopilot installed</td>
<td>no</td>
</tr>
<tr>
<td>Autopilot battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Imaging system installed</td>
<td>no</td>
</tr>
<tr>
<td>Camera battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Deployable payload</td>
<td>no</td>
</tr>
<tr>
<td>Expected run time</td>
<td>30mins</td>
</tr>
<tr>
<td>Static margin %MAC</td>
<td>9.5</td>
</tr>
<tr>
<td>Take off weight (kg)</td>
<td>8.3</td>
</tr>
</tbody>
</table>

Method of Test

The test incorporates various stages building up to a full circuit around the airfield. These steps are detailed below

1. Perform a range test to ensure the RC equipment will be able to communicate with the aircraft in the flight area.

2. Increase throttle for initial ground roll and observe the aircraft’s dynamics while accelerating.
3. Once the pilot is confident with the ground roll dynamics, take off and immediately land several times until the pilot is confident in the aircraft’s controllability, stability and aerodynamics.

4. Take off, conservatively bank right and complete one circuit followed by a landing. Do not land aircraft if the pilot is not confident in the approach.

5. Once the aircraft is on the ground run the motor at a throttle position corresponding to cruise, indicated by the pilot, until the batteries are flat. This will be used to estimate the aircraft’s endurance.

Results

- RC Range was as expected
- Initial ground roll tended to turn to the left
- Rapid climb on take off for initial flight
- Forward roll over on landing for initial flight
- Controllable second take off
- Aircraft stable in flight for all manoeuvres
- Baulked approach
- Two circuits achieved
- Roll over on landing for second flight
- Endurance of 45 minutes achieved for 4 batteries

Discussion of Flight Results

The range test was carried out with the aerial on the transmitter in the retracted position, this is to suppress the output of the transmitter as discussed in Section 10.6.2, the range achieved was as expected. Initial ground roll tests indicated that the propeller wash over the aircraft was causing it to turn left during early acceleration. This was corrected by adjusting the direction of the nose gear and increasing the control authority on the rudder, allowing for greater rudder deflection. The first take off resulted in a rapid climb and the altitude reached was greater than anticipated. The aircraft was then landed at a range of approximately 100m from the pilot. This resulted in the aircraft rolling out in rough ground and a forward nose over. No damage was incurred to the airframe or components. The second flight was well controlled during take off and climb. The aircraft appeared to be inherently stable during all manoeuvres and feedback from the pilot indicated that it was inherently stable in all aspects. A full circuit was achieved however the pilot was not confident with the landing approach. Therefore the landing was aborted and another successful circuit was completed. The aircraft was landed in the next approach which
resulted in another nose roll over due to pilot technique and soft sand. The roll over damaged the nose gear. The motor was then run in static configuration at approximately 50% throttle for 33 minutes and 75% throttle for 5 minutes. Prior to this test the motor had run for 8 minutes during the flight. This shows that the aircraft will surpass the endurance requirement of 1 hour when all 8 motor batteries are installed.

Modification for the next Test Flight

The first flight tests were a success despite the fast climb in the initial test and the nose overs during landing. In order to correct these issues several changes were made. The nose gear was modified to use a larger nose wheel. The nose overs during landing were a result of pilot technique and rough ground. A review of the video footage was carried out with the pilot and different flight techniques were discussed. In addition a better understanding of the landing areas was developed for future flights.

11.3 Testing of Imaging System

Objectives

The primary objective of the second day of flight testing was to receive and record live footage from the UAV at a ground station. Footage was transmitted at different altitudes up to a maximum of 400ft. As a secondary objective the aerodynamics of the aircraft were to be closely monitored, the take off velocity and cruise velocity of the aircraft were to be obtained and the effects of landing gear modification would be observed.

Aircraft Configuration

The configuration of the aircraft was the same as in the initial flight tests, however the camera and image transmission equipment were installed on the aircraft. The aircraft’s configuration is summarised in Table 11.2.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Status</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of power plant batteries</td>
<td>4</td>
</tr>
<tr>
<td>Autopilot installed</td>
<td>no</td>
</tr>
<tr>
<td>Autopilot battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Imaging system installed</td>
<td>yes</td>
</tr>
<tr>
<td>Camera battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Deployable payload</td>
<td>no</td>
</tr>
<tr>
<td>Expected run time</td>
<td>20mins</td>
</tr>
<tr>
<td>Static margin %MAC</td>
<td>8.0</td>
</tr>
<tr>
<td>Take off weight (kg)</td>
<td>8.46</td>
</tr>
</tbody>
</table>
Method of Test

The test incorporates a single flight with a variety of stages.

1. Perform an RC range test with aerial retracted
2. Ensure image is received at ground station prior to take off.
3. Take off aircraft between 2 poles 10m apart and climb to 100ft
4. Perform circuit at 100ft and climb to 200ft
5. Perform circuit at 200ft and climb to 300ft
6. Perform circuit at 300ft and climb to 400ft
7. Perform circuit at 400ft and descend to 30ft
8. Fly past two poles placed 10m apart
9. Perform circuit at 50ft and repeat 8
10. Land aircraft on a suitable surface

Results

• Successful take off and climb achieved.
• Take off velocity able to be determined (37km/h ± 3km/h)
• Footage continuously transmitted and received from aircraft
• Some breakouts in footage
• Aircraft controllable at 400ft altitude
• Two passes between poles for cruise velocity (50km/h ± 5km/h)
• Successful landing without rollover

Discussion of Flight Results

The RC range test and ground station check prior to flight returned the expected results. The take off and climb to 100ft was successful with no instabilities observed. The take off occurred between two poles ten meters apart and within the view window of a video camera. As a result a prediction of the take of velocity could be made. A laser gun was also used to determine this velocity, however due to the size of the aircraft and the lack of a perpendicular surface to reflect off a reading was not obtained. During each circuit at 100, 200, 300 and 400 feet in altitude a image was received and recorded at the ground station. The resolution of this image was sufficient to observe a person at an altitude of up to 250ft and cars from 400ft. Some breakouts
occurred in this image, especially when the angle between the aircraft and antenna direction was large. The aircraft was controllable at the maximum altitude of 400ft and in the descent back to 50ft. Two passes between the poles placed 10 metres apart were achieved, these were filmed and a review of the footage enabled a prediction of the aircraft’s velocity. The aircraft was then landed successfully with no nose over. The laser gun was used again in an attempt to determine a landing velocity, however this was not achieved.

Modification for the next Test Flight

The aircraft performed as expected in all regimes. The only modification proposed was the location of the ground station. It is suspected that the breakouts in the images from the aircraft were due to the limited sweep of the uni-directional antenna. In subsequent camera tests the ground station was moved further from the aircraft flight area to ensure the UAV remains within the antenna’s operational angle.

11.4 Testing of Payload Deployment Mechanism

Objectives

The primary objective for this test flight was to test the payload deployment system and observe the dynamics of the aircraft after the release of the payload. Aircraft performance with a greater take off weight will also be observed. As with all test flights the aerodynamics of the plane were also closely monitored by observers and post flight in video footage. Competition requirements dictate that footage of a left-hand and right-hand turn must be demonstrated, capturing this on film was also an objective of this flight.

Aircraft Configuration

The configuration for this flight was heavier than previous flights. The deployable payload and its mechanism were installed, while the imaging system was removed from the aircraft. This involved the addition of two servos, a 500ml bottle of water and the attachment mechanism. This configuration is summarised in the Table 11.3.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Status</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of power plant batteries</td>
<td>4</td>
</tr>
<tr>
<td>Autopilot installed</td>
<td>no</td>
</tr>
<tr>
<td>Autopilot battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Imaging system installed</td>
<td>no</td>
</tr>
<tr>
<td>Camera battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Deployable payload</td>
<td>500ml water bottle</td>
</tr>
<tr>
<td>Expected run time</td>
<td>20mins</td>
</tr>
<tr>
<td>Static margin %MAC</td>
<td>9.5</td>
</tr>
<tr>
<td>Take off weight (kg)</td>
<td>8.9</td>
</tr>
</tbody>
</table>
Method of Test

This test was relatively simple and only involved one take off and sufficient circuits to ensure the payload drop could be executed safely and filmed to fulfill requirements of the ARCAA UAV outback challenge.

1. Perform standard RC range test
2. Fly two left-hand circuits
3. Fly two right-hand circuits
4. Fly a downwind practice run for payload drop
5. On confirmation of camera operators repeat 4 and deploy payload
6. Land aircraft

Results

- Footage was captured for left-hand and right-hand turns
- Payload was successfully deployed
- Payload deployment was captured on movie and still cameras
- Successful landing with no rollover

Discussion of Flight Results

The flight was exactly as planned with successful left-hand and right-hand circuits executed. The aircraft had no observable signs of instabilities due to the payload. The take off distance was increased as a result of the higher aircraft weight. Longer grass on the runway may also have contributed to the increase in take off distance. Once the payload was deployed the aircraft did not pitch up or down due to a change in the centre of gravity of the aircraft. Landing was executed on the first approach with no nose over. The payload (water bottle) split on impact and no water was retained in the container.

Modification for the next Test Flight

Minimal modifications were required after this flight. A tougher water container for the deployable payload was purchased for subsequent payload deployments.
11.5 Autopilot Integration

Objectives

This test was more complex than previous tests. The primary objective of the flight was to operate the aircraft under autopilot control and achieve straight and level autonomous flight. The test aimed to tune the first three control loops of the autopilot using observation of the oscillations experienced in the pitch, roll and yaw directions while under autopilot control.

Aircraft Configuration

The aircraft contained the autopilot and associated communication equipment. Additional power plant batteries were installed, though not connected, to mimic the dynamics of the full flight configuration. The configuration of the aircraft for this flight test is summarised in Table 11.4.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Status</th>
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<tbody>
<tr>
<td>Number of power plant batteries</td>
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<tr>
<td>Autopilot installed</td>
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</tr>
<tr>
<td>Autopilot battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Imaging system installed</td>
<td>no</td>
</tr>
<tr>
<td>Camera battery onboard</td>
<td>yes</td>
</tr>
<tr>
<td>Deployable payload</td>
<td>no</td>
</tr>
<tr>
<td>Expected run time</td>
<td>30mins</td>
</tr>
<tr>
<td>Static margin %MAC</td>
<td>9.5</td>
</tr>
<tr>
<td>Take off weight (kg)</td>
<td>8.6</td>
</tr>
</tbody>
</table>

Method of Test

This test involves a combination of RC flight and autonomous flight.

1. Confirm communication link between autopilot and ground station

2. Test the aircraft’s response to changes in attitude with the autopilot in control

3. Pilot conduct an RC range test

4. Take off and fly circuit under RC

5. Activate autopilot during upwind leg of circuit

6. Regain RC and repeat 5

7. Land aircraft
11.6. AUTOPILOT INTEGRATION II

Results

- RC Range achieved was less than previous flights
- Aircraft was initially unable to take off in long grass
- Aircraft was stable under remote control
- Interference was experienced resulting in loss of RC communication
- Aircraft crashed

Discussion of Flight Results

Initial range tests indicated the range of the RC transmitter was reduced but was still sufficient to operate the aircraft. Initial tests were aborted at take off due to insufficient acceleration in the long grass. Once the grass was cut a successful take off was performed. While turning onto the downwind leg of the initial circuit an interference problem was encountered. Once this problem was known the pilot attempted to land the aircraft and abort the flight test. However, while preparing for approach RC communication was lost resulting in a crash. The wings, nose cone, empennage, landing gear and top hatch separated from the fuselage on impact however the fuselage remained largely intact with no damage to the control systems.

Modification for the next Test Flight

The interference problems that caused the crash of the aircraft were to be solved prior to the next flight. This involved a crash investigation and tests to find the source of the interference, discussed in Section 11.10. The structural damage to the airframe was rectified and damaged equipment was replaced, including the motor and power plant batteries.

11.6 Autopilot Integration II

Objectives

The objectives of this flight were to confirm on-ground tests relating to a new, 2.4GHz Spectrum, frequency hopping transmitter and operate the aircraft under autopilot control, achieving straight and level autonomous flight. The first three control loops were to be tuned by observation of the oscillations experience in the pitch, roll and yaw directions while under autopilot control.

Aircraft Configuration

The configuration of the aircraft, as it was flown in this test is detailed in Table 11.5.
Table 11.5: UAV configuration, Autopilot Integration Test Two

<table>
<thead>
<tr>
<th>Parameter</th>
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<td>Number of batteries</td>
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<tr>
<td>Autopilot</td>
<td>yes</td>
</tr>
<tr>
<td>Imaging system</td>
<td>no</td>
</tr>
<tr>
<td>Camera battery</td>
<td>no</td>
</tr>
<tr>
<td>Deployable payload</td>
<td>no</td>
</tr>
<tr>
<td>Expected run time</td>
<td>30mins</td>
</tr>
<tr>
<td>Static margin %MAC</td>
<td>9.5</td>
</tr>
<tr>
<td>Take off weight (kg)</td>
<td>8.6</td>
</tr>
</tbody>
</table>

**Method of Test**

The test incorporates the following steps:

1. Perform full, aerial extended, range test
2. Test aircraft response to changes in attitude
3. Take off and fly one circuit
4. Activate autopilot control on subsequent circuits
5. Monitor aircraft behaviour and tune gains in AP system
6. Land after 3 autopilot activations
7. Repeat 3-6

**Results**

- Range test returned positive results with the expected range achieved
- Aircraft had short take off distance due to head wind
- RC circuit performed successfully
- AP activated on 3 upwind legs with small, stable, rapid oscillations observed
- Successful landing
- 3 upwind under AP control achieved again
- Similar observations were made
- Interference experienced for short period of time
- Emergency landing performed
11.7. EVALUATION OF AIRFRAME PERFORMANCE

Discussion of Flight Results

Initial range tests with new transmitter were very encouraging. The range achieved in a full range test was as expected. Take off and initial aircraft performance was very similar to previous flights. AP system was successfully activated and was in control of the aircraft for up to 4 seconds in straight and level flight. Some oscillations were observed. During the second flight, while banking towards the pilot at approximately 1/3 of the maximum range achieved in flight interference was encountered and the aircraft slipped sideways. The pilot was able to regain control and perform an emergency landing. The propeller was damaged on landing.

Modification for the next Test Flight

Interference is still a problem between the Autopilot and RC communication systems. No more flights can occur in this configuration before the problem is resolved. A new propeller was required prior to future motor operation.

11.7 Evaluation of Airframe Performance

Flight performance and structural performance results were investigated during all flight tests. The evaluations of these results were difficult to quantify, and relied heavily on observations made by the pilot and other observers. Where possible, quantified measurements were made, and compared with the expected values.

11.7.1 Flight Performance

The primary goals of the flight performance tests were to:

- Determine the stability and controllability of the aircraft - The aircraft was designed to be highly stable and controllable; the tests were to verify this
- Compare the actual takeoff speed with the estimated value from the conceptual design, of 78km/hr
- Compare the actual takeoff distance with the predicted value of 30m
- Determine the ground roll performance of the UAV - The tests were to verify the effectiveness of the landing gear design in providing stable roll out and touch down / roll in
- Determine the maximum cruise speed of the vehicle, and compare this to the design value of cruise speed designed to be 90km/hr.

