AUTOPILOT DESIGN AND COMMERCIAL AUTOPILOT EVALUATION USING FLYBARLESS HELICOPTER

by

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Approval Signatures

We, the undersigned, approve the Master’s Thesis of Ahmad Alshoubaki.

Thesis Title: Autopilot Design and Commercial Autopilot Evaluation Using Flybarless Helicopter

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A special thank you goes to our pilot, John Mempin, for his hard work and professionalism in flying and maintaining the helicopter. I also want to thank Mr. Ricardo De Jesus in the AUS manufacturing lab for all of his efforts and cooperation to build the helicopter test stand and skid.

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Dedication

It is an honor to dedicate this work to my lovely Mother and my sweet family.
Abstract

In its effort to develop unmanned autonomous systems research capabilities, the College of Engineering is adding a rotary wing UAV to its research activities. The current thesis goal is to use the Maxi Joker 3 commercially-off-the-shelf (COTS) electric RC helicopter as a platform and fit it with a commercial autopilot system to serve as a benchmark for future AUS rotary wing in-house autopilot development. To achieve this goal, the thesis develops the helicopter flight simulator with added hardware-in-the-loop simulation capabilities to aid the rapid prototyping of flight control laws and the guidance algorithms. Rigorous flight dynamics simulation model was implemented with Matlab/Simulink environment. Hardware-in-the-loop simulation was carried out using the freescale MPC555 32 bits microcontroller based autopilot hardware developed in house. Flight tests data was used to refine the dynamics models and improve the simulation. The thesis developed autopilot for aircraft attitude control for hover flight conditions. The control lows are based on PID successive loop closure architecture using the linearized helicopter model. The tuned gains were simulated using the nonlinear model. The in house autopilot hardware-in-the-loop simulation showed promising results compared with flight test data collected with the Micropilot commercial autopilot test results. Later, the in house autopilot was fitted to the MaxiJocker 3 aircraft for flight test evaluation. Limited flight test data showed excellent results compared with the Micropilot commercial autopilot test results.

Search Terms: Rotary wing UAV, Hover model, 6-dof flybarless helicopter model, commercially available autopilot.
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**Latin Variables**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>a</td>
<td>Main rotor lift curve slope</td>
<td>5.5 [rad⁻¹]</td>
</tr>
<tr>
<td>aₜₐᵣ</td>
<td>Tail rotor lift curve slope</td>
<td>5.5 [rad⁻¹]</td>
</tr>
<tr>
<td>C</td>
<td>Rotor blade chord</td>
<td>[m]</td>
</tr>
<tr>
<td>CCPM</td>
<td>Cyclic Collective Pitch Mixing</td>
<td></td>
</tr>
<tr>
<td>CG</td>
<td>Center of Gravity</td>
<td></td>
</tr>
<tr>
<td>COTS</td>
<td>Commercial-Off-The-Shelf</td>
<td></td>
</tr>
<tr>
<td><em>C</em></td>
<td>Coefficient of thrust</td>
<td>[•]</td>
</tr>
<tr>
<td>Fus</td>
<td>Fuselage</td>
<td></td>
</tr>
<tr>
<td>fₓ, fᵧ, fᶻ</td>
<td>Helicopter forces in the X, Y and Z directions, respectively</td>
<td>[N]</td>
</tr>
<tr>
<td>G</td>
<td>Gravitational acceleration</td>
<td>9.81 [m/s²]</td>
</tr>
<tr>
<td>Hₜᵣ</td>
<td>Vertical length from tail rotor hub to helicopter CG</td>
<td>0.0779 [m]</td>
</tr>
<tr>
<td>Iₓₓ, Iᵧᵧ, Iᵣᵣ</td>
<td>Helicopter Moments of Inertia, respectively</td>
<td>[0.1 0.4 0.3] [kg m²]</td>
</tr>
<tr>
<td>L, M, N</td>
<td>Helicopter Roll, Pitch, and Yaw moments, respectively</td>
<td>[N m]</td>
</tr>
<tr>
<td>Lᵣᵣ, Mₐ</td>
<td>Roll Moment and Pitch Moment Derivative, respectively</td>
<td>[0 0] [Nm/rad]</td>
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<tr>
<td>Lᵣᵣ</td>
<td>Horizontal length from helicopter CG to tail rotor hub</td>
<td>1.068 [m]</td>
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<tr>
<td>M</td>
<td>Mass of helicopter</td>
<td>7.3 [kg]</td>
</tr>
<tr>
<td>N</td>
<td>Number of blades</td>
<td>2</td>
</tr>
<tr>
<td>p, q, r</td>
<td>Helicopter body roll, pitch, and yaw rates, respectively</td>
<td>[rad/sec]</td>
</tr>
<tr>
<td>Rₘᵣ</td>
<td>Main Rotor radius</td>
<td>0.89 [m]</td>
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</tbody>
</table>
\( R_{tr} \) : Tail Rotor radius \hspace{2cm} 0.16 [m]

\( S \) : Rotor Solidity \( Nc/(\pi R) \) \hspace{2cm} [\star]

\( T_{mr} \) : Main rotor thrust \hspace{2cm} [N]

\( u_a, v_a, w_a \) : Vehicle airspeed in the x, y, and z directions, respectively \hspace{2cm} [m/s]

\( u, v, w \) : Helicopter airspeed in the x, y, and z directions, respectively \hspace{2cm} [m/s]

\( u_{col} \) : Collective lever input \hspace{2cm} [rad]

\( u_{lat} \) : Lateral stick input \hspace{2cm} [rad]

\( u_{lon} \) : Longitudinal stick input \hspace{2cm} [rad]

\( u_{ped} \) : Rudder pedal input \hspace{2cm} [rad]

\( u_{tr}, v_{tr}, w_{tr} \) : Tail rotor hub airspeed in the x, y, and z directions, respectively \hspace{2cm} [m/s]

\( V, V_\infty \) : Helicopter airspeed (magnitude) \hspace{2cm} [m/s]

\( V_i \) : Normal component of rotor induced velocity \hspace{2cm} [m/s]

**Greek Variables**

\( \alpha \) : Angle of attack. \hspace{2cm} [rad]

\( \beta_{lc}, \beta_{ls} \) : Longitudinal and Lateral Main Rotor Flap Angles, respectively \hspace{2cm} [rad]

\( \beta^c_{lc}, \beta^c_{ls} \) : Commanded Longitudinal and Lateral Main Rotor Flap Angles, respectively. \hspace{2cm} [rad]

\( \delta_{ped} \) : Collective pitch-angle of tail rotor blade \hspace{2cm} [rad]

\( \theta_o \) : Collective pitch-angle of main rotor blade \hspace{2cm} [rad]

\( \lambda_{mr} \) : Inflow ratio for main rotor \hspace{2cm} [\star]

\( \mu \) : Advance ratio. \hspace{2cm} [\star]

\( \mu_z \) : Advance ratio along the z axis \hspace{2cm} [\star]
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
<th>Value</th>
</tr>
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<tr>
<td>$\rho$</td>
<td>Air density (1.275 kg/m$^3$)</td>
<td>[kg/m$^3$]</td>
</tr>
<tr>
<td>$\Omega_{mr}$</td>
<td>Main Rotor rotation rate</td>
<td>151.3 [rad/sec]</td>
</tr>
<tr>
<td>$\tau_{fc}$</td>
<td>Longitudinal Main Rotor Time Constant</td>
<td>[sec]</td>
</tr>
<tr>
<td>$\tau_{fs}$</td>
<td>Lateral Main Rotor Time Constant</td>
<td>[sec]</td>
</tr>
<tr>
<td>$\Omega_{tr}$</td>
<td>Tail Rotor rotation rate</td>
<td>680 [rad/sec]</td>
</tr>
</tbody>
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Chapter 1: Introduction

1.1 Background

Small scale Unmanned Aerial Vehicles (UAVs) are now being investigated for large area monitoring, pipeline monitoring, search and rescue, commercial aerial surveillance, traffic monitoring, fire detection, and remote sensing.

Small UAVs, including Vertical Takeoff and Landing (VTOL) vehicles and helicopters are used in both military and civilian applications. Research and development, integration, prototyping, and field testing are ongoing activities in most research institutions and labs around the world [1].

Helicopters have the ability of vertical takeoff and landing, hovering as well as cruising. This unique capability gives the helicopter the advantage over fixed wing vehicles to access and fly in congested and developed areas without needing a special landing area. On the other hand, helicopters are complex systems with strong non-linear dynamic and aerodynamic coupling between the various degrees of freedom. Helicopters on the other hand are highly agile and present more complexity in designing flight control laws and autopilot design [1].

1.2 Description of Basic Helicopter and Controls

This section describes the basic helicopter theory and corresponding vehicle subsystems and controls. The helicopter is controlled by four inputs: collective, longitudinal cyclic and lateral cyclic, and the rudder. Figure 1.1 shows the main RC helicopter components and controlling inputs.
The lifting force is produced by the main rotor by controlling the collective pitch of the blades, while maintaining the rotor at fixed speed. By tilting the main rotor through the swash plate, which is linked to 120° CCPM servos configuration, this will allow the helicopter to fly in the longitudinal or lateral direction corresponding to the tilt of the spinning rotor. Finally, the tail rotor is used to control the heading direction, and pull against the torque reaction created by the main rotor.