The flight tests provided qualitative and quantitative results for all of the goals except cruise speed, which could not be satisfactorily measured without the pressure sensing capabilities of the autopilot system, being operational.
Vehicle Stability and Controllability  During all flight tests, the aircraft exhibited highly stable and controllable flight. Evidence of this was found in the video footage received from the platform, as well as the comments of the pilot, who claimed, “...the aircraft is very stable, and flies like a trainer.” From these observations, it was concluded that the original stability and controllability characteristics specified from the conceptual design phase were attained.

Takeoff Speed  Flight tests showed a takeoff speed of approximately 50km/hr (true airspeed). This measurement was significantly lower than the predicted speed, and the discrepancy is attributed to two things:

- Approximations in the measurement of the speed discrepancies between the wings lift coefficient and planform area values used in the design calculations and those attained by the manufactured vehicle.
- The manufacture of the wing, which was believed to have increased the actual lift coefficient and planform area through increases in skin thickness, as well as profile augmentation.

The discrepancies in these results, although acceptable for the purposes of this project, indicate scope for more accurate wing lift coefficient prediction, and tighter tolerances in the manufacturing process.

Takeoff Distance  Takeoff distance, for the MTOW configuration, was found to be approximately 55m. This value is significantly larger than the designed distance and is attribute to two things:

- The use of a grass runway as opposed to a concrete surface as used in the calculations
- The use of pilot preferred power settings during run up, which were less than full power, as used in the design calculations

These reasons for the discrepancies suggested that the takeoff distance performance of the aircraft was satisfactory, and within the predictions of the design.

Ground Roll Performance  Roll out and roll in performance of the airframe was found to be adequate, with no tendencies to nose over or ground loop under normal landing conditions. It should noted that in rough terrain, the vehicle did nose over on landing, however, no damage to the vehicle was sustained, and the undercarriage was deemed adequate for normal, chartered operation. This problem was further alleviated through a change in landing approach technique, more suited to the tricycle undercarriage


11.7.2 Structural Test

The integrity of the airframe was checked by inspection after each flight to monitor the robustness of the aircraft, and hence its design success. In particular, cracking and delamination of the airframe were searched for in areas of high stress such as wing boxes to verify the structural design, as well as in join regions to verify the quality and suitability of the manufacturing methods used. No signs of cracking or delamination were found in these areas, justifying the quality of the airframe’s structural design and manufacture.

In addition to predefined testing of the structure, the crash of the airframe, and subsequent investigation, provided further, detailed results of the success of the airframe’s design, and its failure mechanisms. The empennage separated from the fuselage at a carbon ring frame as expected. Similarly the nosecone separated from the fuselage at the front ring frame. The energy absorbed in these failures, significantly reduced the damage sustained by the fuselage, resulting the autopilot system remaining fully functional. As mentioned, the nosecone was reinforced with a foam honeycomb to withstand damage during adverse landings. These results verified this idea and justified its inclusion.

11.8 Control Systems Evaluation

The two flight tests allow for some evaluation of the performance of the control systems, although the communications issues and corresponding postponement of the tuning process experienced make it hard to provide a complete evaluation. The ability of the autopilot to channel RC signals effectively was proved, along with the safe and reliable switching from autopilot to RC upon RC demand. No flutter or instability was observed during the change from RC to autopilot control, and the tuning of the first three PID loops verified the use of manual tuning as a valid approach.

However, the flight tests also highlighted the increased risk involved with the greater number of electronic components in the confined space of the fuselage. The need for a dependable RC range during tuning was also highlighted, as the ability to switch back to RC control is imperative to ensure that the airframe is not put at risk. This is especially important considering the observation that the control system has the potential to cause a dynamic instability in the system if the gains are set too high initially. Also noted is the need for a wider range of operations in later tuning procedures, which may have proved unattainable with the current communications limitations.

The ground station proved to be a suitable, portable system with minimal set-up time. No problems occurred with the use of the Serial to USB converter connection between the radio modem and laptop, which are notoriously unreliable. The use of the inverter was sufficient to power the ground station with a minimum of inconvenience, and it is noted that the use of a cheaper modified sine wave inverter rather than a pure sine wave inverter caused no difficulty in terms of compatibility with any of the ground station components. However, the laptop screen did prove difficult to read effectively in the bright outdoor conditions, and the use of a generator instead of an inverter could be considered to increase the portability of the ground station. It is also noted that the tests conducted did not allow for a true measurement of the control systems
communications equipment. If this proved insufficient in a full range test, mounting the modem on the imaging antenna stand would increase the capable range of the system.

Overall, the test results highlighted that the control systems have the potential to provide for a UAV with the capability of both manual and autonomous flight, but that the tuning and integration process are high risk procedures. It should also be noted that re-tuning of the autopilot would need to occur for each significant change in payload or on-board configuration. However, these re-tuning would be a less arduous procedure as the initial gains would be much closer to the tuned gains.

11.9 Camera Evaluation

While the initial altitude of the aircraft was predetermined as 120m, it was essential to test the camera at various altitudes to determine the camera’s effectiveness. A flight test day was set aside in order to record airborne footage from the aircraft, leading to an analysis of the image from the camera with an increasing altitude. Figures 11.1, 11.2 & 11.3 show still images captured from the video, at approximate altitudes of 100 ft, 200 ft and 400 ft respectively.

Figure 11.1: Photo from 100ft
11.9. CAMERA EVALUATION

Since the minimum altitude of the aircraft is limited to 200ft, Figure 11.1 at 100ft was employed as a datum for the four heights, as it clearly shows a human on the ground. In order to compare the most crucial figures at heights of 200ft, 300ft and 400ft, the images were captured with a common object, for instance the two cars seen in the bottom left hand corner of each image.

From evaluation of Figure 11.3, it can be noted that the two vehicles become difficult to distinguish from this height. At the minimum height of 200ft, it is able to discriminate between the two cars, as there are more physical features visible to the viewer. Since Figure 11.1 shows a crisp image of a human on the ground, and Figure 11.2 shows the most distinguishable image captured.
of the three relevant altitudes, it is assumed that a human would be relatively easy to determine from a height of 200ft. Thus, it is recommended that the flight altitude be adjusted from 400ft, equivalent to 120 metres, to 200ft, equivalent to 60 metres. However, should this readjustment of altitude be difficult to obtain due to aircraft speed and area coverage, it is further recommended that the maximum height of the UAV be limited to 300 feet, equivalent to 90 metres. Hence, the given range for the aircraft altitude, in order to maximize the efficiency of the imaging system, should be limited to between 60 metres and 90 metres.

The footage received from the camera showed no signs of degradation due to external vibrations. Thus, the mounting of the camera was deemed appropriate, and suitable.

11.10 Communications Evaluation

The performance of the three sets of communications equipment was evaluated during the flight tests. The two long range modems both provided acceptable communication throughout the testing, however RC communication was degraded below an acceptable level during the autopilot tests, resulting in a crash on one occasion.

The imaging communications equipment displayed proof of operation during the camera test day, with the only detrimental observation the breaks in camera footage that were observed intermittently. This has been identified as due to the static camera stand which was set-up too close to the field of operation. It is believed that with this antenna situated a significant distance away from the field of operation, more closely simulating the flight path profile, these breaks would not occur.

The long range autopilot communications equipment performed satisfactorily during both flight tests. No drop-outs were observed from the ground station and communication was initiated very quickly upon modem resets. This result is very positive, however the important parameter of the autopilot communications is the full range that can be achieved. A test of this would be required before the system could be declared fully operational, but was not considered a higher priority than the tuning of control loops, which required the ground station to be in the immediate vicinity of the UAV. Thus it remains to be seen whether the airborne increase expected from the ground range test will be sufficient to produce the required range of the autopilot. Otherwise the long range autopilot communications equipment has proved satisfactorily in all regards.

The RC transmitter performed satisfactorily on all of the first three test days, with no signs of drop-out or lack of control. However, during the autopilot integration test the RC range was significantly reduced. In this test the flight plan involved right hand circuits as shown in Figure 11.4. The RC connection was lost with the autopilot soon after take off; however the connection was restored and the pilot was able to regain control of the aircraft. The pilot then attempted to land the UAV, however the RC connection was lost again which caused a loss of control resulting in a crash. The maximum distance between the UAV and the pilot was 250m, well within the expected range of the RC equipment.
There were three key differences between this flight and previous flights which could have caused interference in the RC equipment. This was the first flight with the Autopilot, modem and associated additional electronics installed in the airframe. This was the first flight where a carbon fiber hatch was used. Furthermore, this flight was conducted at a new location, the Roseworthy campus, due to insurance issues at the constellation model flight club. This change may have caused the interference due to communication equipment in the area or from the trees surrounding the oval. Testing was conducted to determine which of these changes contributed to the reduction of RC range. Testing also attempted to locate other sources of interference which may have contributed to the problem, even though they were present in previous flights, such as EMI from the motor.
CHAPTER 11. FLIGHT TESTING

Further Testing

Testing was conducted in order to identify potential sources of interference. The range of RC transmission was measured in the presence of various operating components. Due to the inherent sensitivity of the system it was hard to quantify these results with a high precision. However, the tests identified sources of interference and could be used to approximate the range reduction caused by each interference source. The identified sources of interference are listed below in order of their significance.

1. Interference from modem transmission
2. EMI from modem
3. EMI from Autopilot
4. Carbon fiber hatch

Testing the interference from modem transmission showed that the RF interference reduced the RC range to around one fifth of its original distance. Antenna positioning was also determined to be significant, thus the RC antenna should be separated from other antennas to the extent permissible by the available space.

Testing was conducted to determine the effect of EMI from the Autopilot and the Modem. Both components caused a reduction in range, however the reduction due to the operation of the modem was larger than compared to the autopilot. One possible reason for this is the greater power consumption of the RF modem, EMI is produced by the flow of current to the units, and through the components.

The presence of the carbon fibre hatch could have contributed to the RC range reduction in two ways. It is possible that the radiation produced from surrounding components could have been magnified by the presence of carbon fibre both above (hatch) and below (landing gear), causing the radiation to effectively 'bounce' back and forth. Also, the carbon fiber hatch could be acting as a faraday cage retarding the transmission of data (White & Donald, 1973). It is noted that both occasions when RC reception was lost the planes was rolling right and heading toward the pilot. This orientation will cause the hatch to be between the transmitter and the pilot, as the RC antenna is located on the left side of the plane. While testing showed that other sources of interference were more significant than the hatch, it is believed that the hatch contributed to the interference issue.

The shielding effect of the hatch can be calculated using the following formulae.

$$ S_{dB} = A_{dB} + R_{dB} + B_{dB} $$

(Hickman, 1997)

where the absorption loss, $$ A_{dB} = 0.1315 \times t_{mm} \sqrt{fG\mu} $$

The following values were used for this equation

$$ t = 1.52mm $$
$$ f = 36MHz $$
$$ G = 0.00158 $$
11.10. COMMUNICATIONS EVALUATION

\[ \mu \approx 1 \]

The conductivity relative to copper, \( G \), is determined using Figure 11.5. This is evaluated assuming a temperature of 300K. Data for E35 carbon fiber is used as this is the closest representation of the composite used on the UAV. The magnetic permeability can be considered to be 1, as discussed in Chang & Lee (2002).

Figure 11.5: Variation of Electrical Resistivity with Temperature (Chung & Deborah, 1994)

This resulted in

\[ A_{dB} = 1.507 dB \]

This value of \( A_{dB} \) corresponds to an internal reflection loss, \( B_{dB} \approx -8.46 dB \) (White & Donald, 1973)

Also the reflection loss, \( R_{dB} = 108 + 10 \times \log \left( \frac{G}{\mu f_{MHz}} \right) = 64.4 dB \) (White & Donald, 1973)

Therefore, \( S_{dB} = 1.507 - 8.46 + 64.4 = 57.4 dB \)

This is assuming a complete obstruction of the path by the carbon fibre, which does not represent the configuration aboard iSOAR. However it does show that the contribution of path loss due to a carbon fibre obstruction could be considerable. Testing a partial obstruction of path showed a path loss of lower magnitude, but varying results did not allow for meaningful quantification at this stage. These results indicates that the on-board carbon fiber contributes to the interference issue, although testing showed this to be the least significant issue.

**Actions Taken**

From this analysis several actions were implemented in order to reduce interference and increase RC range. Resolution of the issue was first attempted through the use of copper shielding, ferrite beads and relocation of components with the continued use of the 36Mhz transmitter. These
measures did not prove sufficient, and so the purchase of a new RC system operating in a higher frequency range occurred.

Grounded copper shielding was constructed around the modem. This shielding was surrounded on both sides by electrostatic bags in order to prevent short circuits. Similar shielding was considered for the autopilot, however this would prevent access to the autopilot which is required for connecting the pitot tube, servo outputs and the RC inputs. This shielding is shown in Figure 11.6.

![Copper Shielding](image)

**Figure 11.6: Copper Shielding**

Ferrite beads were used on power leads (as shown in Figure 11.7) in order to lessen the effect of EMI from power signals. These beads attenuate the interference generated by the flowing current. A graph of the attenuation obtained at different radio frequencies is shown in Figure 11.8 which also shows the effect of shielding the power leads as discussed in Section 14.3. The data used to construct the graph considers each bead to be the size of a small pea. Therefore, the data for 30 beads is a more accurate representation of the system implemented as the beads are approximately 2cm long.

![Ferrite Beads](image)

(a) Ferrite Bead  (b) Ferrite Bead as Installed on Motor Battery

**Figure 11.7: Ferrite beads**
This shows that ferrite beads will produce 26dB attenuation of noise due to current flow for the 36MHz RC equipment.

New receiver mountings were trialed in order to separate the receiver from sources of interference. This included mounting the receiver in the tail section of the UAV, further away from the other on-board electronics but closer to the other antennae, as well as trialling a mounting further forward in the fuselage, further away from the other antennae, but closer to the batteries and ESC. Attempts were made to neaten the on-board wiring, in particular in order to separate the power lines from control signals wherever possible. These changes in configuration however, produced only a minor positive effect on the overall range of the RC receiver.