The Maxi Joker 3 RC helicopter was chosen as the platform to test the autopilot. It consists of a flybarless head which reduces the mechanical complexity with simplified flapping mechanism, as shown in Figure 1.2.

![Figure 1.2: Flybarless Helicopter Rotor Head [2]](image)

The 3-Axis Gyro acting as an electronic stabilizer is used in the flybarless helicopters. Figure 1.3 shows flybarless helicopters without 3-Axis Gyro where the helicopter’s response is too quick and difficult to control.

![Figure 1.3: RC Flybarless Helicopter without Auto-stabilizing Technology (3-Axis Gyro) [3]](image)

By using the 3-Axis Gyro, the body swings where the rotor part remains horizontal which gives more stable flight and control, as demonstrated in Figure 1.4.
Before using the 3-Axis Gyro, the flybar rotor, also called Bell-Hiller stabilizer bar, has commonly been used to stabilize small helicopters. It is a rotor containing a pair of paddles linked to the rotor shaft through a teetering hinge. It takes the same cyclic command input as the main rotor but its response is not as fast as the main rotor’s response and it is also affected by airspeed and wind gust. Figure 1.5 shows the RC helicopter rotor head which contains complicated mechanical flybar mechanism.

1.3 Rotary Wing Unmanned Aerial Vehicle Research:

There are numerous research projects on developing helicopter UAV for various purposes. This summary will focus on reviewing rotary wing UAVs in the mini category.

1.3.1 Universiti Teknologi Malaysia Helicopter [4]. The objective of this research activity was to develop an autopilot system that permits the helicopter model to carry out autonomous hover maneuver by using an on-board intelligent computer. The platform used was the Raptor Aircraft’s .90. This vehicle uses a stiff rotor head which has elastomers restrains to perform as unhinged teetering head, as shown in Figure 1.6. The flybar paddles are free to flap about the rotor head and they operate the same manner as a teetering rotor system with no hinge offset. The helicopter weighs 7.7 kg and is equipped with on-board computer which is a single Microchip
PIC18F4520 microcontroller. The ground station consists of PIC16F877A microcontroller, that performs as a mission controller controlling the movement of the UAV. The communication is implemented via a wireless data link from the LPRS EasyRadio ER400TRS module which works at 433-4MHz. The Polaroid 6500 sonar ranging module is used to measure the altitude.

![Figure 1.6: Raptor Aircraft’s .90 [4]](image)

The control development was based on the Lumped Parameter approach and is comprised of four different subsystems such as actuator dynamics, rotary wing dynamics, force and moment generation process and rigid body dynamics. The nonlinear dynamic model of the Remote Control (RC) helicopter was presented. The small perturbation theory was used to linearize the nonlinear helicopter mathematical model for stability analysis and linear feedback control system design. Using the Pole Placement method, the linear state feedback for the stabilization of the helicopter was derived. The flight control laws were done on-board, mission planning and human user interaction take place on ground. The proposed hovering controller was shown to be capable of stabilizing the helicopter attitude angles.

1.3.2 Yale University Helicopter [5]. The objective of this project is to maintain the hovering vehicles, either helicopter or quad rotor, stable and balanced in flight as payload mass is added to the vehicle, and to study the effects of the dynamic load disturbances under a PID controller.

A Heli-command Profi flight autopilot system that employs a PID attitude was fitted in the helicopter. The height and position drift can be controlled using optical feedback. Helicopter attitudes were measured by 3DM-GX3-25 inertial measurement unit (Microstrain, Vermont USA) and communicated via bluetooth to an off-board
laptop. The helicopter weighs 4 kg and a 0.48 m long. An aluminum rail was fixed 0.2 m under the aircraft center of gravity, as shown in Figure 1.7. The rail had mounting holes every 25.4 mm to which a fixed mass. The loads can be moved, or dropped mid-test.

![Yale Aerial Manipulator with payload rail and fixed gear](image)

**Figure 1.7: Yale Aerial Manipulator with payload rail and fixed gear [5]**

The off-the-shelf Proportional- Integral-Derivative (PID) flight stabilizer was used to employ this class of control system without choice to more difficult and costly custom solutions. By instantaneously increasing payload mass, the effect of the introduced dynamic load disturbances showed the influence of these disturbances under PID flight control. Additionally, the stability behavior of a helicopter was demonstrated experimentally by instantaneous step payload changes; also examining the loading limits of PID controller quadrotor platforms and comparing it to helicopter platforms.

1.3.3 **Konkuk University Helicopter [6]**. The objective of this research work is to present a basic model and design method for the flight controller to solve the stability of linear helicopter flight dynamics. The Yamaha R-50 helicopter (see Figure 1.8) was used to build the mathematical model and the control system design. The Yamaha R-50 helicopter consists of a two-bladed main rotor with a Bell-Hiller stabilizer bar.
The investigators used the LQR stabilizing control gain matrix and a state space dynamic model of a small-scaled helicopter was used to check the stability. The controller provides the stability for angular velocity, also for the roll, pitch and yaw.

1.3.4 University of South Florida [1]. The objective of this research is to provide the knowledge necessary to design and implement a safe and reliable UAV helicopter testbed.

The helicopter used is the Maxi-Joker 3 weighs 4.5 kg (Figure 1.9). It is equipped with on-board autopilot of 2 Ghz Intel Pentium M Processor with 2 GB Memory and Microstrain 3DM-GX1 Attitudes sensor. It also uses the Hokuyo URG-04LX laser to measure the altitude and 5 Hz Superstar II GPS sensor. The communication between the UAV and the ground station is through Intel Pro 2200 Mini-PCI wireless card.
The Fuzzy logic control theory was implemented successfully including the design and the integration of a small UAV helicopter equipped with the on-board navigation controller, capable of fully autonomous takeoff, waypoint navigation, and landing.

1.3.5 Naval Postgraduate School [7]. The goal is to develop a 6-degree of freedom (6-DOF), nonlinear model using Matlab Simulink, then embed the control technique to the MicroPilot Autopilot for Hardware in the Loop testing.

The platform used is the Align T-REX 600 which weighs 3 kg (Figure 1.10). The nonlinear mode was developed for the helicopter. And by using a PD controller, the flight path software in the loop (SIL) test was designed.

![Figure 1.10: Align T-REX 600 [7]](image-url)
### 1.3.6 Summary of the Previous Work

#### Table 1.1: Summary of the Previous Helicopter Projects

<table>
<thead>
<tr>
<th>Institution</th>
<th>Goals</th>
<th>Platform</th>
<th>Control Theory</th>
<th>Results</th>
<th>Challenges</th>
</tr>
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<tbody>
<tr>
<td>Universiti Teknologi Malaysia</td>
<td>● Autonomous hover maneuver</td>
<td>X-Cell .60</td>
<td>Pole Placement</td>
<td>● Stabilizing the helicopter attitude angles.</td>
<td>The onboard computer unshielded oscillator caused Electromagnetic and radio interference problems.</td>
</tr>
<tr>
<td>Yale University</td>
<td>● Maintain Hover flight when a payload is added</td>
<td>Yale Aerial</td>
<td>Off-the-self PID Controller</td>
<td>● Examining the loading limits of PID controller quadrotor and compare to helicopter.</td>
<td>Payload cannot be precisely positioned at the center of mass of the vehicle</td>
</tr>
<tr>
<td>Konkuk University</td>
<td>● Hover flight</td>
<td>Yamaha R-50</td>
<td>LQR</td>
<td>● Stabilizing the angular velocities.</td>
<td>Large-amplitude motions cannot be applied.</td>
</tr>
<tr>
<td>University of South Florida</td>
<td>● Fully autonomous UAV</td>
<td>Maxi Joker 2</td>
<td>Fuzzy Logic</td>
<td>● Fully autonomous takeoff, waypoint navigation, and landing.</td>
<td>-</td>
</tr>
<tr>
<td>Naval Postgraduate School</td>
<td>● 6-DOF nonlinear model using MathWorks Simulink</td>
<td>Align T-REX 600</td>
<td>PID</td>
<td>● The flight path software in the loop (SIL) test was designed.</td>
<td>-</td>
</tr>
</tbody>
</table>
1.4 Methodology [8]

The AUS UAV development team selected to use commercial-off-the-shelf avionics systems to aid the in-house development and test of flight control systems.

Figure 1.11: Modeling, Simulation and Autopilot Test and Evaluation Process
Figure 1.11 displays a detailed flowchart of the development plan, test and evaluation procedures to be carried out. The flowchart demonstrates that the test and evaluation procedure is an iterative one.

The procedure in the modeling, simulation and autopilot design, test and evaluation process are discussed below:

1.4.1 Select Autopilot System. The autopilot selection was based on its VTOL capability, low price, and long flying range. Table 1.2 and Figure 1.12 show the commercially available autopilots in the market.