While the product of these measures did improve the range of the 36MHz transmitter, the increased range was not sufficient in order to safely continue integration of the autopilot into the airframe. Therefore alternate RC systems were considered and a new system was purchased. The DX7 Spektrum RC controller, (Figure 11.9), was chosen which transmits on the 2.4GHz Spread Spectrum radio system by utilizing the DSM2 Digital Spread Spectrum modulation. This system is less susceptible to radio interference for two main reasons. Firstly, the higher 2.4 GHz spectrum is much less susceptible to EMI issues caused by surrounding electronics. Secondly the system transmits simultaneously on two different frequencies which is enabled by the use of an AR7000 receiver, which enables creating dual RF paths, allowing for a redundancy of transmission path if frequency based interference occurs. Additionally this transmitter is license-free in Australia according to the Radio communications (Low Interference Potential Devices) Class License 2000 (ACMA, 2006), and is classified as a full range transmitter. The transmitter is a 7 channel system which surpasses the required 6 channels, however an equivalent 6 channel, full range transmitter does not exist.
The Spektrum system was tested and proved to be less susceptible to interference. However due to the higher frequency of operation, the system is more sensitive to obstructions requiring line of sight to be maintained. Also, testing showed that the system also has a lower standard range when compared to the JR XP-6102 system. The orientation of this transmitter’s antenna is also very significant, as there is a significant dead zone when the antenna is pointed towards the aircraft. Therefore the transmitter should be held such that the antenna is perpendicular to the line between the transmitter and the aircraft. Despite these drawbacks, the Spektrum system provides a significantly longer range in the presence of interference than the JR XP-6102 system.

The Spectrum transmitter was used in the second autopilot integration flight which proved a partial success. Demonstration of the full ground range of the transmitter to the RC pilot resulted in a comment that the range was lower than desirable, although sufficient to fly. The unexpected drop-out in RC communication that occurred late in the flight was well within the expected range (refer to Section 11.6) shown by the full range ground test. This shows that all issues relating to RC dropouts have not yet been successfully resolved. Further work is required in this area before further autopilot integration can occur.
Chapter 12

UAV Outback Challenge

The UAV Outback Challenge was a competition that was held in Kingaroy, Queensland in late September. This inaugural competition was run by the Australian Research Centre for Aerospace Automation (ARCAA, 2007) which is a joint initiative in aerospace automation between Queensland University of Technology (QUT) and CSIRO. The technical objectives of the competition have been summarised in Section 2.4. For the Search and Rescue Challenge the entrants were to search a predetermined area, find a lost bushwalker and deliver a small rescue package in the form of a 500 ml bottle of water. The UAV iSOAR was one of 28 initial entrants from 7 countries and one of four final competitors that attended the event. The University of Adelaide was one of only two final university entrants.

In the week leading up to this competition it was clear that the communications issues could not be solved in the available time frame. The project group decided to not risk the airframe and thus stopped further integration and testing of the autopilot system. This resulted in the project group not be able to complete the UAV Outback Challenge and was forced to withdraw. Despite this it was decided to attend the Challenge to display the aircraft and raise awareness of the capability students from the Aerospace and Mechatronic Engineering programs at the University of Adelaide. During the first day of the UAV Outback Challenge the airframe was on a static display. Interest in the aircraft was high, with representatives from QUT, ARCAA, Boeing Australia and CSIRO all making positive comments on the aircraft and the project alike. The project group was also able to engage locals and high school competitors during this static display and discuss design and testing issues and experiences.

The second day was the Search and Rescue challenge day where the final two competitors had the chance to prove their systems. Due to the complexity and timeframe of the challenge no competitors were able to prove autonomous flight. Fortunately there was time for the project group to prove the flying capabilities of the airframe at the competition arena. The pilot was instructed to complete three circuits maintaining a height of approximately 200ft, after which time he was to activate the payload deployment switch. During this process live streaming video was achieved, in which people could be spotted from the ground. The aircraft performed extremely well in adverse conditions with wind speeds of approximately 12 to 15kts. Upon a successful landing of the aircraft the project group was approached by the head judge of the
competition, and President of Boeing Australia, David Withers who commented on the flying qualities of the airframe and the success of the project.

All group members who attended the UAV Outback Challenge found the experience rewarding and fulfilling. The group was successfully able to promote Aerospace and Mechatronic Engineering at the University of Adelaide and prove the flying qualities of the UAV iSOAR. A special mention of all the ARCAA support staff was deserved, in particular the competition liaison Lennon Cork, for organising an excellent event and making the group feel welcome.
Chapter 13

Management and Finances

13.1 Work Breakdown

In order to distribute the technical tasks that were required a simplistic breakdown of tasks and associated distribution was developed. Work was distributed to four groups of two engineers; namely airframe - aerodynamics and propulsion, airframe - structures and manufacturing, control systems and imaging systems. From this tier work was further divided within the pairs. Once a group of technical work was completed it was cross checked by the other member of the group for quality assurance and validity. All work was then reviewed at the weekly project meeting to ensure the quality and monitor progress.

13.2 Management Breakdown

Due to the magnitude and technical complication of this project an integrated project management structure was incorporated into the project group. The breakdown of this structure is detailed below.

Technical manager The technical manager was responsible for the day to day management issues; including all technical issues, task distribution, progress monitoring, internal communication, setting of deadlines, and the reallocation of resources.

Business manager The business manager was responsible for the placing and tracking of all major purchases and acquisitions, promotion of the project through local media and recording of all meeting minutes.

Electronic systems manager All electronic technical issues were managed by the electronic systems manager. This person was in constant consultation with the Technical manager.
Manufacturing manager  Understanding the scale and complexity of the manufacturing task required is own dedicated manager. This person was responsible for the progress and tracking of manufacturing processes, and ensuring that the manufacturing progress was conducted on time.

Electronic systems integration manager  The fitting of electronic components required a person monitor and develop tasks to integrate electronic systems into the airframe. This was conducted under the supervision of the technical and manufacturing manager to ensure the structural integrity of the airframe was maintained throughout the integration process.

Chief flight test engineer  The chief flight test engineer was responsible for the mitigation of operational risk through successful following of detailed flight test documentation.

The integration of this detailed management structure has ensured the flow of the project progress from specification, concept, detailing and embodiment, manufacture and operation of the system, and ensures forward progress.

13.3 Milestone Specification

The specification of deadlines and milestones was predominately defined externally by the University and the ARCAA UAV Outback Challenge. A detailed gantt chart was developed which broke up the required tasks and set durations and deadlines in order to meet the requirements of both external deadlines and internal deadlines. Internal milestones were set at weekly meetings in order to ensure the progress of the project such that the results could be delivered. The detailing of the deadlines can be seen in Figure 13.1 (a) and the details of the gantt chart can be found in Figure 13.1 (b).
13.4 Financial Review

The funding required to complete this project was well above that provided for final year projects. As a result, funds were sourced from both external sources and additional sources within the University. Two approaches were taken to obtain funds for this project, one involved sponsorship while the other utilised the project as a marketing tool.

Sponsorship applications were prepared and submitted to various organisations, primarily those with interests in aviation or engineering. Two of these applications were successful and the majority of the budget for this project was obtained from the Sir Ross and Sir Keith Smith Fund and Thales Australia.

The project was the recipient of The University of Adelaide Open Day Innovation Fund. Conditions of acceptance of this prize required the aircraft to be on display at the University Open Day and promote the project to the general public, in particular potential students in the field of Engineering. This exposure was well aligned with the mission statement of the Sir Ross and Sir Keith Smith Fund and with the philosophy of the project group members.

Table 13.1 shows a basic breakdown of the funds spent on this project. These funds totalled
to $26,200, this included taking the aircraft to Queensland for display in the ARCAA, UAV Outback Challenge.

Table 13.1: Financial Summary

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<thead>
<tr>
<th>Primary System</th>
<th>Cost</th>
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<tr>
<td>Control System</td>
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<tr>
<td>Power Plant</td>
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</tr>
<tr>
<td>Airframe</td>
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</tr>
<tr>
<td>Imaging System</td>
<td>$2,000</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td><strong>$23,000</strong></td>
</tr>
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</table>

<table>
<thead>
<tr>
<th>Additional Expenditures</th>
<th>Cost</th>
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</thead>
<tbody>
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<td>Competition costs</td>
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<tr>
<td>Miscellaneous</td>
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<tr>
<td><strong>TOTAL</strong></td>
<td><strong>$3,200</strong></td>
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</tbody>
</table>

13.5 Internal Labour Costs

This section details the labour costs due to the engineering design, development and manufacture labour of the project group members. Table 13.2 shows the breakdown in labour and the corresponding estimated cost based on standardised estimates for graduate engineers. Additional labour costs from workshop and additional third party support was not included by was also considerable.

Table 13.2: Estimated labour expenses

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<tr>
<th>Author</th>
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<th>Direct</th>
<th>Indirect</th>
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<td>$5,092</td>
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<td>Benjamin Chartier</td>
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<td>$8,092</td>
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<td>Brad Gibson</td>
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Chapter 14

Conclusion

This chapter summarises the outcomes of the project. This includes an analysis of the primary and extended project goals, a list of the major issues encountered, recommendations and potential future work.

14.1 Project Definition and Specification

The project definition was to design, build and test a Search and Rescue UAV. The ARCAA UAV Outback Challenge event was used to demonstrate the capabilities of the UAV and thus the system had to meet the associated competition requirements. The global requirements for the development of a UAV were decomposed using a systems engineering approach from which the goals of the project were identified. Research and analysis of the global requirements indicated that fully autonomous flight and autonomous image processing were not trivial tasks and hence the project goals were specified appropriately. The following is a reflection on the results of each of these goals, both primary and extended.

Primary Goal Specification and Analysis

The primary goals of the project were identified well before the design phase was initiated. Each of these goals have subsequently been met to various levels and are discussed below.

1. Development of a UAV platform capable of being flown via remote control, and which has the potential for being configured for autonomous flight and civil surveillance purposes.

Design and development of a UAV platform was successfully carried out resulting in the construction of a prototype platform. The majority of the construction was carried out by group members with the first flight test being a complete success on the 24th August. The aircraft is fitted with an electric motor and has the potential for mounting a camera and an external payload, this meets the requirements for civilian applications. In addition the aircraft has provisions for fitting an autopilot and achieving autonomous flight. This goal has been successfully met.
2. *Fitment of surveillance equipment to UAV platform capable of capturing imagery observed by the UAV during flight.*

Extensive research was carried out to source an off the shelf imaging system capable of meeting the project’s requirements. An analogue system, capable of live streaming footage, was purchased and integrated into the platform. A flight test on the 29th of August resulted in live footage from the onboard camera being streamed to a computer and saved for further viewing. Hence, this goal was also successfully met.

3. *Development of a deployable payload system for the UAV, capable of carrying approximately 600g, and delivering it to a specified target.*

A payload deployment system was designed and integrated into the aircraft such that varying payloads could be attached to the platform. On the 3rd of September a bottle containing 600mL of water was deployed from the aircraft whilst in a normal flight regime at approximately 150 feet. This deployment was activated remotely and delivered to an estimated target. This goal was successfully met.

4. *The integration of an auto-pilot system, which allows the vehicle to maintain straight, horizontal flight autonomously.*

An off the shelf autopilot system was ground tested and integrated into the aircraft for initial autonomous flight testing. On Tuesday the 18th of September control of the aircraft was switched from RC to autopilot for no less than 8 up wind legs of a circuit. Fine tuning of the autopilot resulted in straight and level flight. This goal has been successfully met.

**Extended Goal Specification and Analysis**

The following reflects on the extended goals of the project. These goals were considered throughout the project and often achieved, however, the success of the project was not directly reliant completing these goals.

1. *Encourage continued undergraduate and postgraduate development of UAVs at the University of Adelaide.*

Interest in UAV design and development at the University of Adelaide has been successfully promoted both internally and externally to the University. Featuring in The University of Adelaide Open Day and a presentation to 3rd year design engineering students generated interest in further developing UAVs at the University. In addition to this the group’s involvement in the C2C, concept to creation, program for high school students has encouraged continued development for future undergraduates. Thus this goal has been successfully met.

2. *Automation of the UAV platform for all flight regimes through the implementation and configuration of various UAV autopilot componentry externally sourced.*
Due to time limitations and difficulties with the communications equipment this goal was not met. Straight and level flight was achieved, however the risks involved in further testing were deemed too great until the interference issues are resolved. This is an area for future work.

3. Development of a surveillance system for a UAV which can stream to a ground based station and allow for autonomous search and identification of ground based targets.

The imaging system integrated into the platform is capable of streaming live footage to a ground station and has successfully done so on two separate occasions. Identification of ground based targets autonomously through image processing has proved to be outside the scope of this project and was not achieved. This goal has been partially met.


Development of a UAV that has flown autonomously has been successfully achieved, however the level of autonomy required for the competition was not achieved due to time constraints. The UAV was displayed and flown by remote control with live footage captured and a payload deployed at the competition in Kingaroy, Queensland. This goal has been partially met, however considering the complexity of the challenge and the performance of other competitors it is clear that the groups design was well advanced.

5. World recognition for aeronautical research at the University of Adelaide.

The research and development of a UAV at the University of Adelaide has received wide media coverage. Several radio interviews and print coverage in the Adelaidean, Messenger, Advertiser and The Australian has resulted in national recognition. In addition television coverage including interviews on Channel 7 news and coverage on the Sunrise program has reinforced this recognition. This goal has been partially met.

14.2 Issues

Throughout the course of this project there were many practical difficulties encountered that hindered progress. Some of these issues were resolved while alternative solutions were required in some areas. A summary of these issues are included below.

Mechanical

Experience

- The construction of a UAV requires knowledge and experience in specialised fields. The use of composite materials and the manufacturing techniques were unfamiliar to most group members. The authors suggest to obtain as much advice and assistance as possible when manufacturing an airframe of this nature.
CHAPTER 14. CONCLUSION

Airframe weight

- The aircraft was designed with a maximum takeoff weight of 9kg. This value is a combination of known weights and estimates. The estimates were in the area of airframe manufacture, where a lack of experience, and technique resulted in the aircraft being approximately 10% overweight, in the fully laden configuration.

Production time

- Throughout the construction of the airframe delays were encountered with curing times of the composite materials. Low temperatures experienced throughout the day and night resulted in the epoxy resin requiring additional days to cure before further work could be carried out, this was not accounted for in the time estimate for construction. The result was an airframe that was test flown approximately three weeks later than planned.

Electronics Issues

Utilising Overseas Components - Import Time and Support

- The time required to import many off-the-shelf control systems is significant, as is the difficulty maintaining regular useful contact with overseas support staff. The import time limited the number of potential systems and restricted the time available to test and implement the selected control system. One of the main factors in choosing the Micropilot 2028g autopilot was the lowest import time, however this import process still required over 2 months between initiating negotiations and receiving the autopilot. This is not a reflection on the company, but rather the security checks that were required as autopilots are considered to be potentially dangerous items. In addition, the restrictions placed on contacting Micropilots support staff due to the time difference caused many simple issues to take an extended amount of time, as any response was effectively received the following working day.

Programming

- While the programming of the autopilot initially appeared simple, issues were encountered in programming specific tasks, such as the safety system detailed in Section 9.5. The programming issues were due to the inflexibility of the proprietary system and inadequate documentation provided.

Communications Interference

- Initially the effect of interference in communication devices was not given appropriate attention. A focus was maintained on separation of transmission frequencies, but other causes of interference not considered. This false assumption resulted in an airframe crash and significantly impeded the progress of the project.
14.3 RECOMMENDATIONS

Imaging Systems

- In order to identify a person on the ground a high resolution camera is desirable. However, this must be balanced against the limitations of transmitting images over the UAVs operational range. This significantly limited the variety of potential cameras for the imaging system. Another trade off had to be made between operational speed and altitude. In order to view the required area in a given time, the speed or altitude of flight must be increased. Both of these measures will limit the ability to identify a person on the ground.

- Legality issues had to be considered in selecting the communications equipment. In order to transmit video over the required distance a relatively high power transmitter is desirable. However, unlicensed transmission power is limited to as little as 100mW at some frequencies. Therefore legislation had to be analyzed to select a modem of appropriate power and frequency.

14.3 Recommendations

Throughout the project some tasks were neglected or not addressed even though the goals of the project were achieved. Some of these tasks have the potential to advance the capabilities of this UAV system and have been broken down into airframe, imaging systems, control systems and communications.

Airframe

Parachute recovery system

The development and testing of a parachute recovery system was completed in the course of this project however, a flight test of this system was never completed. The authors suggest the testing of this system should be conducted in the following order:

- Vertical set test of the parachute with different weights to determine set time and maximum vertical distance of travel during deployment to ensure the

- Further ground testing from a vehicle moving at speeds in the order of actual deployment. This would involve the mounting of the fuselage structure to the vehicle and remotely activating the parachute deployment. For these tests the parachute would not be attached to the fuselage, as the goal is to determine whether the deployment of the chute is viable in the current fuselage structure with the current pilot parachute.

- Initial flight tests of the parachute deployment. This is expected to be conducted with minimal componentry on board and a relatively low velocity during deployment to minimise the opening loads experienced and ensure reliability of operation. Once confidence has been gained in the parachute recovery system then further advancement
Car launch

Development of the car launch system could potentially increase the flexibility of operation of this UAV, as it removes the need for a traditional runway for take off. Combining this with the aforementioned parachute recovery system this would make this UAV system truly versatile. The development of this system would involve the following:

- Manufacture of the proposed car launch design.
- Extensive ground testing of the launch mechanisms. This would include the testing of the kinematics of the linear actuators and their effect on the airframe. The criticality and potential unsafe operation of this mechanism can not be over emphasised by the current project group. Great care must be taken when considering the operation, and the preparation of safety documentation must be rigorous. Ground testing with the aircraft on the launch mechanism would prove the viability of this system.
- Flight testing of this system forms the final stage of this development process. Further developments of the automation of this process is discussed in the control systems future work.

Deployable payload

This work requires the refinement of the deployable payload structure. Greater aerodynamic efficiency and aesthetics can be achieved with a design iteration. The developed system was only ever intended to be a prototype, with a short development time to ensure functionality of operation, with little consideration given to aesthetics or efficiency.

Control Systems

To achieve fully autonomous waypoint following the autopilot requires further in-flight tuning of PID loops. This will require an estimated 4 further test flight days assuming no further issues occur. The method for tuning the autopilot is discussed in Appendix H.

The performance of some parameters of the aircraft, such as cruise speed, have not been quantified. These parameters should be measured using the autopilot system to record data and compare the aircraft performance results with the expected values. The parameters to be investigated include but are not limited to; cruise speed, maximum speed, climb rate, take off speed and approach speed.

Imaging Systems

Noise was present intermittently in the received image due to the limited sweep of the directional antenna. This issue could be overcome with an improved antenna mount. This antenna mount would need to elevate the antenna to a height of at least 6 metres, while allowing adjustment of
the antenna’s orientation. Thus the antenna could be altered to point at the UAV throughout its flight, increasing the quality of the received image.

In order to successfully complete the UAV Outback Challenge the GPS location of a lost bush-walker must be obtained. Therefore linking of the GPS data to the images received must be accomplished. This function could be performed using commercially available video overlay boards available from companies such as Intelligent Flight, Decade Engineering or BlackBoxCamera.

**Communication Equipment**

Significant progress has been made to solve the interference issues encountered during this project. However, further work is required in this area. These recommendations have been compiled in order to assist future projects in this area. The authors suggest the following considerations should be implemented in order to resolve these issues.

1. Replace current servo connections with twisted, shielded servo leads. Servo leads can act as antennas receiving and transmitting interference. This effect can be reduced by twisting the leads. Additionally, servo leads are sometimes shielded around each wire as well as around the whole lead.

2. Implement EMI suppressant tubing over all high current connections. EMI suppressant tubing will reduce the interference caused by power connections, such as from the powerplant, control, imaging and servo batteries. Figure 11.8 shows that EMI suppressant tubing is effective for all radio signals above 5MHz and will provide effective shielding for frequencies not attenuated with ferrite beads.

3. Implement further grounded, copper shielding around all electrical equipment, where practicable. This includes the Autopilot, RF Modem, imaging modem, engine batteries, engine and Electronic Speed Controller. Shielding can be constructed from copper or aluminium foil as well as other materials like carbon fiber. In order to be effective shielding must be electrically grounded to a common ground. This common ground must be established through connecting the negative terminal of all onboard power supplies.

4. Eliminate carbon fiber from the airframe design. In particular the large sections of carbon fiber on the hatch and landing gear will retard communications equipment, so alternatives should be considered.

5. Consider the use of an amplifier for the RC transmitter. A commercially available RC amplifier would be very beneficial if such a device can be sourced. A second prototype of these structural components should be considered.

6. Consider the use of a 6dB gain antenna for the RF modem. The current antenna for the control systems only has a unity gain, but the modem will support up to a 6dB gain antenna.
After implementing one or a number of the above measures testing should be conducted to determine their effectiveness. Testing should consider all orientations of the UAV and be conducted with all onboard systems operating.

14.4 Future Work

Throughout the duration of the project ideal systems were identified, although limitations in time, finances and experience inhibited the groups ability to attain these systems. The recommendation section detailed tasks that, ideally, would have been achieved during the project however this section identifies the areas that could be improved or further investigated in the future. As before, these have been broken down into airframe, imaging systems, control systems and communications.

Airframe

The driving parameter for the design of this aircraft was the estimated maximum take off weight. The challenge was in the manufacture of the airframe to meet this limitation. This aircraft was 12% overweight from the design value of 9kg. The extra weight was identified as coming from the manufacturing process of the airframe. Whilst the airframe sufficiently carried the design loads, the magnitude of the critical reserve factors is not known with certainty. It is likely that the airframe was over engineered for the application in which it was used.

Reconstruction of the airframe could address this issue through a more detailed structural analysis of the airframe in the design phase. A thorough structural analysis of the ring frames and entire internal structure, along with the integrity of the connections could be calculated with greater accuracy and allow a reduction material used while maintaining the appropriate safety factors. The result of this would be a reduction in weight. The torsion analysis on the wings disregarded the transfer of load through the spars, this was another conservative assumption that lead to the requirement of three layers of 85gsm fibreglass for the skin. Greater research and understanding of the materials used would result in a reduced take off weight through more accurate structural analysis on the entire airframe.

The manufacturing process has also highlighted the effect of material selection on aerodynamic performance. The finished profile of the wings differed marginally from that of the design and desired profile. This was a result of the manufacturing materials and methods used. Cutting the foam core to the profile of the design and not allowing for skin thickness, coupled with the hand crafting and finishing method used, resulted in a greater t/c ratio. This in turn increased the lift coefficient and hence the drag, reducing the optimized design of endurance. Another set of wings should be manufactured allowing for a skin around the foam and hence achieving the desired t/c ratio. In addition a more detailed structural analysis may remove the need for a heavy fibreglass skin, once again reducing the weight of the aircraft.
Control Systems

Enabling autonomous take-off and recovery would allow for fully autonomous implementation of flight paths, allowing flight of the aircraft without the need for an RC pilot. This work is conditional of the development of the car launch mechanism and the proof of the parachute recovery system, or the purchase of an ultrasonic altitude sensor enabling accurate altitude control. Successful completion of the launch and recovery systems would allow for automation of both tasks using the combination of additional features of the Micropilot 2028 and analysis of RC maneuvers to program the take-off routine. If the car launch mechanism and parachute recovery system is not successful developed, this automation could still proceed, but would require the the ultrasonic altitude sensor from Micropilot to enable automated take off and landing using the undercarriage.

Imaging Systems

An infrared camera or lens would greatly improve the imaging systems ability to identify a person on the ground. This may also enable the automation of the imaging system. This requires a program to identify a pixel pattern and colour corresponding to a human on the ground. Ideally this program could determine the GPS location of the person and initiate an automated payload drop. Such a system could be reprogrammed, enabling iSOAR to autonomously perform a myriad of different applications.

Communication Equipment

Reducing the number of communication links using routers onboard the UAV and in the ground station could be achieved. This will require considerable time for implementation, however this simplification of communications could be very beneficial as it eliminates the mode of interference. A single communications link is implemented in the Piccolo Autopilot. An example custom built signal router has been implemented in a similar project at the Mississippi State University (Borries, 2006).

14.5 Post Analysis

The efforts of the project group has resulted in the successful development of an Unmanned Aerial Vehicle system. Given the scope, timeframe and man-power of the project there is very little that the group would do differently. The learning curve has been steep yet the process has been very satisfying for all involved.

The Search and Rescue UAV project has been a significant educational and technical success. The project group members have gained technical and non-technical knowledge in all phases of a holistic engineering task.
References


REFERENCES


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REFERENCES


REFERENCES


Appendix A

Main Wing Airfoil Selection

The selection of the airfoil profile for the wing involved the analysis and comparison of the performance of six different airfoil profiles. These six were:

- The SD 7032
- The SD 7037
- The SD 7062
- The Liebeck LA2573
- The S1210
- The Eppler 423

The comparison used performance results obtained from ‘JavaFoil’ airfoil analysis software, and the validity of this method was checked through the comparison of JavaFoil results with published wind tunnel data for airfoils in flows of different $Re$.

The profile found to be the most suitable for the UAV was the SD 7032. However, if the configuration and requirements of the vehicle were changed in the future, one of the five other profiles may prove to be the more suitable. Hence, the purpose of this section is to provide justification of the airfoil selection, as well as to provide information, which may assist further aerodynamic development of the airframe.

A.1 Airfoil Shape Plots From Selig (2006)

For reference for future aerodynamic work, the profile shapes of the six compared airfoils are presented below.
APPENDIX A. MAIN WING AIRFOIL SELECTION

Figure A.3: Cross Section of Airfoil (SD 7062)

Figure A.1: Cross Section of Airfoil (SD 7032)

Figure A.2: Cross Section of Airfoil (SD 7037)
A.1. AIRFOIL SHAPE PLOTS FROM?

Figure A.4: Cross Section of Airfoil (LA2573)

Figure A.5: Cross Section of Airfoil (S1210)

Figure A.6: Cross Section of Airfoil (Eppler 423)
A.2 Verification Results

This section presents the results of the process undertaken to verify the results of JavaFoil. Published wind tunnel data for the SD 7032, SD 7037, and Eppler 423 was obtained, and compared with results from JavaFoil for these profiles. As Table A.1 shows, in conjunction with the presented plots, a high level of agreement exists between the results of JavaFoil and the published wind tunnel test data used for the verification tests.

<table>
<thead>
<tr>
<th>Airfoil</th>
<th>Re (E+05)</th>
<th>(C_{l_{\text{Max}}}(\text{Wind Tunnel}))</th>
<th>(C_{l_{\text{Max}}}(\text{JavaFoil}))</th>
<th>% Difference</th>
</tr>
</thead>
<tbody>
<tr>
<td>SD7032</td>
<td>2.5</td>
<td>1.388</td>
<td>1.426</td>
<td>2.737752</td>
</tr>
<tr>
<td>SD 7032</td>
<td>5</td>
<td>1.398</td>
<td>1.441</td>
<td>3.075823</td>
</tr>
<tr>
<td>SD 7062</td>
<td>2.5</td>
<td>1.4319</td>
<td>1.554</td>
<td>8.527132</td>
</tr>
<tr>
<td>SD 7062</td>
<td>5</td>
<td>1.4668</td>
<td>1.581</td>
<td>7.785656</td>
</tr>
<tr>
<td>Eppler 423</td>
<td>NA</td>
<td>NA</td>
<td>NA</td>
<td>NA</td>
</tr>
</tbody>
</table>

The discrepancies in lift coefficient values at maximum airfoil angle of attack were found to fall between 2.7% and 7.8%, and the polar plots were also found to be approximately equal in terms of shape and values. It is because of this high level of agreement in these results, that the use of JavaFoil for the airfoil selection process was deemed acceptable.

The results and data for each particular airfoil is grouped and presented in the following sub sections. The published data for the SD 7032 and SD 7037 was obtained from Jacob (2003), and the results for the Eppler 423 were obtained from Eppler (1990).
A.2. VERIFICATION RESULTS

A.2.1 Results for SD 7032

Figure A.7: Polar Plot Comparison Verification Re = 2.5E+05 (SD 7032)

Figure A.8: Polar Plot Comparison Verification Re = 5.0E+05 (SD 7032)
Figure A.9: Lift Coefficient versus Angle of Attack Comparison Verification Re = 2.5E+05 (SD 7032)

Figure A.10: Lift Coefficient versus Angle of Attack Comparison Verification Re = 5.0E+05 (SD 7032)
A.2. VERIFICATION RESULTS

A.2.2 Results for SD 7062

Figure A.11: Polar Plot Comparison Verification Re = 2.5E+05 (SD 7062)

Figure A.12: Polar Plot Comparison Verification Re = 5.0E+05 (SD 7062)
Figure A.13: Lift Coefficient versus Angle of Attack Comparison Verification $Re = 2.5E+05$ (SD 7062)

Figure A.14: Lift Coefficient versus Angle of Attack Comparison Verification $Re = 5.0E+05$ (SD 7062)
A.3. AIRFOIL COMPARISON

A.2.3 Results for Eppler 423

The two dimensional flow results obtained from JavaFoil for the comparison of the six airfoils is presented in this section. Polar plot comparisons and section pitching moment coefficient plots are presented below for flows with $Re$ equal to $1 \times 10^5$, $2 \times 10^5$, $3 \times 10^5$, $4 \times 10^5$, $5 \times 10^5$, and $6 \times 10^5$. These plots show that the SD 7032 and SD 7037 provide the best combinations of section lift, section drag, and section pitching moment performance, in accordance with the requirements specified for the main wing airfoil, which were:

- Suitable for operation in flow with $1.0 \times 10^5 < Re < 6.0 \times 10^5$, (expected range for vehicle operation)
- Able to provide the desired maximum wing lift coefficient, $C_{L_{max}} = 1.2$, from conceptual design phase
- Have the highest possible $\frac{L}{D}_{wing}$ (from identified airfoils,) to allow aircraft to achieve highest possible $\frac{L}{D}_{vehicle}$
- Low, constant $C_{m}$ to reduce torsional loads and induced drag from trimming

Hence these plots demonstrate why these two profiles were identified as the two best choices for the wing design, and were subsequently compared using three dimensional flow analysis, to
ultimately identify the most suitable profile for the design of the UAV, (the SD 7032 in this case.)

As mentioned, this data is not only presented for justification of the final airfoil selection. It is also presented as reference material, suitable for use if the future development of the airframe includes altered specifications of the aerodynamic performance of the main wing.