Table 1.2: Commercially Available Autopilots

<table>
<thead>
<tr>
<th>System</th>
<th>Family</th>
<th>World</th>
<th>Speed</th>
<th>RAM</th>
<th>Programmable logic</th>
<th>Weight</th>
<th>Ref.</th>
</tr>
</thead>
<tbody>
<tr>
<td>MP2128g</td>
<td>RICS processor</td>
<td>32-bit</td>
<td>150mips</td>
<td>3MB</td>
<td>NO</td>
<td>28g</td>
<td>[8] [9]</td>
</tr>
<tr>
<td>Kestrel 2.4</td>
<td>Rabbit 3000</td>
<td>8-bit</td>
<td>29MHz</td>
<td>512KB</td>
<td>NO</td>
<td>16.7g</td>
<td>[8]</td>
</tr>
<tr>
<td>Piccolo SL</td>
<td>Motorola MPC555</td>
<td>32-bit</td>
<td>40MHz</td>
<td>Unknown</td>
<td>NO</td>
<td>124g</td>
<td>[8]</td>
</tr>
</tbody>
</table>

1.4.2 Select the Flight Test Platform. The Maxi-Joker 3 is a perfect testing platform for flight test autopilot. It has light weight due to its carbon fiber chassis. Also, it can carry up to 20 kg, as claimed by the manufacturer. It has good performance and a 3D flying capability; it also is popular with the research community such as University of California, Berkeley and the University of South Florida. Moreover, Maxi Joker 3 has an electric engine which means less mechanical vibration and noise.
1.4.3 Develop the Aircraft Dynamics Model. A 6-Degree of freedom nonlinear near hover helicopter model has to be developed to simulate the real flight parameters and develop flight control laws. This dynamic model is used to generate the aircraft stability derivatives using linearization techniques provided by Matlab/Simulink toolbox.

1.4.4 Integrate the Dynamic Model into Simulation Packages. A hardware in-loop-simulation (HILS) is the ideal simulation tool as it is the closest to the real environment. The dSpace system was chosen to carry out this task. dSpace is a real-time rapid prototyping system designed for rapid control prototyping and hardware-in-the-loop simulation. Also, dSpace is available in the aeronautics laboratory.

1.4.5 Perform Open Loop Flight Test. To verify the dynamic model, open loop flight tests should be performed to make sure that the model performance is acceptable. The aircraft and the model will be subjected to the same commands inputs.

1.5 Problem Statement

This investigation aims at developing a rotary wing test platform for the purpose of developing an unmanned autonomous VTOL research at the American University of Sharjah. The investigation starts by selecting a VTOL platform that is cost effective, reliable, electric, and has good payload capabilities. Then a mathematical model is develop for the aircraft that captures important dynamics appropriate for developing control laws. The model should be augmented with flight test data to tune the model performance of the real aircraft. The investigation should result establishing a flight simulator with real time capabilities. The investigation then selects a commercial autopilot to be used for flying the aircraft autonomously. Careful autopilot aircraft integration should be carried out with provisions for vibration suppression. Further gain tuning of all autopilot loops needs to be achieved through flight test data logging and analysis. Even though this is difficult, plans need to be established.

Once the MicroPilot integration is accomplished and flight tests are satisfactory, then the next phase of this investigation is to develop an autopilot using successive loop closure PID controllers that are capable of flying the vehicle in the hover mode.
The flight control laws need to produce similar performance to the MicroPilot in the simulation and in the real time hardware in the loop simulation. Finally, use the in-house autopilot to fly the aircraft based on the developed flight control laws and compare its performance to the MicroPilot autopilot system.

In this research a MicroPilot, which is a commercially off-the-shelf (COTS) autopilot system, as selected to be used to fly the Maxi-Joker 3 RC model scale helicopter. The autopilot needs to be integrated and flight tests evaluated through a well-established flight test process to be performed to determine if the MicroPilot could safely control the helicopter for later detailed research and development.

MicroPilot system integration and flight testing require thorough understanding of the helicopter dynamics and control through modeling and simulation of the helicopter for users to be able to integrate the autopilot and go through the appropriate autopilot gain tuning. The Maxi-Joker 3 electric helicopter was chosen as the platform to develop and prove modeling and simulation tools, collect real flight test data for a specific mission, and compare the results with the developed 6-dof helicopter model.

In this work, the helicopter project is established as part of the autonomous system project under development. This Rotary Wing Unmanned Autonomous Vehicle (RUAV) includes integrating, configuring, and flight testing the MicroPilot autopilot system as a benchmark for future development of the helicopter autopilot system. This work is based on developing a model for the helicopter used in the project which is a flybarless helicopter. The objective, once the integration and flight testing of the MicroPilot is completed, is to design PID control laws for hover flight mode for the Maxi-Joker 3 aircraft and to compare the control laws performance with the MicroPilot ones. The control laws are implemented using the Mazari Autopilot which needs to be modified to fit the Maxi-Joker 3 platform.

1.6 Contribution

The contributions of this thesis are: Integrate the Micropilot to a model RC helicopter, perform flight testing including way point navigation, produce a reliable mathematical model for the aircraft, HILS setup, and then design of attitude stabilization system based on Mazari autopilot.
1.6.1 Establishing the AUS helicopter project. As a new platform RUAV at AUS, the outcome of this thesis is to establish the first RUAV helicopter project at AUS.

1.6.2 Designing the platform. The Maxi-Joker3, from minicopter, is the platform that is used in this project, also a MicroPilot autopilot system has been fitted into the Maxi-Joker3. The contributions are stated in the following modifications:

1. Design and build a 6-DOF test stand to experiment the helicopter inside the lab environment.
2. Configure the MicroPilot autopilot system and fit it in the helicopter.
3. Establish commercial RUAV test platform to benchmark RUAV development at AUS in the future.
4. Develop a helicopter flybarless nonlinear model and the control algorithm for the model.
5. Perform Hardware in the loop simulation (HILS) to the non-liner model and a PI hover controller.

1.7 Thesis Outline

After introducing the helicopter’s literature in chapter 1, this thesis will be divided into six chapters as follows:

Chapter 2 derives the non-linear mathematical model of the Maxi Joker 3 helicopter.

Chapter 3 describes the helicopter platform and structure, as well as, the used hardware including MicroPilot autopilot and MPC555 embedded computer. Finally, the safety precautions and the test stand will be presented in this chapter.

Chapter 4 presents the PI controller design for the linear and nonlinear helicopter model.

Chapter 5 consists of two sections: the hardware-in-the-loop-simulation (HILS), the real implementation test, and the micropilot flight test results. An intensive simulation and test results will be presented in this chapter to validate the proposed control algorithms in Chapter 4.
In Chapter 6, we conclude the work carried out in this master’s thesis and suggested future work.
Chapter 2: Helicopter Mathematical Model

2.1 Introduction

This chapter presents the non-linear model of the helicopter with the aid of references [10, 7, 11]. The model will be modified, so that the flybar dynamics is neglected to simulate flybarless Maxi-Joker 3 helicopter model. The model is divided into four blocks. The first block contains the actuators dynamics in which the commanded signals will be generated to the flapping and the thrust equations, and then force and torque equations are derived to get the rigid body dynamics which describes the position and the velocities on the helicopter relative to the earth frame.

In this approach, the helicopter top down modeling are defined where each block is described individually. All the blocks are connected and simulated using Matlab/Simulink to represent a complete model, as shown in Figure 2.1.

![Figure 2.1: Overview of the helicopter non-linear model](image)

The position of the helicopter has to be described. Therefore, description of reference frames and coordinate systems and transformations between the coordinate systems will be discussed in more detail in the next section.

2.2 Reference Frames

To define the equation of motions of a helicopter, frames and notations need to be described precisely:

2.2.1 Body Frame. Body Frame (BF) is the frame whose origin is in the Centre of Gravity (CG) of the helicopter, and it is refereed as \((x^b, y^b, z^b)\). According to the right-hand rule, the BF will identify the orientation of the helicopter; Figure 2.2 illustrates the BF. The BF is important to:

1. Describe Aerodynamic forces and torques.
2. Describe Equations of motion.
3. Measure the data from rate gyros and accelerometers.

2.2.2 Earth Frame. The earth frame (EF) is earth fixed frame denoted by \((x^e, y^e, z^e)\), and it is used to describe the position and the translational motion of the helicopter. The EF can be located on the earth’s surface on a fixed point. The EF is necessary for [13]:

1. GPS and Magnetometer measurements.
2. Flight trajectories and map information.

2.3 Transformation

2.3.1 Rotation Matrix. Forces and moments acting on the vehicle are best described in the body frame (BF), however the guidance and navigation of the helicopter is best presented with reference to the EF. Therefore, the rotation matrix \(R^e_b\) is used to express the BF in EF.