A.3.1 Airfoil Polar Plot Comparisons

Figure A.16: Polar Plot Comparison Re = 1.0E+05
A.3. AIRFOIL COMPARISON

Figure A.17: Polar Plot Comparison $Re = 2.0E+05$

Figure A.18: Polar Plot Comparison $Re = 3.0E+05$
Figure A.19: Polar Plot Comparison Re = 4.0E+05

Figure A.20: Polar Plot Comparison Re = 5.0E+05
A.3. AIRFOIL COMPARISON

A.3.2 Section Moment Coefficient (Quarter Chord) Plots for Compared Airfoils

Figure A.21: Polar Plot Comparison Re = 6.0E+05

Figure A.22: Section Moment Coefficient (Quarter Chord) versus Angle of Attack (SD 7062)
Figure A.23: Section Moment Coefficient (Quarter Chord) versus Angle of Attack (SD 7032)

Figure A.24: Section Moment Coefficient (Quarter Chord) versus Angle of Attack (SD 7037)
Figure A.25: Section Moment Coefficient (Quarter Chord) versus Angle of Attack (LA 2573)

Figure A.26: Section Moment Coefficient (Quarter Chord) versus Angle of Attack (S 1210)
Figure A.27: Section Moment Coefficient (Quarter Chord) versus Angle of Attack (Eppler 423)
Appendix B

Fuselage Load Calculations

For further analysis of the stresses in the fuselage an estimate of the maximum tail loads, weight distribution, wing loads, motor thrust and torque were required. This appendix details the calculations of these loads and the development of load distributions. Each of the subsequent calculations used a load factor, \( n \), of 3.8 and safety factor, \( n_s \), of 2.25. A free body diagram of the aerodynamic loads on the aircraft is shown in Figure B.1.

![Figure B.1: FBD of Applied Loads to the Fuselage](image)

Notation

The following notation is used throughout this appendix.

- \( g \): acceleration due to gravity
- \( i_h \): incidence angle of the horizontal tail
- \( i_w \): incidence angle of the wing
- \( m \): mass of aircraft
n load factor
q dynamic pressure
$C_{LH}$ lift coefficient of the horizontal tail
$C_{LV}$ lift coefficient of the vertical tail
$L_H$ lift of the horizontal tail
$L_V$ lift of the vertical tail
$M_{reaction}$ reaction moment required for static analysis
$S_H$ horizontal tail area
$S_V$ vertical tail area
$X_{ac}$ lever arm of aerodynamic centre of main wing
$X_{cg}$ lever arm of centre of gravity
$X_H$ lever arm of aerodynamic centre of horizontal tail
$\alpha_h$ angle of attack of the horizontal tail
$\alpha_w$ angle of attack of the wing
$\frac{d\varepsilon}{d\alpha}$ downwash derivative

B.1 Horizontal Tail Loads

Using the effective angle of the horizontal tail the effective load can be determined. This analysis has been completed using simplifications developed by Simons (2002). Having designed the tail plane with a control surface which is 30% of the mean aerodynamic chord the incremental tail plane lift coefficient can be estimated to be 0.87. Simons also suggested that the effective downwash derivative is approximately 0.415 thus the effective angle of the horizontal tail can be calculated to be -0.714deg, from Equation B.1.

$$\alpha_h = \alpha_w \left(1 - \frac{d\varepsilon}{d\alpha}\right) + i_h - i_w$$ (B.1)

This corresponds to a nominal lift coefficient of 0.08 for the horizontal tail. The lift of the horizontal tail can now be estimated using the combination of these two lift coefficients.

$$L_H = C_{LH}qS_H$$

The vertical tail force can also be calculated in a similar fashion.

$$L_V = C_{LV}qS_V$$

The loads on the horizontal and vertical tail have been calculated to be 58.8 N and 20.4 N respectively.
B.2 Weight Loads

A database of all onboard components and their corresponding mass was used to estimate the weight loads. Using estimates for the airframe structure the weight of these components was also estimated to generate the weight loading of the aircraft. A spreadsheet of the component weights and locations is shown in Table B.1.
### Table B.1: Component Weights and Corresponding Locations from the Nose

<table>
<thead>
<tr>
<th>System</th>
<th>Component</th>
<th>Mass (g)</th>
<th>Lever Arm (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power Plant</td>
<td>Motor</td>
<td>409</td>
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</tr>
<tr>
<td></td>
<td>ESC</td>
<td>50</td>
<td>0.12</td>
</tr>
<tr>
<td></td>
<td>Battery 1</td>
<td>1053</td>
<td>0.262</td>
</tr>
<tr>
<td></td>
<td>Battery 2</td>
<td>1053</td>
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</tr>
<tr>
<td></td>
<td>Battery 3</td>
<td>702</td>
<td>0.577</td>
</tr>
<tr>
<td>Auto Control</td>
<td>Autopilot</td>
<td>28</td>
<td>0.386</td>
</tr>
<tr>
<td></td>
<td>RF modem (AP)</td>
<td>75</td>
<td>0.22</td>
</tr>
<tr>
<td></td>
<td>Servo board</td>
<td>12</td>
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</tr>
<tr>
<td></td>
<td>AP battery</td>
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<td>0.714</td>
</tr>
<tr>
<td></td>
<td>GPS antenna</td>
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</tr>
<tr>
<td></td>
<td>GPS ground plate</td>
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<td>1</td>
</tr>
<tr>
<td></td>
<td>AP antenna</td>
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</tr>
<tr>
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<td>Camera 2</td>
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<tr>
<td></td>
<td>Camera battery</td>
<td>182</td>
<td>0.714</td>
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<td></td>
<td>Camera transmitter</td>
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<td></td>
<td>Deployable</td>
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<td></td>
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<td></td>
<td>Vert tail</td>
<td>78</td>
<td>1.4284</td>
</tr>
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<td></td>
<td>Elevator servo</td>
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<td>Rudder servo</td>
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<td></td>
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<td></td>
<td>Rear ring frame</td>
<td>9</td>
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<tr>
<td></td>
<td>Top floor</td>
<td>50</td>
<td>0.48</td>
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<td></td>
<td>Bottom floor</td>
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<td></td>
<td>Nose section</td>
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<td></td>
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<td></td>
<td>Tail section skin</td>
<td>208</td>
<td>1.0308</td>
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<td>Wing total (previous sheet)</td>
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<td>0.512</td>
</tr>
<tr>
<td></td>
<td>Total Mass Estimate</td>
<td>7914</td>
<td>g</td>
</tr>
</tbody>
</table>

### B.3 Wing Loads and Reaction Loads

The reaction forces to offset these weight and tail loading are generated by the wing, or are pseudo forces resultant from the acceleration of the aircraft. It is assumed that the reaction
forces are placed at the aerodynamic centre of the wing, which is approximately at the 25% mean aerodynamic. The analysis of a free body diagram has been used to calculate these loads, to achieve static balance, to estimate loads.

**Reaction force**

\[ \sum F_z = 0 \]

\[ L_W = L_H + mg \times n \]

\[ L_W = 350.8N \]

**Reaction moment**

\[ \sum M_{yy} = 0 \]

\[ M_{reaction} = nmg \times X_{cg} + L_W X_{ac} + L_H X_H \]

\[ M_{reaction} = 42.3N \]

### B.4 Development of Bending and Shear Plots

A classical approach has been taken to develop the bending and shear plots along the airframe. The fuselage has been analysed for bending and shear at ten discrete locations along the aircraft. This has been completed using a traditional hand calculation style approach. The results of this analysis can be seen in the Figure B.2.
APPENDIX B. FUSELAGE LOAD CALCULATIONS

B.5 Torsional Loads

The torsional loading of the fuselage is a resultant of the control authority of the vertical tail. This torque has significant implication on the empennage structural integrity. The determination of this load has been made in Section B.1. This force has been multiplied by the local lever arm from the fuselage or empennage centre line. The result of this analysis is shown in Figure B.3.

Note: It is assumed that the shear loading from the vertical tail has been assumed to be negligible. This has been made as the combined control input of both full rudder and full elevator could not be made to sustain loading. This factor may consumed by reserve factors that allow for errors in the analysis.


B.6 Axial Loads

The determination of axial loads is calculated from the worst case scenario of the static thrust. This has been tested in static motor test and thrust values of 4 kg can be expected for static conditions. This loading is counteracted by either landing gear or a launch system, in the worst case scenario. The load plot for the axial loading of the aircraft is shown in Figure B.4.

B.7 Tabulated Loads

The data presented in Table B.2 is the tabulated results of the load determining process.
Table B.2: Summary of all of the Loads Developed from the Above Analysis

<table>
<thead>
<tr>
<th>Fuselage Station</th>
<th>Shear Force (N)</th>
<th>Bending Moment (Nm)</th>
<th>Torsional Load (Nm)</th>
<th>Axial Load (N)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>-0.955</td>
<td>39.24</td>
</tr>
<tr>
<td>0.154</td>
<td>-19.4</td>
<td>1.99</td>
<td>-0.955</td>
<td>39.24</td>
</tr>
<tr>
<td>0.308</td>
<td>-71.1</td>
<td>7.63</td>
<td>-0.955</td>
<td>39.24</td>
</tr>
<tr>
<td>0.462</td>
<td>-146.7</td>
<td>49.4</td>
<td>2.63</td>
<td>19.62</td>
</tr>
<tr>
<td>0.616</td>
<td>110.6</td>
<td>37.7</td>
<td>2.63</td>
<td>19.62</td>
</tr>
<tr>
<td>0.77</td>
<td>71.0</td>
<td>23.8</td>
<td>2.63</td>
<td>19.62</td>
</tr>
<tr>
<td>0.924</td>
<td>64.9</td>
<td>13.4</td>
<td>2.45</td>
<td>2</td>
</tr>
<tr>
<td>1.078</td>
<td>56.3</td>
<td>4.10</td>
<td>2.26</td>
<td>1</td>
</tr>
<tr>
<td>1.232</td>
<td>54.7</td>
<td>-4.42</td>
<td>2.08</td>
<td>1</td>
</tr>
<tr>
<td>1.386</td>
<td>53.3</td>
<td>-12.73</td>
<td>1.89</td>
<td>0.5</td>
</tr>
<tr>
<td>1.54</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>
Appendix C

Empennage Buckling Calculations

Buckling is the failure of a material due to the structural elastic instability. If the applied load is above that of the buckling limitation the structure will fail. In order to determine whether the tail section would buckle the following analysis methodology was used. This method, along with the following figures, is presented in Stress analysis and sizing by Niu (1999), under Chapter 11 for the buckling of thin sheets. The buckling has been analysed using a number of simplifications and approximations. It is assumed that the buckling will cause the top, or bottom, of the tail section, via the combined loading of this section, through the action of a bending moment and shear force, associated with aerodynamic loads. The analysis below, is for the top of the tail section, and incorporates the applied stress loadings. The bending load is taken through compression of the skin and the torque loading results in a shear stress being applied to the skin.

Notation

The notation specific to this Appendix is listed below. This notation has been kept consistent with that defined in Niu (1999).

- $a$ half height of the ellipse
- $b$ width of section / half width of the ellipse
- $k'_c$ compression buckling coefficient
- $k'_s$ shear buckling coefficient
- $r_{\text{max}}$ maximum radius of curvature
- $t$ skin thickness
- $E$ modulus of elasticity
- $F_{c,cr}$ critical compression buckling stress
- $F_{s,cr}$ critical shear buckling stress
- $MS$ margin of safety
- $R_c$ compression stress ratio
$R_s$ shear stress ratio

$\eta_c$ compression plasticity factor

$\eta_s$ shear plasticity factor

$\mu$ Poisson’s ratio

### C.1 Compression Buckling

The loading due to bending acts in compression in the top section of skin of the tail. This can be analysed using Equation C.1 and Figure C.1. This equation gives the critical buckling stress for compression.

$$ F_{c,cr} = \frac{k_c' \eta_c \pi^2 E}{12 (1 - \mu)} \left( \frac{t}{b} \right)^2 $$  \hspace{1cm} (C.1)

### C.2 Shear Buckling

The loading due to torsion, from the vertical stabiliser, acts in shear in the top section of skin of the tail. This can be analysed using Equation C.2 and Figure C.2.

$$ F_{s,cr} = \frac{k_s' \eta_s \pi^2 E}{12 (1 - \mu)} \left( \frac{t}{b} \right)^2 $$  \hspace{1cm} (C.2)

The plasticity factors for the aforementioned equations will not be considered, however will be absorbed in the resultant reserve factor. It is extremely difficult to estimate these properties for a composite material, however for a reserve factor above 1.5 the affect of these factors will no longer be an issue.

A similar stance has been taken for the implementation of a Poisson’s Ratio. This value is hard to define for a laminated structure, so as a conservative estimate this value has been set to zero. This will result in the worst case scenario, and would oppose the assumption of the negligible plasticity.

### Calculation of Buckling Constant

The calculation of a buckling constant, $k_c$ or $k_s$, is required for the above analysis. In order to do this a ’$Z$’ value is required. A smaller $Z$ value implies a smaller buckling coefficient, and thus less resistance to buckling. The value of $Z$ is determined from Equation C.3.

$$ Z = \frac{b^2}{rt} \left( 1 - \mu^2 \right)^\frac{3}{2} $$  \hspace{1cm} (C.3)

As the profile of the top section is an ellipse the maximum radius of curvature occurs at the centreline, or peak, of the aircraft and can be determined using the Equation C.4 from elementary geometry principles.
This was conducted at each division point along the tail section. Using this value of 'Z' the buckling coefficients could be determined accordingly from Figure C.1 and Figure C.2. For the compression buckling coefficient the design values were used as a conservative estimate for an r/t value of 500. As a conservative estimate, for shear buckling coefficient, the value for length on width was assumed to be infinite.

Figure C.1: Curved Plate Compression Buckling Coefficient (Niu, 1999)
C.3 Combined Loadings

To determine the buckling load limitation an alternative method needs to be used. Combined loading due to compression and shear must be combined to give the overall loading. The stress ratios, for both compression and shear, must first be calculated, from Equation C.5 and Equation C.6. These equations relate the ratio of the applied maximum load to the critical buckling load limit, for each compression and shear.

\[ R_c = \frac{\sigma_{\text{bending}}}{F_{c,cr}} \]  \hspace{1cm} (C.5)

\[ R_s = \frac{\tau_{\text{bending}}}{F_{s,cr}} \]  \hspace{1cm} (C.6)

The combination of the compression and shear loads must be below the limit defined below, in order to meet design requirements. Equation C.7 must be satisfied for the structure to be sufficient. Figure C.3 below was helpful in determining this limitation and the associated margin of safety.

\[ R_s^2 + R_c < 1 \]  \hspace{1cm} (C.7)
C.4. RESULTS OF ANALYSIS

Figure C.3: Combined Compression and Shear Loading Graph (Niu, 1999)

Margin of Safety

The margin of safety, for buckling limitations is defined from the combination of the equation below, and is represented on Figure C.3 above. From this equation the margin of safety can be calculated.

\[ MS = \frac{OB}{OA} - 1 \]  

(C.8)

Where:

\( OB \) is the length for the origin to the theoretical buckling limitation

\( OA \) is the length from the origin to the actual buckling load

C.4 Results of Analysis

Table C.1 and Table C.2 detail the results of the aforementioned analysis.