The Euler angles \([\phi, \theta, \psi]\) are used to perform the rotation around \(x\)-axis then \(y\)-axis and at last \(z\)-axis, respectively to align the EF with BF. By using the right handed coordinate frame \((x^e, y^e, z^e)\) [12].

\[
R(z) = \begin{bmatrix}
\cos(\psi) & -\sin(\psi) & 0 \\
\sin(\psi) & \cos(\psi) & 0 \\
0 & 0 & 1
\end{bmatrix}
\]  

2-1
\[
R(y) = \begin{bmatrix}
\cos(\theta) & 0 & \sin(\theta) \\
0 & 1 & 0 \\
-\sin(\theta) & 0 & \cos(\theta)
\end{bmatrix}
\]

\[
R(x) = \begin{bmatrix}
1 & 0 & 0 \\
0 & \cos(\phi) & -\sin(\phi) \\
0 & \sin(\phi) & \cos(\phi)
\end{bmatrix}
\]

where \(R(x), R(y)\) and \(R(z)\) denote transformation from BF to the EF. Due to the principal of orthonormality transposing principle, the resulting transformation matrix \(R^e_b\) is:

\[
R^e_b = R(x)R(y)R(z) = \\
\begin{bmatrix}
\cos(\theta)\cos(\psi) & \cos(\psi)\sin(\theta)\sin(\phi) - \sin(\psi)\cos(\phi) & \cos(\phi)\sin(\theta)\cos(\phi) + \sin(\psi)\sin(\phi) \\
\cos(\theta)\sin(\psi) - \sin(\theta)\cos(\phi) & \cos(\psi)\sin(\phi)\sin(\theta) + \sin(\psi)\cos(\phi) & \sin(\psi)\cos(\phi) - \cos(\psi)\sin(\theta) \\
-\sin(\theta) & \cos(\theta)\sin(\phi) & \cos(\theta)\cos(\phi)
\end{bmatrix}
\]

The functionality of the rotation matrix can be used to state the position vector \((X^b)\) in BF to its corresponding vector \((X^e)\) in EF:

\[2.3.2 \text{ Euler Rates.} \] The body angular rates of the helicopter are designated as \(\omega^b = [p \; q \; r]^T\), and it is expressed about the BF. The Euler rates \(\hat{\Theta} = [\dot{\phi} \; \dot{\theta} \; \dot{\psi}]^T\) are the angular velocity of the BF with respect to the EF. The relationship between \(\omega^b\) and \(\hat{\Theta}\) can be expressed in the following relationships [13]:

\[
X^e = R^e_b \cdot X^b
\]

\[
\begin{bmatrix}
p \\
q \\
r
\end{bmatrix} = \begin{bmatrix}
1 & 0 & 0 \\
0 & \cos(\phi) & -\sin(\phi) \\
0 & \sin(\phi) & \cos(\phi)
\end{bmatrix} \begin{bmatrix}
\dot{\phi} \\
\dot{\theta} \\
\dot{\psi}
\end{bmatrix} + \begin{bmatrix}
1 & 0 & 0 \\
0 & \cos(\phi) & -\sin(\phi) \\
0 & \sin(\phi) & \cos(\phi)
\end{bmatrix} \begin{bmatrix}
\cos(\theta) & 0 & \sin(\theta) \\
0 & 1 & 0 \\
-\sin(\theta) & 0 & \cos(\theta)
\end{bmatrix} \begin{bmatrix}
0 \\
0 \\
0
\end{bmatrix}
\]

Inverting the transformation matrix results in:

\[
\begin{bmatrix}
\dot{\phi} \\
\dot{\theta} \\
\dot{\psi}
\end{bmatrix} = \begin{bmatrix}
1 & \tan(\theta)\sin(\phi) & \tan(\theta)\cos(\phi) \\
0 & \cos(\phi) & -\sin(\phi) \\
0 & \sec(\theta)\sin(\phi) & \sec(\theta)\cos(\phi)
\end{bmatrix} \begin{bmatrix}
p \\
q \\
r
\end{bmatrix}
\]
2.4 Rigid Body Dynamics

The rigid body equation of motion of the helicopter is presented. The equations of motion of the rigid body are as follows:

\[
\dot{V} = \frac{F}{m} - \omega \times V \\
\dot{\omega} = I^{-1}(M - \omega \times I\omega)
\]

where, \( V \): Velocity in the BF (m/s), \( \omega \): Body angular velocity (rad/sec), \( M \): Net moments about the CG (N.m), \( F \): Net force acting on the helicopter (N), and \( I \): Moment of inertia Matrix (kg.m^2):

\[
V = \begin{bmatrix} u \\ v \\ w \end{bmatrix}, \quad F = \begin{bmatrix} f_x \\ f_y \\ f_z \end{bmatrix}, \quad \omega = \begin{bmatrix} p \\ q \\ r \end{bmatrix}, \quad I = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix}, \quad \text{and} \quad M = \begin{bmatrix} L \\ M \\ N \end{bmatrix}
\]

2.5 Forces and Moments

The forces and moments are derived with respect to the helicopter CG, and they are expressed in the BF, as shown in Figure 2.3. The complexity of the forces and moments equation need to be simplified, therefore thrust direction is assumed to be perpendicular to the tip path plane (TTP), also the vertical fin and horizontal stabilizer effect will be ignored because of their small influence on the helicopter’s dynamics at hover.

![Figure 2.3: Moments and forces acting on helicopter [10]](image)
2.5.1 Forces. The resulting force $F$ acting on the helicopter, in the BF, is decomposed along three axes $f_x, f_y$ and $f_z$. These forces consist of the following four main components:

$F_{MR}$: The resulting force generated by the main rotor thrust, and it is acting in the center of the main rotor disc.

$F_{TR}$: The resulting force generated by the tail rotor thrust, and it is acting on the center of the tail rotor disc.

$F_g$: The resulting force generated by the gravitational acceleration, and it is acting in the CG.

$F_{fus}$: The resulting drag force generated by the fuselage, and it is acting in the CG.

The main rotor thrust is defined by $\beta_{1s}$ and $\beta_{1c}$ in the lateral and longitudinalordinations, respectively, as demonstrated in Figure 2.4.

The components of the forces caused by the main rotor will be a function of $\beta_{1s}$ and $\beta_{1c}$:

\[
\begin{align*}
    f_{x,MR} &= -T_{MR} \cdot \sin(\beta_{1c}) \cos(\beta_{1c}) \\
    f_{y,MR} &= T_{MR} \cdot \sin(\beta_{1s}) \cos(\beta_{1s}) \\
    f_{z,MR} &= -T_{MR} \cdot \cos(\beta_{1s}) \cdot \cos(\beta_{1c})
\end{align*}
\]

Figure 2.4: The lateral ($\beta_{1s}$) and longitudinal ($\beta_{1c}$) angles between the hub plane (HP) and the tip path plane (TPP) shows the main rotor orientation
The force generated by the tail rotor is only in y direction and can be written as:

\[ f_{y,TR} = -T_{TR} \quad \text{2-14} \]

The force generated by the gravitational acceleration has three components, and can be written as:

\[ f_{x,g} = -\sin(\theta) \cdot m \cdot g \quad \text{2-15} \]
\[ f_{y,g} = \sin(\phi) \cdot \cos(\theta) \cdot m \cdot g \quad \text{2-16} \]
\[ f_{z,g} = \cos(\phi) \cdot \cos(\theta) \cdot m \cdot g \quad \text{2-17} \]

For hover flight the induced velocity for the main rotor thrust was obtained assuming that the inflow is steady and uniform. Figure 2.5 shows the helicopter inflow in the hover mode.

![Rotor Tip Vortex](image)

Figure 2.5: Helicopter inflow at hover [14]

The induced velocity is found based on momentum theory assuming that the vehicle is in hover flight condition using the following relation [10]:

\[ v_{ihover} = \sqrt{\frac{mg}{2\pi \rho R_{mr}^2}} \quad \text{2-18} \]

\( v_{ihover} = 3.41 \) m/sec for Max-Joker 3 model. This induced flow caused by the rotor downwash generates drag forces opposing the direction of the helicopter movement. The fuselage drag forces in the X, Y, and Z axes are given by:

\[ f_{x,fus} = -S_x \frac{1}{2} \rho v_{ihover} u \quad \text{2-19} \]
where \( V_\infty \) can be expressed as:

\[
V_\infty = \sqrt{u_a^2 + v_a^2 + (w_a + v_{ihover})^2}
\]

The vehicle cross sectional areas are estimated to: \( S_x^{fus} = 0.0662 \), \( S_y^{fus} = 0.0872 \), and \( S_z^{fus} = 0.0993 \, m^2 \).

2.5.2 Moments. The resulting moments acting on the helicopter are resolved in the BF along three axes are \( L, M \) and \( M \). These moments results from the following moments:

\( \tau_{MR} \): The moment generated by the tilting of the thrust vector and the restraint in the blade attachment of the main rotor.

\( \tau_{TR} \): The moment generated by the tail rotor.

\( \tau_D \): The moment generated by drag on the main rotor.

The drag moment caused by the tail rotor is neglected because it has relatively small effect on the helicopter. The moments are defined positive clockwise direction. The rotor flapping moments are illustrated in Figure 2.6 in the roll direction. The restraints in the blade attachment can be approximated as a linear torsional spring with a constant stiffness coefficient \( K_\beta \) [10].