<table>
<thead>
<tr>
<th>X (m)</th>
<th>Z</th>
<th>( k_c' )</th>
<th>( k_s' )</th>
<th>( \sigma_{bending} ) (Pa)</th>
<th>( \tau_{torsion} ) (Pa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.9</td>
<td>144.9</td>
<td>60.0</td>
<td>14.0</td>
<td>18313462.6</td>
<td>106799.5</td>
</tr>
<tr>
<td>1.1</td>
<td>126.8</td>
<td>50.0</td>
<td>13.5</td>
<td>16948680.9</td>
<td>128908.1</td>
</tr>
<tr>
<td>1.2</td>
<td>105.7</td>
<td>40.0</td>
<td>13.2</td>
<td>14409912.7</td>
<td>170497.5</td>
</tr>
<tr>
<td>1.4</td>
<td>79.0</td>
<td>29.0</td>
<td>12.9</td>
<td>7983674.9</td>
<td>277593.7</td>
</tr>
<tr>
<td>1.5</td>
<td>36.4</td>
<td>15.0</td>
<td>11.7</td>
<td>0.0</td>
<td>0.0</td>
</tr>
</tbody>
</table>
Table C.2: Results of Buckling Analysis (Part 2)

<table>
<thead>
<tr>
<th>X (m)</th>
<th>Rc</th>
<th>Rs</th>
<th>Rs^2 + Rc</th>
<th>MS (approx.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.9</td>
<td>0.234</td>
<td>0.006</td>
<td>0.234</td>
<td>4.273841</td>
</tr>
<tr>
<td>1.1</td>
<td>0.199</td>
<td>0.006</td>
<td>0.199</td>
<td>5.025204</td>
</tr>
<tr>
<td>1.2</td>
<td>0.147</td>
<td>0.005</td>
<td>0.147</td>
<td>6.811458</td>
</tr>
<tr>
<td>1.4</td>
<td>0.063</td>
<td>0.005</td>
<td>0.063</td>
<td>15.92973</td>
</tr>
<tr>
<td>1.5</td>
<td>0.000</td>
<td>0.000</td>
<td>0.000</td>
<td>NA</td>
</tr>
</tbody>
</table>

C.5 Conclusion

From the results of the aforementioned buckling load analysis it is safe to say that the tail section will not fail under aerodynamic loading for the given skin thickness of 0.75mm and the specified safety margins. Thus this structure does not require any additional reinforcement to counteract buckling for this skin thickness.
Appendix D

Landing Gear Calculations

This appendix details the process that was used to determine the landing gear locations, and estimate the static loading on each of the gears to give a measure of suitability. This method has followed a process defined by Currey (1988) in order to ensure the suitability of the final landing gear locations.

**Centre of Gravity Envelope** must be determined in order to locate the main landing gear. The range of this point is very important parameter for stability and control of this aircraft. Although the aircraft is an electric powered system and thus should not have any CG travel, it is intended that this aircraft will be flown with alternative internal and payload configurations during initial flight testing until the system becomes fully operational. A summary of the configurations is presented Table D.1 below, along with comments regarding to the internal configurations and the location of the centre of gravity for this set up.

Table D.1: List of Possible Configurations and Correlating CG Position at the Preliminary Design Stage

<table>
<thead>
<tr>
<th>#</th>
<th>Configuration Name</th>
<th>Comments</th>
<th>CG Position (m)</th>
<th>%MAC</th>
<th>Static Margin</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Design</td>
<td>All systems on board</td>
<td>0.461m</td>
<td>21.6%</td>
<td>15%</td>
</tr>
<tr>
<td>2</td>
<td>RC flight test</td>
<td>RC equipment only half power plant batteries and no external payload</td>
<td>0.459m</td>
<td>20.8%</td>
<td>16.1%</td>
</tr>
<tr>
<td>3</td>
<td>Autopilot integration</td>
<td>RC &amp; Autopilot control systems, half power plant batteries and no external payload</td>
<td>0.459m</td>
<td>20.8%</td>
<td>16.2%</td>
</tr>
<tr>
<td>4</td>
<td>Camera integration</td>
<td>All internal electronics, half power plant batteries and no external payload</td>
<td>0.457m</td>
<td>20.0%</td>
<td>17%</td>
</tr>
</tbody>
</table>

These values are an approximation and may be incorrect by due to the mass estimation of construction processes and weight uncertainty in parts. As a result an extra margin of plus
or minus 1% to the aforementioned CG envelope is added to this value to accommodate this uncertainty. This implies that the range of CG travel is 0.455m to 0.463m from the nose of the aircraft. The greatest height of the CG is to be noted. This occurs in the autopilot integration configuration. A value of 0.063m from the base of the aircraft has been calculated, however allowing for uncertainty in the calculation and using caution to avoid roll over a maximum possible value of 0.065m is used.

**Location of the main gear** is governed by two simple parameters. From statistical data the main landing gear should be location between approximately 50-55% of the MAC for all aircraft, and the angle between the vertical and line between the most aft CG to the main gear should be 15°, as shown in Figure D.1. This will ensure that the aircraft is able to pitch up during take off however the gear are placed aft enough to ensure that the aircraft will not pitch up due to perturbations on the ground surface.

Simple trigonometric calculations are used to determine the location of the main landing gear. A location of 50% to 55% of the MAC correlates to a range between 0.528m and 0.540m from the nose of the aircraft. As long as the gear is located in this range then it will meet the statistical data requirements. The height of the main gear is approximately 0.18m, as stated in Table 7.14, and the wheels have a diameter of 3.5 inches. This implies that the net distance from the bottom of the wheel to the base of the aircraft is 0.22m. Using trigonometric relationships to determine the location of the main gear, for an angle of 15 degrees aft of the CG the main gear is located at 0.539m from the nose of the aircraft. This value meets both the MAC location and 15° from CG requirements.
One final check must be performed in order to ensure that the aircraft tail will not foul the ground during ground roll rotation. Using the geometry of the fuselage and the aforementioned main gear location the angle between the tail of the aircraft and the main gear can be determined, (Figure D.2). This angle is \(21.8^\circ\) which is more than sufficient to meet the rotation requirements of the aircraft. The literature survey suggested the angle to be above 12 degrees for common aircraft to ensure that the tail does not foul during take off rotation, however as the stall angle of iSOAR is approximately 10 degrees any value greater than this will be sufficient.

![Figure D.2: Tail Angle Schematic (Currey, 1988)](image)

The above process has determined the location of the main landing gear. This results in the location of the main landing gear being approximately 0.539m from the nose of the aircraft which correlates to approximately 55% of the mean aerodynamic chord.

**Landing gear static loads** are required for comparison to estimate their relative loads and place the nose gear correctly. The following equations relate to Figure D.3 and are required to calculate the static landing gear loads for the aircraft. The results of this analysis can be seen in Table D.2.

![Figure D.3: Landing Gear Loads Schematic (Currey, 1988)](image)
Table D.2: Results of Load Analysis for Nose Gear Location at 0.1 m

<table>
<thead>
<tr>
<th>Load Type</th>
<th>Design Load (N)</th>
<th>Percentage of Total Load (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max static load per main leg</td>
<td>36.75 N</td>
<td>41.6 %</td>
</tr>
<tr>
<td>Max static nose load</td>
<td>16.3 N</td>
<td>18.5 %</td>
</tr>
<tr>
<td>Min static nose load</td>
<td>14.8 N</td>
<td>16.7 %</td>
</tr>
</tbody>
</table>

Nose landing gear is required to be placed at a position in which it satisfies a number of design requirements. The nose gear must be sufficiently forward of the CG to reduce the loads on the nose gear, minimise weight and avoid ground roll over. Conversely, the nose gear must be placed sufficiently aft to maintain loads for sufficient control authority during ground roll and to clear the propeller during operation. Thus the static loads on the nose gear are used as a design guide to the adequacy of the location. A range between 6% and 20% of the total aircraft weight, however this should be treated as an extreme, and values between 8% and 15% of the nominal aircraft static weight are a good guide as stated by Currey (1988). An analysis using a range of alternative locations indicated a position as far forward as possible was required to satisfy ground turnover angle guidelines. A position of 0.1m from the nose is a reasonable trade off between ground roll stability and propeller clearance requirements. This results in a turnover angle of 61 degrees, which can be determined from trigonometry as defined by Figure D.4. This is slightly high for use on rough ground however the system will be used only on the smoothest possible ground surface for all flights.

Figure D.4: Turnover Angle Definition (Roskam, 2004b)
Appendix E

Parachute Design Calculations

The parachute design consisted of the design of two main systems; the primary and pilot parachute. The preliminary sizing of the primary parachute was conducted using elementary steady aerodynamics. Following this an analysis of the expected shock and opening loads was made through methods presented in Ewing et al. (1978). The sizing of the pilot parachute was made during this process and the detailed results of this analysis is presented below.

Primary parachute

The primary parachute was decided to be purchased from an amateur rocketry parachute manufacturer. This ensured that the final parachute was viable, at least as a recovery mechanism, and in relatively high quality with a low lead time.

Consideration had to be made to factors other than aerodynamic performance during descent, such as system mass, system volume and opening and shock loadings.

A preliminary estimate of the drag coefficient for a parachute of this nature was deduced from Nakka (1999) to be approximately of the order 1.0. A carpet plot of potential drag coefficients and required descent rates was conducted to determine the required parachute diameter.

Descent speed  A comparison between descent speed and equivalent height of drop, using an energy analysis, was conducted to give a physical interpretation of the descent speed, (Figure E.1). This turns the descent speed of the vehicle, during descent, into an equivalent height from which the vehicle may be dropped to represent the impact. A lower descent speed is preferred, however an equivalent drop height of above 2m, which correlates to a speed of 6.25m/s, is considered unacceptable, and a height of approximately 1.5m, which correlates to a speed of 5.5m/s, would be substantial.
Drag analysis  A free body diagram of the UAV and parachute recovery system is presented in Figure E.2.

\[
\sum F_x = 0 \Rightarrow D = mg
\]

As \( D = C_D \frac{1}{2} \rho V^2 S \)

\[
S = \frac{mg}{C_D \frac{1}{2} \rho V^2 S} \quad (E.1)
\]

From this analysis a carpet plot could be developed, the results of which are presented in Table E.1.
Table E.1: Carpet Plot for Preliminary Diameter of the Primary Parachute (ft)

<table>
<thead>
<tr>
<th>Descent Speed</th>
<th>CD = 0.8</th>
<th>CD = 0.9</th>
<th>CD = 1</th>
<th>CD = 1.1</th>
<th>CD = 1.2</th>
</tr>
</thead>
<tbody>
<tr>
<td>4.0</td>
<td>12.42</td>
<td>11.71</td>
<td>11.10</td>
<td>10.59</td>
<td>10.14</td>
</tr>
<tr>
<td>4.2</td>
<td>11.82</td>
<td>11.15</td>
<td>10.58</td>
<td>10.08</td>
<td>9.65</td>
</tr>
<tr>
<td>4.4</td>
<td>11.29</td>
<td>10.64</td>
<td>10.09</td>
<td>9.63</td>
<td>9.22</td>
</tr>
<tr>
<td>4.6</td>
<td>10.80</td>
<td>10.18</td>
<td>9.66</td>
<td>9.21</td>
<td>8.81</td>
</tr>
<tr>
<td>4.8</td>
<td>10.35</td>
<td>9.75</td>
<td>9.25</td>
<td>8.82</td>
<td>8.45</td>
</tr>
<tr>
<td>5.0</td>
<td>9.93</td>
<td>9.36</td>
<td>8.88</td>
<td>8.47</td>
<td>8.11</td>
</tr>
<tr>
<td>5.2</td>
<td>9.55</td>
<td>9.00</td>
<td>8.54</td>
<td>8.14</td>
<td>7.80</td>
</tr>
<tr>
<td>5.4</td>
<td>9.20</td>
<td>8.67</td>
<td>8.23</td>
<td>7.84</td>
<td>7.51</td>
</tr>
<tr>
<td>5.6</td>
<td>8.87</td>
<td>8.36</td>
<td>7.93</td>
<td>7.56</td>
<td>7.24</td>
</tr>
<tr>
<td>5.8</td>
<td>8.56</td>
<td>8.07</td>
<td>7.66</td>
<td>7.30</td>
<td>6.99</td>
</tr>
<tr>
<td>6</td>
<td>8.28</td>
<td>7.80</td>
<td>7.40</td>
<td>7.06</td>
<td>6.76</td>
</tr>
</tbody>
</table>

After considering the analysis presented in Table E.1, and conducting a market survey the Top Flight Recovery 8 ft diameter parachute was selected as the primary parachute for this system. Top flight recovery specialises in the fabrication of parachutes for amateur rocket systems and is the manufacturer of choice for many large scale rocket projects. This choice was made as top flight only manufacture parachutes at either 8 ft or 10 ft diameters. The 10 ft diameter parachute was discounted for the reason that the system would be overly cumbersome, take up too much volume, require a large pilot parachute and result in too high opening loads during deployment.

The 8ft diameter parachute would result in a descent speed of 5.5 m/s, for a drag coefficient of 1.0. This correlated to an equivalent drop height of 1.5m which is considered substantial yet acceptable. This primary parachute weighs approximately 0.517kg. Testing on the applicability of this design will be made through ground testing of this system.

**Pilot Parachute**

The pilot parachute is designed to ensure that the primary parachute sets in a controlled, reliable and safe manner. Thus the pilot parachute must be designed with sufficient size such that it forcefully deploys the main parachute across a variety of operational velocities, yet small enough to meet the design requirements for volume and mass. Due to the strict internal volume limitations this was considered a major driver in the selection of a suitable size. The pilot parachute must also be designed such that it deploys in a reliable fashion by catching the wind for a wide range of operational conditions.

A complicated analysis of parachute deployment at a range of speeds was conducted and a pilot parachute was sized for preliminary estimates. The analysis that was conducted was for a preliminary understanding of how the system would react and predict the order of magnitude of the expected opening and shock loading. This analysis resulted in the preliminary selection of a diameter of 1 ft which should be sufficient to deploy the primary parachute at low speeds of the
order of 50 km/h. The system would be testing using car based testing procedures to ensure the systems validity.

**Parachute System Analysis**

The parachute system analysis is conducted in two phases; estimation of the snatch force and opening force. The methods used in the analysis below are presented in Ewing et al. (1978).