![Figure 2.6: Main rotor moments acting on the helicopter fuselage [10]](image)

The moments generated by the main rotor consists of three components:
For the flybarless helicopter roll moment and pitch moment derivatives ($L_b$, $M_a$) will be set to zero because they represent the flybar dynamics. The total torque coefficient is given by [15]:

\[
L_{MR} = f_{y,MR} \cdot h_{mr} - f_{z,MR} \cdot y_{mr} + K_{\beta} \cdot \beta_{1s} - L_b \tau_{fs} p \\
M_{MR} = f_{x,MR} \cdot h_{mr} + f_{z,MR} \cdot l_{mr} + K_{\beta} \cdot \beta_{1c} - M_a \tau_{fc} q \\
N_{MR} = -\rho(\Omega R)^2 \frac{b c R^2}{s} C_{Q}^{MR}
\]

where the drag coefficient $c_d = 0.024$, and $s$ is the rotor solidity which is total blade area to the disk area.

Figure 2.7: $h_{mr}$ is the distance from CG the main rotor, and $h_{tr}$ is the distance from CG the tail rotor
Figure 2.8: $l_{mr}$ is the distance from CG to the main rotor, and $l_{tr}$ is the distance from CG to the tail rotor. $y_m$ is the distance from CG to the main rotor.

The moments generated by the tail rotor consists of two components:

\[ L_{TR} = f_{y,TR} h_{tr} \]  \hspace{1cm} 2-28
\[ N_{TR} = -f_{y,TR} l_{tr} \]  \hspace{1cm} 2-29

2.6 Flapping and Thrust Equations

This section presents the derivation of the flapping and thrust equations generated by the main and tail rotors.

2.6.1 Main Rotor Thrust and Inflow Equations. The main rotor thrust is directly linked to the pilot controlled collective and indirectly associated with the translatory velocity states ($u, v, w$) and the rotor induced velocity ($v_i$). This relation is given by [11, 16] as:

\[ T_{MR} = \left[ \theta_o \left( \frac{1}{3} + \frac{\mu^2}{2} \right) - \left( \frac{\mu z + \lambda_i}{2} \right) \right] \frac{a S}{2} \rho (\Omega R)^2 A_d \]  \hspace{1cm} 2-30

The rotor inflow, $\lambda_i$, has a direct impact on the thrust which makes the calculation of the thrust difficult. The advance ratio, $\mu$, is the ratio between helicopter translational velocity and main rotor tip speed, where $\mu_z$ is the ratio of vertical velocity to main rotor tip speed, $\lambda_i$ is the inflow ratio for main rotor, and $a$ is the main rotor lift curve slope.
Padfield [11] demonstrates that the rotor thrust is proportional to the rotor inflow. Also, the thrust increases when the pilot commands the collective, which is assigned by $\theta_0$. The main rotor thrust equation is solved iteratively until $T_{MR}$ and $v_i$ are converged using Newton-Raphson iteration technique explained in [7].

The advance ratio between helicopter translational velocity and main rotor tip speed is expressed by:

$$\mu = \frac{\sqrt{u_a^2 + v_a^2}}{\Omega R} \quad 2-31$$

The advance ratio in the z direction:

$$\mu_z = \frac{w_a}{\Omega R} \quad 2-32$$

The hover inflow $\lambda_{hover}$ where the main rotor thrust is equal to the helicopter weight is obtained from momentum theory using the following equation:

$$v_{i_{hover}} = \sqrt{\frac{T}{2 \rho A_d}} \quad 2-33$$

The main rotor thrust at hover is equal to the helicopter weight $W = 7.3 \times 9.81 = 71.613 N$. The main rotor disk area is $A_d = \pi R^2 = \pi \times (0.89)^2 = 2.4885 m^2$, that yields to $v_{i_{hover}} = 3.41 m/s$, and then normalizing by the rotor tip speed ($\Omega R$) results with $\lambda_{i_{hover}} = 2.533 \times 10^{-4}$. The thrust coefficient at hover can be expressed as:

$$C_T = \frac{T_{MR}}{\rho (\Omega R)^2 A_d} \quad 2-34$$

Also, solving equation 2-30 to find the main rotor collective trim at hover yields:

$$\theta_{0_{hover}} = 3 \left[ \frac{2 c_T}{a_s} + \frac{\lambda_{i_{hover}}}{2} \right] \quad 2-35$$

$$\theta_{0_{hover}} = 0.0713 \text{ rad (4.086°)}$$

The main rotor collective ranges between -3° to +10° with a zero collective input that generates the trim main rotor collective of 4.086°.
2.6.2 Tail Rotor Thrust and Inflow Equations. The tail rotor performs the same way the main rotor does. A tail rotor collective command controls the tail rotor. The total rotor thrust is given by:

\[ T_{TR} = \left[ \delta_{ped} \left( \frac{1}{2} + \frac{\mu_{tr}}{2} \right) - \left( \frac{\mu_{ztr} + \lambda_i}{2} \right) \right] \frac{a_s}{2} \rho (\Omega_{tr} R_{tr})^2 A_{dtr} \] \hspace{1cm} (2-36)

The tail rotor inflow velocity increases with increasing the rotor collective command \( \delta_{ped} \) results in increasing the tail thrust; the same iterative technique will be used to find the convergent inflow for the tail rotor. Where the normalized rotor inflow is given by [10]:

\[ \mu_{ztr} = \frac{v_{tr}}{\Omega_{tr} R_{tr}} \] \hspace{1cm} (2-37)

The tail rotor hub airspeed in the y direction is given by [10]:

\[ v_{tr} = v_a - l_{tr} r - h_{tr} |p| \] \hspace{1cm} (2-38)

where \( v_a \) is the resultant airspeed of the vehicle in the y direction subtracted for wind air speed. The tail rotor hub airspeed in the z direction is given as:

\[ w_{tr} = w_a - l_{tr} q - Kv_{imr} \] \hspace{1cm} (2-39)

where, \( w_a \), which is the vehicle airspeed in the z direction subtracted for wind air speed. The term \( K \) is a wake intensity factor that rises as the tail rotor becomes more submerged in the main rotor wake. The wake intensity factor can be considered zero near hover flight [10]. The tail rotor hub airspeed in the x direction is given by [7]:

\[ u_{tr} = u_a + |l_{tr} q + l_{tr} r| \] \hspace{1cm} (2-40)

The tail rotor advance ratio is stated as [10]:

\[ \mu_{tr} = \frac{\sqrt{u_{tr}^2 + w_{tr}^2}}{\Omega_{tr} R_{tr}} \] \hspace{1cm} (2-41)

The torque produced by the main rotor is \( N_{MR} = 7.9331 N.m \) (from equation 2-25). Dividing this by the tail arm \( l_{tr} = 1.068 m \), this yields to the force needed by the tail rotor to cancel the main rotor torque effect:
$f_{y,TR(\text{hover})} = 7.4279 \, N$

$f_{y,TR(\text{hover})}$ has to be positive in the y-direction to counteract the positive yaw produced by the main rotor. The tail rotor coefficient of thrust can be calculated at hover flight regime from the airspeed and angular rate conditions as described by Padfield [11].

$$f_{y,TR} = \rho(\Omega_{tr} R_{tr})^2(\pi R_{tr}^2)C_{T(tr)}f_T$$  \hspace{1cm} 2-42

The torque produced by the main rotor is $N_{MR} = 7.9331 \, Nm$. Dividing this by the tail arm $l_{tr} = 1.068 \, m$, this yields to the force needed by the tail rotor to cancel the main rotor torque effect:

$$f_{y,TR(\text{hover})} = 7.4279 \, N$$

$f_{y,TR(\text{hover})}$ has to be positive in y-direction to counteract the positive yaw produced by the main rotor. The tail rotor coefficient of the thrust can be calculated at hover flight regime from the airspeed and angular rate conditions as described by Padfield [11].

$$f_{y,TR} = \rho(\Omega_{tr} R_{tr})^2(\pi R_{tr}^2)C_{T(tr)}f_T$$  \hspace{1cm} 2-43

where the coefficient of the tail rotor thrust is:

$$C_{T(tr)} = \frac{T_{TR}}{\rho(\Omega_{tr} R_{tr})^2(\pi R_{tr}^2)}$$  \hspace{1cm} 2-44

$$f_{y,TR} = T_{TR}f_T$$  \hspace{1cm} 2-45

where the blockage factor $f_T$ is for the thrust losses in pusher type tail rotors, and since this is a tractor type tail rotor [7], the blockage factor will be ignored, therefore:

$$f_{y,TR} = T_{TR}$$  \hspace{1cm} 2-46

Solving for the thrust coefficient at hover gives:

$$C_{T(tr)}^{\text{hover}} = 6.356 \times 10^{-3}$$

At hover, the tail rotor thrust can be expressed by the relation [10]: $T_{TR} = 2\rho A_{c_{tr}v_{itr}^2}$, and solving for the induced velocity at hover yields the value of the tail induced velocity:
\[
\psi_{i(\text{tr})}^{\text{hov}} = 6.1382 \text{ m/s}
\]

\[
\lambda_{i}^{\text{hov}} = \frac{\psi_{i(\text{tr})}^{\text{hov}}}{\Omega_{\text{tr}}R_{\text{tr}}} = 56.361 \times 10^{-3}
\]

Solving equation 2-36 for the trim tail rotor collective at hover gives:

\[
\delta_{\text{ped}}^{\text{hov}} = 3 \left[ \frac{2C_{T(\text{tr})}^{\text{hov}}}{\text{a.s}} + \frac{\lambda_{i}^{\text{hov}}}{2} \right] \quad 2-47
\]

\[
\delta_{\text{ped}}^{\text{hov}} = 0.143 \left( 8.2^\circ \right)
\]

The tail rotor collective ranges used for this model will be \(-16^\circ\) to \(+22^\circ\) with a zero collective input that generates the trim tail rotor collective of \(8.2^\circ\).

### 2.6.3 Main Rotor Flapping Dynamics.

The main rotor flapping dynamics derived by Mettler [17] will be used in the present model. The longitudinal and lateral flap angle commands, \(\beta_{c}^{\text{c}}\) and \(\beta_{s}^{\text{s}}\), are proportional to the inputs \(u_{\text{long}}\) and \(u_{\text{lat}}\). The linear flapping dynamics can be express as:

\[
\begin{bmatrix}
\dot{\beta}_{1c} \\
\dot{\beta}_{1s}
\end{bmatrix} = \begin{bmatrix}
-\frac{1}{\tau_{fc}} & 0 \\
0 & -\frac{1}{\tau_{fs}}
\end{bmatrix} \begin{bmatrix}
\beta_{1c} \\
\beta_{1s}
\end{bmatrix} + \begin{bmatrix}
0 & -1 & A_{\delta_{\text{lon}}} \\
-1 & 0 & 0 \\
0 & -1 & B_{\delta_{\text{lat}}}
\end{bmatrix} \begin{bmatrix}
p \\
q \\
\beta_{1c}^{c}
\end{bmatrix} \quad 2-48
\]

where \((\tau_{fc}, \tau_{fs})\) are the main rotor time constant, they can be defined form the system identification. At hovering conditions the value of \(\tau_{fc} = 0.113\ \text{sec}\), \(\tau_{fs} = 0.101\ \text{sec}\), \(A_{\delta_{\text{lon}}} = 4.2\), and \(B_{\delta_{\text{lat}}} = 4.2\) which are obtained from [17] and [18].

### 2.7 Actuator Models

The ranges of the commands for the cyclic pitch angles, collective angle, and peddle angle are defined in degrees:

\[
u_{\text{lat}}^{\text{max}} = \pm 10^\circ
\]

\[
u_{\text{lon}}^{\text{max}} = \pm 10^\circ
\]

\[
u_{\text{col}}^{\text{max}} = -3\sim10^\circ
\]

\[
u_{\text{ped}}^{\text{max}} = -16^\circ\sim22^\circ
\]
The servos model is obtained from [10]:

\[
H_{\text{servo}}(s) = \frac{s/T_z + 1}{s/T_p + 1/s^2 + 2\zeta\omega_n s + \omega_n^2}
\]  

where \( T_z = 104 \) sec, \( T_p = 33 \) sec, \( \omega_n = 36 \) rad/sec, and \( \zeta = 0.5 \). The tail servo was approximated by a second order system with the undamped natural frequency of 7 Hz and the damping ratio of 0.6.

The servo rotations are transferred through mechanical linkages that control the helicopter control surfaces. Figure 2.9 shows the command signal flow from the servo input to the actual flap angle \( \beta_{1c} \).

\[K_{MR}\] is the mechanical linkage gain between the servo and actual flap angle, which is estimated to 0.5 using the following equation and Figure 2.10:

\[
\theta_b = \frac{c r}{e b} = K_{MR} \theta_{sw}
\] 

The Maxi Joker 3 has three servos controlling the swash plate where they are linked together using a 120° cyclic collective pitch mixing (CCPM) servo configuration, see Figure 2.11. The servo mixing can be expressed using a set of liner
equations to calculate the servo positions with respect to the control signal \((u_{col}, u_{lat}, u_{long})\), as follows [20]:

\[
\begin{bmatrix}
    u_{col} \\
    u_{lat} \\
    u_{long}
\end{bmatrix} =
\begin{bmatrix}
    1 & \cos(120^\circ) & \sin(0^\circ) \\
    1 & \cos(-120^\circ) & \sin(-120^\circ) \\
    1 & \cos(120^\circ) & \sin(120^\circ)
\end{bmatrix}
\begin{bmatrix}
    \text{servo}_{\text{front}} \\
    \text{servo}_{\text{right}} \\
    \text{servo}_{\text{left}}
\end{bmatrix}
\]

Figure 2.11: Maxi Joker 3 servo configuration [20]
Chapter 3: Experimental Setup

3.1 Introduction

This chapter describes the helicopter testbed and the testing-stand for indoor flying. MicroPilot autopilot system integration is also presented showing the process of configuring and flight testing the UAV platform. Finally, The Mazari Autopilot based on MPC555 embedded system, used for testing the developed control algorithms, is presented in this chapter.

3.2 Description of the Hardware Platform

The Maxi-joker 3 is a commercial-off-the-shelf helicopter that is:

- Capable of carrying payloads of more than 4.5 kg.
- Features brushless dc motor with 12s/5000 mAh Lithium-Polymer Battery.
- Offers high maneuverability at a reasonable cost that can be affordable to be used as a research platform to minimize development cost.
- Extremely stable and rigid chassis.
- Integrated host of Flybarless electronics in a well-protected chassis.
- Compartments for components above the flight battery.
- Easily removable tail boom for transportation and maintenance purposes
- Direct CCPM linkages to the swashplate without the push-pull linkages.
- Robust flybarless rotor damping with different hardness.

The Maxi-Joker 3 platform is shown in Figure 3.1. The Maxi-Joker 3 is an electric platform which has an advantage of low vibration compared with a gasoline and turbine based helicopters; however it has only average run time nearly 15-20 minutes.

The structure of the Maxi-joker 3 has been modified to meet the project requirements, as shown in Figure 3.2. A new skid has been made from aluminum alloy to give the helicopter enough space above the ground to achieve a safe and smooth landing.
Additionally, a carbon fiber compartment has been mounted at the back of the helicopter in order to make enough room for the autopilot and its subsystems; where it is supported with shock absorbers to isolate the system and avoid excessive vibrations.

The Maxi-Joker 3 is also equipped with a Microbeast heading hold, a Futaba BLS9251 tail servo, and three Futaba BLS452 main servos gyro (see Appendix A). This platform was chosen over the previously mentioned platforms due to its cost, approximately $3000 USD ready-to-fly, desire to avoid carrying and storing explosive fuel, reduced vibrations, relatively small size, and ability to handle wind gust exceeding 20 mph. Also, the electric propulsion system is faster to setup and to control. Table 3.1 demonstrates the helicopter’s equipment and main physical parameters.
### Table 3.1: Parameters of AUS Maxi Joker 3 Helicopter

<table>
<thead>
<tr>
<th>Base platform</th>
<th>Maxi-Joker 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dry weight</td>
<td>4.22 kg (without batteries)</td>
</tr>
<tr>
<td>Powerplant</td>
<td>DC Brushless motor, 12 cell Battery (LiPo)</td>
</tr>
<tr>
<td>Endurance:</td>
<td>15-20 min</td>
</tr>
<tr>
<td>Autopilot</td>
<td>MP2128g/AUS Mazari</td>
</tr>
<tr>
<td>Speed Controller:</td>
<td>Kontronic 80+ HIV</td>
</tr>
</tbody>
</table>

**Helicopter parameters**

- **L = 1.50284 m**  
  helicopter length
- **H = 0.5615 m**  
  helicopter height
- **m = 7.297 kg**  
  helicopter mass
- **Ixx=0.16347 kg.m²**  
  pitching moment of inertia
- **Iyy=0.419 kg.m²**  
  rolling moment of inertia
- **Izz=0.304 kg.m²**  
  yawing moment of inertia
- **Dv/GW = 0.012371925**  
  vertical drag and gross weight ratio

**Main Rotor:**

- **R = 0.89 m**  
  main rotor radius
- **Ad = 2.488456 m²**  
  main rotor disc area
- **hm = 0.240301 m**  
  main rotor hub height above CG
- **c = 60 mm**  
  main rotor chord
- **\( K_\beta \) = 91.66 N.m/rad**  
  hub torsional stiffness
- **\( \tau_{fl} = 0.1011 \) sec**  
  lateral flapping time constant
- **\( \tau_{flc} = 0.113 \) sec**  
  longitudinal flapping time constant
- **\( a_{sw} = 5.5 \) rad−1**  
  blade lift curve slope
- **\( c_a = 0.024 \)**  
  drag coefficient
- **N = 2**  
  number of blades

**Tail Rotor**

- **R_tr = 0.16 m**  
  tail rotor radius
- **A_tr = 0.0804 m²**  
  tail rotor disc area
- **h_tr = 0.077903 m**  
  tail rotor hub height above CG
- **l_tr = 1.068559 m**  
  tail rotor hub location behind CG
- **c_tr = 30 mm**  
  tail rotor chord

### 3.3 6-DOF Test Stand

The 6-DOF test stand was designed for flight testing and PID gains tuning indoor without the need for a skilled pilot and to avoid wind disturbances. The test stand was designed using SolidWorks 3D modeling, as shown in Figure 3.3. A steel rod can
be inserted through the helicopter’s CG which gives a free motion around the y-axis, also a steel fork can rotate freely around the x-axis offering the roll motion. Linear bearings have been installed inside the main shaft in order to make free motion around and in z-direction simulating the yaw and heave motions. The test stand can move in x and y-directions, but these extra freedoms are constrained for safety purposes.