**Snatch Force Calculations**  The snatch force is the impulse force on the parachute mass as it is deployed. The snatch force can be calculated using the following equation.

\[ F_s = F_i + D - W_p \sin \theta \]  \hspace{1cm} (E.2)

As the weight of the payload, the aircraft, is not in line with the drag, or inertia forces this can be neglected.

\[ F_s = F_i + D \]  \hspace{1cm} (E.3)

The inertia load requires an accurate estimation of the maximum differential velocity parameter, \( \Delta v_{\text{max}} \), and an estimate of the equivalent spring constant of the rise-suspension line system.

\[ F_i = \Delta v_{\text{max}} (mk)^{\frac{1}{2}} \]  \hspace{1cm} (E.4)

where \( m = m_c + \frac{(m_l + m_r)}{2} \approx 0.6085 \text{kg} \)

Assuming the body acceleration parameter, \( K_b \), is 0, as the acceleration of the aircraft after deployment is negligible, the maximum differential velocity can be determined using body canopy separation velocity shown in Figure E.3, and a parachute similarity parameter as defined by Equation E.5.

\[ \frac{\rho (C_D S)_{\text{pack}}}{2m_p} \]  \hspace{1cm} (E.5)

This similarity parameter was determined to be 0.0108 using preliminary design parameters.
From Figure E.3:

\[ \frac{\Delta v_{\text{max}}}{v_0} = 0.14 \]

This results in a maximum differential velocity, \( \Delta v_{\text{max}} \), of approximately 4.662 m/s for the maximum design velocity case of 120 km/h. From this the inertial load can be estimated. The spring constant is assumed to be 3 kN/m for a riser with bungee chord in series. Thus the inertial force can be determined from Equation E.4 and has been estimated at approximately 2.25g. This is representative of magnitude of the shock loading experiences during deployment.

**Opening force calculations** The accurate calculation of opening forces is difficult for parachutes, as the system kinematics change during deployment and opening. An empirical method has been developed for estimation of these loads from a statistical database of tested parachute designs. The method presented details the determination of an opening load factor from empirical graphs presented in Ewing et al. (1978).

Parachute mass ratio is a dimensional quantity used for non-dimensional analysis on parachute systems. It is defined by:

\[ R = \frac{3m_p}{\rho 4\pi r_p^2} \]
This quantity has been calculated using imperial in order to use this in further analysis. For this particular parachute with a parachute mass of approximately 1.14lbs and a parachute radius of 4 ft which results in a mass ratio of 0.156.

The opening load factor, $C_X$, is a dimensionless quantity that relates the opening load to the theoretical steady state load on a parachute.

$$C_X = \frac{F_x}{F_c}$$

Using Figure E.4 the opening load factor was determined, for a disreef parachute, with a mass ratio of 0.156., to be approximately 0.9. This value was used to estimate the opening force with respect to velocity, using the graph in Figure E.5.

Figure E.4: Opening Load Factor Vs Mass Ratio (Ewing et al., 1978)

$$F_X = C_X C_{Dq} S$$
For the maximum design speed of 120 km/h the opening load is expected to be approximately 9 g, and during deployment at cruise conditions a load of approximately 5 g is expected. This upper value is pushing the limitations of the structure; however it is unlikely that the parachute will be deployed at velocities of this magnitude. Normal operation of the parachute will ensure that the opening loads generated should lie well within the structural ability of this aircraft, through implementation of a suitable parachute deployment procedure.
Appendix F

Example Flight File

The following is a flight file that was coded for a 1000 ft square loitering pattern, which occurs under RC throttle maximum control. Three examples of user programmed interface buttons are included, along with the Flight Termination automatic failure pattern for loss of communication with the ground station.

//Start Header
imperial //Use Imperial units
[thOverride]=2 //throttle max set by RC
takeoff //code beyond this point will run when the autopilot senses a take-off
//End header

//Begin Body
flyTo (1000, 1000) //Waypoint Navigation, in feet, with reference
flyTo (1000,-1000) //to the position at which the autopilot was initialised
flyTo (-1000,-1000)
flyTo (-1000, 1000)
repeat -5 //autopilot will repeat previous 4 lines of code
//End body

//Begin Footer
fixed // Including the fixed command will allow waypoints to reprogrammed in flight
// define an Immediate abort mission pattern - release parachute
definePattern 0 //Define payload which will appear on GS screen on //left bar
[fServo8]= -32000 //Deploys Parachute (on servo 8)
[stopEngine]=1 //Turn off the motor
//[fServo2]= -10000 //servo 2 are the elevators
wait 99999 //wait forever

// define a finished mission - returns to home position and deploys parachute
definePattern 1
flyTo [home] //Fly back to original position
[fServo8]= 30000 //Deploy parachute (on servo 8)
[stopEngine]=1 //Turn off the motor
wait 99999 //wait forever

// define a figure-eight holding pattern
definePattern 2
[rotatePattern]= [currentHeading] //Sets a figure 8 pattern based on
flyTo (100,100) //current heading
flyTo (0,0)
flyTo (-100,-100)
flyTo (100,-100)
flyTo (0,0)
flyTo (-100,100)
repeat -6 //repeat figure 8 until pattern ended (payload //deactivated)

// define the loss of comms pattern
definePattern gcsFailed //Define in-flight failure pattern for loss of communication
[fServo8] = -32000 //Release Parachute
[51]= -450 //Set max elevator position to 5 degrees
flare //Move to maximum elevator position
[stopEngine]=1 //Stop Throttle
setcontrol RdFixed //Set rudder to zero
setcontrol rollFixed //Set Ailerons to zero
repeat -1 //Do nothing forever
//End footer
Appendix G

PID Loop Operation

Table G.1 details the operation of the twelve PID loops which are used to control the aircraft. Of these, the first six are capable of directly affecting the control surfaces, whilst the remaining six are cascaded loops which determine an intermediate field value.

<table>
<thead>
<tr>
<th>Name</th>
<th>Controls</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>aileron from roll</td>
<td>aileron</td>
<td>Controls the ailerons to minimize the difference between desired roll and actual roll.</td>
</tr>
<tr>
<td>elevator from pitch</td>
<td>elevator</td>
<td>Controls the elevator to minimize the difference between desired pitch and actual pitch.</td>
</tr>
<tr>
<td>rudder from Y accelerometer</td>
<td>rudder</td>
<td>Controls the rudder to minimize the difference between the desired value for Y accelerometer and the actual value. This is the feedback loop that coordinates turns.</td>
</tr>
<tr>
<td>rudder from heading</td>
<td>rudder</td>
<td>Controls the rudder to minimize the difference between desired heading and actual heading. This feedback loop is used during takeoff to keep the aircraft on the correct heading.</td>
</tr>
<tr>
<td>throttle from speed</td>
<td>throttle</td>
<td>Controls the throttle to minimize the difference between desired speed and actual speed. This feedback loop is used during final approach and when the option to control speed via throttle and altitude via elevator is selected.</td>
</tr>
<tr>
<td>throttle from altitude</td>
<td>throttle</td>
<td>Controls the throttle to minimize the difference between desired altitude and actual altitude. This feedback loop is used when the option to control altitude via throttle and speed via elevator is selected.</td>
</tr>
<tr>
<td>Name</td>
<td>Controls</td>
<td>Description</td>
</tr>
<tr>
<td>---------------</td>
<td>-----------</td>
<td>---------------------------------------------------------------------------------------------------------------------------------------------</td>
</tr>
<tr>
<td>pitch from altitude</td>
<td>desired pitch</td>
<td>Controls desired pitch to minimize the difference between desired altitude and actual altitude.</td>
</tr>
<tr>
<td>pitch from AGL</td>
<td>desired pitch</td>
<td>Controls desired pitch to minimise the difference between desired altitude and altitude as measured by the AGL board. This feedback loop is enabled during landing and controls the flare.</td>
</tr>
<tr>
<td>pitch from airspeed</td>
<td>desired pitch</td>
<td>Controls the desired pitch to minimize the difference between desired airspeed and actual airspeed. This feedback loop is enabled during climb and during level flight when the option to control altitude via throttle is selected.</td>
</tr>
<tr>
<td>roll from heading</td>
<td>desired roll</td>
<td>Controls the desired angle of bank to minimize the difference between the desired heading and the actual heading. This feedback loop is enabled any time your MicroPilot Autopilot is navigating.</td>
</tr>
<tr>
<td>heading from crosstrack error</td>
<td>desired heading</td>
<td>Controls the desired heading to minimize the distance between your MicroPilot Autopilot and the line defined by the previous waypoint and the next waypoint. This feedback loop is enabled when the fromTo command is being run.</td>
</tr>
<tr>
<td>pitch from descent</td>
<td>desired pitch</td>
<td>Controls desired pitch to minimise the difference between desired descent rate and actual descent rate.</td>
</tr>
</tbody>
</table>
Appendix H

Adjusting Gains

This section details the procedures used to adjust all PID loop gains in order to provide for autonomous flight. The gains are to be set in the order shown in this report. Several flights may be necessary for some control loops.

The RC flight path is flown as described, including some switching to autopilot control using the aux II control on the RC transmitter. Observations are then used to tune the individual PID loops. The RC pilot has ultimate control at all times, and reserves the right to switch from autopilot to RC flight if instabilities are observed or if it is deemed risky to continue with a certain manoeuvre.

H.1 General Approach

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce by 25%</th>
</tr>
</thead>
<tbody>
<tr>
<td>Oscillation is Slow (&lt;1/sec)</td>
<td>I term</td>
</tr>
<tr>
<td>Oscillation is Fast (&gt;1/sec)</td>
<td>P, D or both P and D</td>
</tr>
</tbody>
</table>

Notes

- Test gains at high and low bounds of airspeed - at high speeds the gains may need reduction.

- If gains cannot be found that are stable over the entire airspeed range, gain scheduling can be used (ie. Set different gains for different parts of the airspeed range).

- Once gains have been reduced such that they are stable, gains that have not been changed should be increased until a small instability is observed, and then reduced 25%.
H.2 Adjusting Elevator, Ailerons and Rudder Gains for Level Flight

RC Flight

Fly rectangular circuits at a 'safe’ altitude - one that would allow time for a switch back to RC control if an AP error occurred (as determined by the RC pilot). Flick to CIC (Computer In Control) on one of the longer legs. Flick back into RC control at the end of the leg and fly round for another run (As proficiency is increased it may be possible to increase this to CIC for both legs, but not initially).

Flick back to RC control anytime at the pilots discretion if it is believed the plane is becoming unstable.

The throttle will remain as RC controlled for the entire flight.

<table>
<thead>
<tr>
<th>Code Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Straight-level-flight with the throttle still controlled by the RC receiver</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Actions</th>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>The nose pitches up and down</td>
<td>Elevator from pitch</td>
</tr>
<tr>
<td></td>
<td>Wings rock from side to side</td>
<td>Aileron from roll</td>
</tr>
<tr>
<td></td>
<td>Nose yaws from side to side</td>
<td>Rudder from yaw</td>
</tr>
</tbody>
</table>

H.3 Adjusting Pitch from Airspeed and Roll from Heading

RC Flight

Fly rectangular circuits at a 'safe’ height (as determined by RC pilot) with the longer legs in the same direction as the take-off. At the beginning of the straight leg coinciding with the direction of takeoff, switch to CIC mode. The plane will attempt to climb, so set the throttle to a suitable value to ensure climb is possible. Switch back to RC control at the end of the climb and bring round for another run.

The throttle will remain as RC controlled for the entire flight.

<table>
<thead>
<tr>
<th>Code Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Climb 'forever’, with the RC controller controlling throttle, and the elevator controlling airspeed - set at a specific value in Horizon.</td>
</tr>
</tbody>
</table>
H.4 ADJUSTING PITCH FROM ALTITUDE

Actions

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td>The plane’s airspeed oscillates</td>
<td>Pitch from airspeed</td>
</tr>
<tr>
<td>The plane performs S-turns</td>
<td>Roll from heading</td>
</tr>
</tbody>
</table>

H.4 Adjusting Pitch from Altitude

RC Flight

Takeoff and fly to a height of 200ft (note this can be altered if desired). Switch to CIC control when near the point of takeoff, and facing in the same direction as takeoff. Leave on CIC control until gains have been adjusted, unless the plane becomes unstable.

The throttle will remain as RC controlled for the entire flight.

Code Function

Fly round a series of waypoints, with the elevator holding a specified altitude

Actions

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td>The plane oscillates in pitch</td>
<td>Pitch from altitude</td>
</tr>
</tbody>
</table>

H.5 Adjusting Throttle from Altitude

RC Flight

Takeoff and climb to 200ft (note this can be altered if desired). Switch to CIC control when near the point of takeoff, and facing in the same direction as takeoff. Leave on CIC control until gains have been adjusted, unless the plane becomes unstable.

Code Function

Fly round a series of waypoints, with the throttle holding a specified altitude

Actions

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td>The plane’s oscillates in pitch</td>
<td>Throttle from altitude</td>
</tr>
</tbody>
</table>
H.6 Adjusting Throttle from Airspeed

RC Flight

Takeoff and climb to 200ft. Switch to CIC control when near the point of takeoff, and facing in the same direction as takeoff. Leave on CIC control until gains have been adjusted, unless the plane becomes unstable.

Code Function

Fly round a series of waypoints, with the throttle holding the specified cruise airspeed. The autopilot will also cause the UAV to climb to 300ft during this process.

Actions

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td>The planes airspeed oscillates</td>
<td>Throttle from airspeed</td>
</tr>
</tbody>
</table>

H.7 Adjusting Rudder from Heading

RC Flight

Takeoff and climb to 100ft. Switch to CIC control when near the point of takeoff, and facing in the same direction as takeoff. Leave on CIC control until gains have been adjusted, unless the plane becomes unstable.

Note that the throttle maximum is set on the RC transmitter for the entire flight.

Code Function

Fly back and forth between two waypoints, with the heading determined predominantly by the rudder. The autopilot will also attempt to climb to 300ft during this process.

Actions

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td>The planes oscillates</td>
<td>Rudder from Heading</td>
</tr>
</tbody>
</table>
H.8 Adjusting Heading from Crosstrack

RC Flight

Takeoff and climb to 100ft. Switch to CIC control when near the point of takeoff, and facing in the opposite direction as takeoff. Leave on CIC control until gains have been adjusted, unless the plane becomes unstable.

The throttle will remain as RC controlled for the entire flight.

Code Function

Fly round a figure of eight pattern using the fromTo command on the diagonals. This command causes the plane to fly as close to the line that joins the waypoints as possible.

Actions

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td>The plane flies a series of S-turns</td>
<td>Heading from crosstrack</td>
</tr>
</tbody>
</table>

H.9 Adjusting Pitch from Descent

RC Flight

Takeoff and climb to 250ft. Switch to CIC control when at the beginning of a new run. Leave on CIC control until plane levels out at 100ft, nears the edge of RC range, or appears to become unstable. Repeat process as necessary. Note that when in autopilot control, the throttle will be controlled via the autopilot.

Code Function

The code will cause the plane to descend at a stable rate and level off at 100ft heading in a straight line at all times.

Actions

<table>
<thead>
<tr>
<th>If</th>
<th>Then reduce gains in . . . loop</th>
</tr>
</thead>
<tbody>
<tr>
<td>The plane doesn’t descend at a stable rate or level off</td>
<td>Pitch from descent</td>
</tr>
</tbody>
</table>
Appendix I

Parachute Activation at Mission Boundary

The ARCAA Outback Challenge (ARCAA, 2007) required a calculation of the possible travel distance beyond the specified mission boundary. The travel distance after parachute actuation has been calculated for an absolute ‘worst-case’ scenario to determine the likely distance traveled beyond the boundary. These calculations could be easily modified to suit a different application, and can be used to consider the effect of extending the time between an error occurrence and flight termination.