![Diagram of 6-DOF Helicopter Test Stand](image)

**Figure 3.3:** 6-DOF Helicopter Test Stand (Upper Part Only)

### 3.4 Hardware Description

The main hardware components of the RUAV system consist of MP2128\textsuperscript{LRC} autopilot or Mazari. The next few section describe each autopilot in more details.

#### 3.4.1 MP2128\textsuperscript{LRC} Autopilot

The MP2128\textsuperscript{LRC} (see Table 3.2) weighs 437g supported with two microhard modems. It has also roll and pitch sensors of 30Hz with ±90° dynamic range, accelerometers of 5Hz update rate with ±2G dynamic range, and rate gyros of 30 Hz update rate with ±150°/s dynamic range. Also, the MP2128\textsuperscript{LRC} is provided with RISC processor with speed of 150 mips and a RAM of 3MB where the date can be logged either at 5 Hz or 30 Hz, and then saved in a flash memory. In this project the data is logged at 30 Hz to capture the helicopter’s fast dynamics [9].
Table 3.2: MP2128\textsuperscript{LRC} Autopilot [21]

<table>
<thead>
<tr>
<th>Autopilot</th>
<th>MP2128\textsuperscript{LRC}</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight</td>
<td>437g</td>
</tr>
<tr>
<td>Dimensions(H W L)</td>
<td>43mm×78mm×137mm</td>
</tr>
<tr>
<td>Measurements</td>
<td>Roll &amp; Pitch</td>
</tr>
<tr>
<td>Update Rate</td>
<td>30Hz</td>
</tr>
<tr>
<td>Dynamic Range</td>
<td>±90°</td>
</tr>
<tr>
<td>Accuracy (in static)</td>
<td>±0.3°</td>
</tr>
<tr>
<td>Accuracy (in dynamic)</td>
<td>±1.7°</td>
</tr>
<tr>
<td>Accelerometers</td>
<td></td>
</tr>
<tr>
<td>Update Rate</td>
<td>5Hz</td>
</tr>
<tr>
<td>Dynamic Range</td>
<td>±2G</td>
</tr>
<tr>
<td>Resolution</td>
<td>5mg at 60 Hz Bandwidth</td>
</tr>
<tr>
<td>Sensor Resonant Frequency</td>
<td>10kHz</td>
</tr>
<tr>
<td>Temperature Operating Range</td>
<td>−40→+85°C</td>
</tr>
<tr>
<td>Rate Gyro</td>
<td></td>
</tr>
<tr>
<td>Update Rate</td>
<td>30Hz</td>
</tr>
<tr>
<td>Dynamic Range</td>
<td>±150°/s</td>
</tr>
<tr>
<td>Resolution</td>
<td>12.5mV/°/s</td>
</tr>
<tr>
<td>Linear Acceleration Effect</td>
<td>0.2°/s/g</td>
</tr>
<tr>
<td>Sensor Resonant Frequency</td>
<td>14kHz</td>
</tr>
<tr>
<td>Temperature Operating Range</td>
<td>−40→+85°C</td>
</tr>
<tr>
<td>Data log rate</td>
<td>5Hz or 30Hz</td>
</tr>
<tr>
<td>Power</td>
<td>7.3V – 12V</td>
</tr>
<tr>
<td>Modems Range</td>
<td>50 Km</td>
</tr>
<tr>
<td>Processor</td>
<td>RICS, 150mips</td>
</tr>
<tr>
<td>RAM</td>
<td>3MB</td>
</tr>
<tr>
<td>AGL</td>
<td>Ultra Sonic Sensor</td>
</tr>
</tbody>
</table>

A complete detail of the autopilot is provided in MicroPilot Autopilot Installation and Operation manual [21].

The MICROBEAST [22], three axis-gyros, has been replaced with the MP2128LRC to control the swash palate servos. However, the tail servo is controlled from the three axis gyro, since MP2128LRC does not support fast tail servos.

3.4.2 **MP2128LRC Autopilot Experimental Setup.** To achieve safe and reliable flight tests with acceptable performance, many technical issues had to be overcome, such as mechanical vibrations, acoustic noise, and compass calibration.
3.4.2.1 Mechanical Vibrations. One of the main problems is the mechanical vibration which causes the helicopter’s attitudes to change if the sensors saturation happens due to excessive vibration levels; therefore, appropriate shock absorbers have been installed underneath the autopilot, as shown in Figure 3.4.

![Shock Absorbers Located Below the Autopilot](image)

Figure 3.4: Shock Absorbers Located Below the Autopilot

Figure 3.5 represents the vibration levels before and after installing the shock absorbers. The figure shows the mechanical vibrations power spectra were reduced by more 100 times after installing the shock absorbers.
Figure 3.5. Frequency response. (a) Before Inserting the Shock Absorbers, (b) After Inserting the Shock Absorbers

3.4.2.2 Acoustic Noise. The acoustic noise was due to the rotor and vibrations that affected the SONAR sensor reading. Therefore, the sensor was isolated using foam and then inserted inside an aluminum cone which helps to focus the sound wave and enhance the readings as shown in Figure 3.6.
Figure 3.6: SONAR Sensor Setup

Figure 3.7 shows the ultrasonic sensor readings enhancement after adding the aluminum cone.

3.4.2.3 Compass Calibration. The compass module is used to give an accurate heading. Thus, necessary calibration is needed to remove the biases caused by the electromagnetic fields generated by the electronics and ferrous materials around. A special mechanism has been designed to calibrate the compass while it is installed on the helicopter, as shown in Figure 3.8.
3.4.3 AUS Mazari Rotary Wing Autopilot. The rotary Wing UAV is based on fixed wing Mazari UAV Autopilot. This avionics system was developed and improved [18, 23] using freescale MPC555 processor. The software for the Rotary Wing UAV was then loaded to the flight control computer (MPC555 processor). A detailed description of the fixed wing AUS Mazari Autopilot is presented in [23].

3.4.3.1 MPC555 Motorola Microcontroller. The MPC555 Motorola Microcontroller is a high-speed 32-bit CPU Unit that contains a floating point unit (32-bit Power PC MPC555 CPU 40MHz) [24]. The MPC555 has high-performance data manipulation and relatively large on-chip Flash memory (512kB Flash EPROM) with powerful peripheral subsystems makes from the MPC555 a powerful single board computer. The MPC555 is capable to process the control algorithms while communicating with many sensor devices instantaneously. It contains eight channel 16-bit PWM systems and contains three Time Processing Units (TPUs) reading and sending PWM signals with different devices. Each TPU will distribute the interrupt processing load giving the CPU less load to process the control algorithm. Each TPU is considered as a sub-microcontroller that is employed to share the processing loads with the main CPU.

Other features of the MPC555 used in this project are: an additional 26 KB on-chip SRAM and 448 KB on-chip Flash memory is available on the MPC555. The 272-pin BGA controller and boasts a 64-bit Floating Point Unit, dual on-chip Full 2.0B
CAN, SPI, also dual 16-channel A/D converters with 10-bit resolution [25]. Figure 3.9 shows the phyCORE-MPC555 from phytec that is used in the onboard system.

Figure 3.9: phyCORE-MPC555 board

The peripherals used in this project are:

- UART-1 to read data from the laser sensor.
- UART-2 to communicate with the ground system through Mazri ground station.
- Six pulse width generation for the actuators.
- CanBusA to collect the data from the MTIDG unit after processing the data using PIC 18F microcontroller. [23]

The MPC555 microcontroller is the Matlab/Simulink which provides the Target Support Package FM5 allowing the user to deploy a production code created from Real-Time Workshop Embedded Coder onto the MPC555 microcontroller. Drivers such as pulse width modulation (PWM), AN, serial, analog output, and TPU are supported by Target Support Package FM5. Theses drivers are combined with control system algorithms for real-time implementation.