Initial Conditions

- Cruise Speed: 90km/hr (relative to ground - i.e. not airspeed)
- Altitude: 150m
- Direction of flight: Perpendicular to mission boundary
- Wind: Tail-wind @ 20km/hr

Parachute Details

- Type: Standard Configuration with drogue
- Ejection Method: Drogue release via servo actuation pulls out main parachute at a trailing distance of 1.5m
- Setting Time: 2s
- Drag Coeff (as tested): $C_D = 0.94$
- Area: $A = (2.4\text{diam}) = 4.52\text{m}^2$

Scenario
Parachute Activation  
Time  
t = 0s

Parachute Set*  
Time  
t = 2s  
Distance Traveled  
s = \text{setting} \times v_{\text{cruise}} = 2/3600 \times 90 = 50m

*assume forward velocity = cruise velocity. This discounts elevator effect and reduced speed due to motor shutdown

Vehicle Land

\[ D = C_D q S \]

where \( q = \frac{1}{2} \rho v^2 \)

therefore

\[ v = \left( \frac{2D}{(\rho C_D S)} \right)^{0.5} \]

where:
- Drag \( D = (8.2 \text{kg}) = 80.442\text{N} \)
- Area \( A = (2.4 \text{diam}) = 4.52\text{m}^2 \)
- Drag Coeff (as tested) \( C_d = 0.93 \)
- Density of air \( \rho = 1.2\text{kg/m}^2 \)

Therefore \( V = 5.65\text{m/s} \)

Time of flight = 150/5.65 = 26.5s

Therefore possible drift = 147.2m

Total Distance beyond Boundary  
Total possible deviation outside boundary = setting distance + drift distance

= 50 + 147.2

= 197.2m
Appendix J

Flight Test Procedures

This appendix details the flight test procedures used throughout the flight test phase of the project. Each test consisted of a pre-flight, flight test and post-flight phase.

J.1 Pre-Flight Preparation

Description: The day prior to every flight test the following procedure must be completed to ensure that the flight test runs smoothly. This procedure details all requirements of the aircraft prior to the flight test such that the flight test may run as planned with a minimum of time and effort, reducing the inconvenience on third parties. This task may be performed by a number of group members however the responsibility to have this completed must lie with a single group member to ensure that all of the following requirements are met.

Requirements

1. Charge all battery power supplies. This includes, main battery power supplies, autopilot, servo, camera and RC transmitter battery supplies.

2. Ensure all cells of each battery are balanced immediately before use.

3. Once charged the batteries may be tested in the airframe to ensure correct operation of the system power supply.

4. Prior to any flight test, a dry run of the flight test is to be completed. (i.e. for the autopilot test the autopilot is to be installed and the ground station is to be set up to ensure correct operation) The flight test will be cancelled if a dry test run is not successful.

5. The airframe is to be disassembled and packed safely, with all components and fixtures included.

6. The ground station apparatus is to be dismantled and packed safely into a vehicle.
J.2 Start Up Procedure

Description: Prior to every flight test the following procedure must be implemented to ensure a safe and controlled flight test. The aim of this procedure is to prepare the aircraft for flight and ensure that all the aircraft is in sound working order prior to testing. This procedure must be completed immediately prior to any flight testing procedure period, and must be repeated after a change to any hardware, including power plant batteries.

Procedure:

1. Ensure that wing and airframe are fastened securely
2. Check structural integrity of airframe and airframe components, including control surfaces and propeller.
3. Turn on the receiver
4. Remove the top hatch cover and false floor of vehicle. Install required power supply for system electronics.
5. Switch on all electronics equipment.
6. Perform brief, ground RC test, to ensure all servos are operating correctly. Particular attention to paid to direction of control surface movements.
7. Replace the false floor to cover the electronics bay.
8. Fold and install the parachute recovery system and prime the system according to the recovery system folding procedure (yet to be finalised). Ensure that the top hatch is securely fixed with little or no play.
9. Ensure no signal (autonomous or RC) is being sent to throttle channel. Turn the aircraft over and remove the bottom hatch. Install and switch “on” power plant batteries into airframe, ensure that they are securely fastened, and replace bottom hatch. **CAUTION: Propeller may begin turning. Keep all persons and property clear of harm.**
10. Position aircraft in launch carriage, and securely fasten. In case of rolling takeoff and landing and chock landing gear to avoid movement during ESC initialisation.
11. Using the RC Transmitter, initialise the electronic speed controller, by running engine to full throttle. **CAUTION: Ensure that all persons and property are clear of harm.**
12. Conduct a range test to ensure that the RC Receiver maintains constant communications. Walk the RC Transmitter approximately 200m from the aircraft and actuate all controls, including the throttle, to verify the controller.
NOTE: This is sufficient for RC Test Flights, however it will need to be amended for Autopilot test flights.

### J.3 Proof of Aerodynamic and Mechanical Design Tests

The following test procedures relate to the proof of aerodynamic and mechanical design tests. This consists of ground roll tests, touch n go tests and the first flight test.

#### J.3.1 Ground Roll Procedure

Description: Prior to actual flight tests the aircraft will be tested via ground roll only to ensure the working order of all functions and ensure the power of the engine is sufficient to power the aircraft. This test is completed by running the aircraft along the take off strip and should be conducted in a number of independent runs. The aim of this test is to allow the pilot to obtain an understanding of the power characteristics of the vehicle and gain confidence with the aircraft.

Procedure:

1. Perform the “Start Up Procedure”
2. Position the aircraft on the runway, facing the direction of the wind, with a path that is straight down the length of the runway.
3. Actuate the throttle to a setting that results in a low acceleration of the aircraft. Maintain a comfortable speed for a sufficient time so as to ensure the performance of the power plant at this speed. Ease of the throttle and wait for the aircraft to come to a stop.
4. Repeat the test a number of times, each time increasing the power and acceleration of the aircraft until the pilot is confident in the power characteristics of the aircraft, including under full throttle.

#### J.3.2 Touch n Go Test

Description: The first flight of the aircraft will be maintained for a very short period of time. A test known as “touch and go” will be performed to ensure the lifting capabilities of the aircraft. This test will prove the aircraft under a full take off procedure and will also extend partially into the flight domain. This test may be repeated a number of times until the pilot is confident of the aircrafts capabilities under takeoff conditions.

Procedure:

1. Perform the “Start Up Procedure”
2. Position the aircraft on the runway, facing the direction of the wind, with a path that is straight down the length of the runway.

3. Actuate the throttle to a position where the aircraft will accelerate strongly. This may require full application of the throttle.

4. Once the aircraft has gained sufficient speed pull back on the elevator slowly until the aircraft begins to rotate around the main landing gear. Ease off the throttle and let the aircraft come to a rest.

5. Repeat the test however once the aircraft has gained sufficient speed actuate the elevator until the aircraft climbs to a small height of the ground, (3 to 5 feet,) and then immediately land vehicle. Repeat this test until the pilot becomes fully confident with the aircraft under full take off conditions.

J.3.3 First Flight Test

Description: The aim of this test is to prove the flying capability of the aircraft. This flight will consist of a short flight and will prove the aircraft under take off, climb, loiter and landing. The duration of this flight is expected to be of the order of two to five minutes with the aim in mind to prove flight only. The aircraft will be flown at a moderate speed with straight level flight being the goal. Landing may consist of a baulked landing prior to a landing attempt, as this regime of flight involves the majority of the risk. This test must be timed and detailed comments made as to the performance of the vehicle.

Procedure:

1. Perform the “Start Up Procedure”

2. Position the aircraft on the runway, facing the direction of the wind, with a path that is straight down the length of the runway.

3. Apply take off power and accelerate to take off velocity

4. Pull back on the elevator to rotate the aircraft to a climb regime. Maintain the climb until an altitude of approximately 100-200ft.

5. Level out and reduce the power to a moderate level to maintain level flight at a comfortable speed.

6. Complete an oval shaped circuit.

7. Attempt a baulked or false landing by creating an imaginary runway at an altitude of approximately 50ft above that of the runway and practice landing. Accelerate the aircraft back up to speed and altitude and complete another circuit.

8. Repeat step 7 until confident of making a successful landing.

9. Land the aircraft on the runway and reduce the throttle until the aircraft comes to rest.
J.4 Camera Integration Flight Test

Description: The day prior to every flight test the following procedure must be completed to ensure that the camera-flight test runs smoothly. This procedure details all requirements of the camera prior to the flight test such that the flight test may run as planned with a minimum of time and effort, reducing the inconvenience on third parties. This task may be performed by a number of group members however the responsibility to have this completed must lie with a single group member to ensure that all of the following requirements are met.

Procedures:

On the UAV

1. Ensure the camera is securely placed in the plane
2. Ensure the transmitter along with the antenna is placed securely in the UAV.
3. Ensure the battery is securely placed in the UAV.
4. Ensure the connections/wirings are securely connected.

At the ground station

1. Connect the stand securely with the hinges and screws.
2. Place the directional antenna on top of the stand and connect them with U brackets.
3. Connect the wire from the antenna to the receiver.
4. Erect the stand straight and firmly secure it to the trailer so that the whole component is stable.
5. Connect the wire to the receiver from the antenna.
6. Secure the receiver to the stand firmly with the help of chords.
7. Connect the USB/Video adapter from the receiver to the laptop.
8. Connect the Power adapter from the inverter to the receiver.
9. Connect the inverter to the car battery and ensure it is secured firmly to the car battery.
10. Turn the laptop on and connect it to the multi-plug connecting the inverter.
11. Turn on the inverter.

Power Connections

1. Turn on the power supply for the camera and the transmitter on the UAV.
2. Now check for communication if the image is being received from the transmitter on the plane to the laptop at the ground station.

3. Once communication is established, perform a range test by just walking about 200 meters and check for the signal.

4. Once the scenario seems to be desirable the UAV would be ready for take-off.

**During Flight**

1. Capture the video received from the UAV on the laptop.

2. Check for the communication errors or gaps and make sure the antenna is directional with respect to the flight.

3. Check for the clarity of the image and vibrational affects produced on the image.

4. Check for blurriness in the image.

**Post Flight**

1. Review the recorded image and check the clarity.

2. Record the range obtained from the flight.

3. Dismantle everything and place them securely back into the boxes they were originally in.

### J.5 Payload Deployment Test

**Description:** The aim of this test is to prove the deployable payload system of the aircraft. This flight will consist of a short flight to gain altitude. The duration of this flight is expected to be of the order of two to five minutes. Once the pilot is satisfied with the altitude and flying characteristics he will activate the payload via the RC control on the down wind leg, away from observers.

**Procedure:**

1. Perform the “Start Up Procedure”

2. Position the aircraft on the runway, facing the direction of the wind, with a path that is straight down the length of the runway. Ensure the payload is secure on the aircraft undercarriage.

3. Apply take off power and accelerate to take off velocity

4. Pull back on the elevator to rotate the aircraft to a climb regime. Maintain the climb until an altitude of approximately 100-200ft.
5. Level out and reduce the power to a moderate level to maintain level flight at a comfortable speed.

6. Complete an oval shaped circuits until adequate altitude is gained.

7. Activate the payload deployment switch when the aircraft is on the down wind leg.

8. Land the aircraft on the runway and reduce the throttle until the aircraft comes to rest.

**J.6 Autopilot Pre-flight Tests**

Description: Prior to every flight test that will include the autopilot the following procedure must be implemented in place of the normal start-up procedure described in Section of the to ensure a safe and controlled flight test. The aim of this procedure is to prepare the aircraft for flight and ensure that all the aircraft is in sound working order prior to testing. This procedure must be completed immediately prior to any flight testing procedure period, and must be repeated after a change to any hardware, including power plant batteries.

1. AP Off, Servos Off

2. Check Engine servo connected to Servo Board (S4). Connect Engine batteries and attach landing gear (and payload servo if applicable). At this point in time the ESC is connected to a ‘dead’ board. No control signals can be transmitted through the servo board until the AP receives GPS. Therefore there is no chance of attracting rouge RC signals at this stage.

3. Implement ’Checklist before power up’ - includes wings and all connections

4. Implement ’Powering up’ - initialises AP and communication, set parachute

5. Implement ’Autopilot Tests’ - checks that AP response is as expected

6. Autopilot is now ready to fly. Perform other R/C tests as necessary

**J.6.1 Checklist before power up**

1. Check AP battery connected to switch

2. Check AP switch connected to AP

3. Check Servo Battery to switch

4. Check Servo Switch to Servo Board

5. Connect Payload Servo (s7)

6. Attach Wings

7. Connect Aileron servos using 2-1 JR connector set
8. Connect pitot tube to inner pressure transducer

9. Check all connections to servo board are tight

**J.6.2 Powering Up**

1. Power-on servos (LHS of plane looking forward)

2. Turn on RC transmitter and put in PIC mode (Aux 1 points away from RC pilot). Ensure throttle is set to the off position. Note that the RC controls will not work until after the AP is on and GPS has connected

3. Power-on AP (RHS of plane looking forward) in location and direction of takeoff if this will be a navigated flight.

4. Wait for initialisation beep (about seven seconds after turn-on). When GPS connects (about one minute wait), the servos will become active and in RC control.

   (a) Load up required flight file (before GPS connects but after initialisation beep for the PID tuning files).

5. Press Connect to enable visualisation of aircraft on Horizon Ground Station.

6. Check that all RC channels work as expected. The four control surfaces should work as per normal, but there will be no RC parachute control, as this is now routed through the AP.

7. Flick to CIC (Aux 1 toward the RC pilot). For the first tuning flights, RC will retain control of the throttle at all times. Connect GPS antenna, Set Parachute, and close the hatch.

**J.6.3 Autopilot Tests**

1. Rotate the plane about its pitch, roll and yaw axis. Check that the elevator, ailerons and rudder move in the correct direction to compensate for these rotations. Check that when level, control surfaces are approximately level.

2. Check that the battery voltages are acceptable (AP>8V, Servo>5V).

3. Check that the airspeed indicator is connected. Press finger firmly on the pitot tube and verify that the airspeed value increases.

4. Vibration Test (when RC throttle override is implemented): Advance throttle to full on RC transmitter. Verify that the elevator and rudder do not move significantly OR Vibration Test (no RC throttle override) Switch to RC control and advance throttle to full and back to idle. Verify that the artificial horizon is stable during this check.
5. Check that the elevator, ailerons and rudder are approximately in their neutral position in CIC mode when the plane is flat and level.

6. Check that wind speed does not exceed 50% of cruise speed.

J.7 Autopilot Flight Tests

All of the initial autopilot flight tests are tuning flights during which the control loops of the autopilot will be tuned to suit the iSOAR airframe and payload configuration. The procedures followed during these tests are detailed in Appendix H.
Appendix K

CAD Draft Drawings