3.4.3.2 **MIDG IMU / GPS.** The Microbotics MIDG II is an Inertial Navigation System (INS) with Global Positioning System (GPS) enclosed in a very small package. This INS GPS package is ideally suitable Mazari Rotary UAV autopilot for its light weight, low power consumption, and quiet small size. The MIDG specifications are show in Table 3.3.
Table 3.3: MIDG IMU / GPS SPECIFICATIONS [26]

<table>
<thead>
<tr>
<th>Sensor</th>
<th>MIDG IMU / GPS</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Weight</strong></td>
<td>55 g</td>
</tr>
<tr>
<td><strong>Dimensions</strong></td>
<td>1.5” x 0.81” x 1.725”</td>
</tr>
<tr>
<td><strong>Power</strong></td>
<td>10 VDC - 32 VDC</td>
</tr>
</tbody>
</table>

**Measurements**

<table>
<thead>
<tr>
<th></th>
<th>Attitude (pitch and roll, with GPS)</th>
<th></th>
<th>Attitude (heading with GPS and maneuvering)</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Update Rate</strong></td>
<td>50Hz</td>
<td><strong>Update Rate</strong></td>
<td>50Hz</td>
<td></td>
</tr>
<tr>
<td><strong>Accuracy</strong></td>
<td>0.4° (1 σ)</td>
<td><strong>Accuracy</strong></td>
<td>2° (1 σ)</td>
<td></td>
</tr>
</tbody>
</table>

**GPS**

<p>| | | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Update Rate</strong></td>
<td>4Hz</td>
<td><strong>Accuracy (Position)</strong></td>
<td>2m (CEP) with WAAS/EGNOS available, 3m (CEP) otherwise</td>
</tr>
<tr>
<td><strong>Accuracy (Altitude)</strong></td>
<td>3m (SEP) with WAAS/EGNOS available, 5m (SEP) otherwise</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Accuracy (Velocity)</strong></td>
<td>&lt; 0.2 m/s</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

**Accelerometers**

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Update Rate</strong></td>
<td>50Hz</td>
<td></td>
</tr>
<tr>
<td><strong>Dynamic Range</strong></td>
<td>±6G</td>
<td></td>
</tr>
<tr>
<td><strong>Non-Linearity</strong></td>
<td>0.1% of FS</td>
<td></td>
</tr>
<tr>
<td><strong>Noise Density</strong></td>
<td>0.1 °/sec / √Hz</td>
<td></td>
</tr>
<tr>
<td><strong>3dB Bandwidth</strong></td>
<td>20 Hz</td>
<td></td>
</tr>
</tbody>
</table>

**3-Axis Rate Gyro**

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Update Rate</strong></td>
<td>50Hz</td>
<td></td>
</tr>
<tr>
<td><strong>Dynamic Range</strong></td>
<td>±300 °/sec</td>
<td></td>
</tr>
<tr>
<td><strong>Non-Linearity</strong></td>
<td>0.1% of FS</td>
<td></td>
</tr>
<tr>
<td><strong>Noise Density</strong></td>
<td>0.1 °/sec / √Hz</td>
<td></td>
</tr>
<tr>
<td><strong>3dB Bandwidth</strong></td>
<td>20 Hz</td>
<td></td>
</tr>
</tbody>
</table>

**Environment**

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Temperature</strong></td>
<td>-40° C to 85° C, operating and storage</td>
<td></td>
</tr>
<tr>
<td><strong>Humidity</strong></td>
<td>10% to 90% RH, non-condensing</td>
<td></td>
</tr>
<tr>
<td><strong>Survival Shock</strong></td>
<td>100 g, 8ms., ½ sine</td>
<td></td>
</tr>
<tr>
<td><strong>Survival Vibration</strong></td>
<td>6 grms, 10 Hz to 2000 Hz, random</td>
<td></td>
</tr>
</tbody>
</table>
3.4.4 RC Transmitter. The Futaba T7-C remote control unit is used as the RC interface to the autopilot due to the availability of PPM signal generation and interfaces. This unit has also seven channels output, see Figure 3.10.

Figure 3.10: Futaba T-7C RC unit
Chapter 4: Linearization and Control

4.1 Introduction

The linearization of the equations of motion around the equilibrium points results with the stability derivatives of the helicopter in a linear representation of the helicopter nonlinear model. To obtain the linear model derived from the helicopter nonlinear model, a Matlab linearization tool ‘linmod’ is applied to generate a state space model. This linear model is obtained for the hover equilibrium state. The resulting linear model will be used to design the PI controllers. The Matlab linearization codes were obtained following Miranda [7] and modified to fit Maxi-Joker-3 model.

4.2 Trim conditions and linearization

The trim condition of the equations of motion is obtained for hover state. The inputs that maintain the hover conditions are the steady state trim values of the inputs. The trim control inputs are:

\[ u_e = [u_{lon}^e \quad u_{lat}^e \quad u_{col}^e \quad u_{ped}^e] \]

which maintain the helicopter at required states:

\[ x_e = [u \quad v \quad w \quad \phi \quad \theta \quad \psi \quad p \quad q \quad r] \]

For trimmed flight the rate of change of the helicopter’s state vector must be maintained at zero.

\[ \dot{x}_e = 0 \]

The small perturbation theory is then applied to obtain the helicopter behavior if the aircraft is perturbed from this state by either the control inputs or other disturbances [11]:

\[ x = x_e + \delta x \]

The small perturbation theory can be applied to estimate the helicopter states about trim operating point, once the trimmed states and the control inputs are determined. The operating point for the hovering flight condition is when all the states are zero:

\[ x_e = [u \quad v \quad w \quad \phi \quad \theta \quad \psi \quad p \quad q \quad r] = 0 \]

The state space model is obtained after trimming the helicopter using Matlab command ‘trim’ and then obtaining the model using ‘linmod’ function. The resulting state-space system is:
\[\ddot{x} = A\dot{x} + B\ddot{u}\] 4-4
\[\ddot{y} = C\dot{x} + D\ddot{u}\] 4-5

4.3 The characteristic helicopter modes

For stability analysis, helicopter dynamics can be considered to include a linear combination of characteristic modes, each having its frequency, damping, and response characteristics. The linear estimate that permits this analysis is essential for physical understanding of the complicated dynamics in disturbed flight [27]. The helicopter motion can be described by:

\[\ddot{x} = A\dot{x} + B\ddot{u}\] 4-6

The dynamic model involves 10 states which are the linear and angular velocities, vehicle attitudes, and cyclic longitudinal and lateral flapping. The system dynamics matrix \(A\) in equation 4-6 resulting from the linearization about the hover condition is as follows:

\[
\begin{bmatrix}
\dot{u} \\
\dot{w} \\
\dot{q} \\
\dot{\theta} \\
\dot{\beta_{1c}} \\
\dot{v} \\
\dot{\rho} \\
\dot{r} \\
\dot{\phi} \\
\end{bmatrix}
= \begin{bmatrix}
-0.019 & 0.0005 & 0.0736 & -9.81 & -9.732 & 0.1384 & 0 & 0 & 0 & 0 \\
0.0009 & -3.283 & -0.1038 & 0.0129 & -0.0014 & -0.5478 & 0.1179 & 0 & -0.9887 & 0.05177 \\
0 & 0.002 & 0 & 0 & 178 & 0 & 0.0464 & 0.184 & 0 & 0 \\
0 & 0 & 0.9949 & 0 & 0 & 0 & 0 & 0 & -0.1008 & -0.1377 \\
0 & 0 & -1 & 0 & 0 & -8.85 & 0 & 0 & 0 & 0 \\
-0.1384 & 0.5662 & 0 & 0.0013 & 0 & -0.025 & -0.0736 & 0.1038 & 9.76 & 9.732 \\
0 & 0.18730 & 0.0974 & 0 & 0.0001 & 0 & 0 & 0 & 0 & 665.1 \\
0 & -2.667 & -0.4613 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & -0.0001 & 0.1377 & 0 & 0 & 1 & -0.0013 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & -1 & 0 & 0 & 0 & -9.891 \\
\end{bmatrix}
\begin{bmatrix}
\dot{u} \\
\dot{w} \\
\dot{q} \\
\dot{\theta} \\
\dot{\beta_{1c}} \\
\dot{v} \\
\dot{\rho} \\
\dot{r} \\
\dot{\phi} \\
\end{bmatrix}
\]

The eigenvalues and eigenvectors of the system matrix \(A\) were calculated. The obtained eigenvalues show that the helicopter has both complex eigenvalues indicating modes with an oscillatory motion. The stability of the helicopter can be determined by the signs of the real parts. The eigenvalues with a positive real part signify instability, wherein a negative real part shows stability [27]. The damping factor and undamped natural frequency for \(A\) matrix eigenvalues is shown in Table 4.1.
Table 4.1: Damping factor and undamped natural frequency for the helicopter eigenvalues in hover.

<table>
<thead>
<tr>
<th>Eigenvalue</th>
<th>Damping</th>
<th>Freq. (rad/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>-4.9456±25.3104i</td>
<td>0.19</td>
<td>25.80</td>
</tr>
<tr>
<td>-4.4227 ±12.5906</td>
<td>0.31</td>
<td>13.30</td>
</tr>
<tr>
<td>-3.17</td>
<td>1</td>
<td>3.17</td>
</tr>
<tr>
<td>-0.4497 ± 0.4285i</td>
<td>0.72</td>
<td>0.621</td>
</tr>
<tr>
<td>0.3754 ± 0.3929i</td>
<td>-0.69</td>
<td>0.543</td>
</tr>
<tr>
<td>0.0089</td>
<td>1</td>
<td>0.0089</td>
</tr>
</tbody>
</table>

There are two eigenvalues with positive real parts which define the instability behavior of the helicopter. Table 4.2 shows the eigenvectors corresponding to the above eigenvalues. The eigenvectors show the dominant dynamics associated with each eigenvalue. Also, Figure 4.1. displays polar plots for the eigenvectors at hover flight regime.

Padfield [11] states that eigenmodes can be separated into longitudinal and lateral motions in spite of the fact that they are coupled. Also, it can be noticed that at hover flight regime the shaded values (see Table 4.2) contribute largely to its corresponding state which means the system has weak coupling between longitudinal and lateral-directional mode. Therefore, it can be decoupled into longitudinal- and lateral-directional subsystems. Hence, A Matrix consists from [28]:

\[
A = \begin{bmatrix}
A_{\text{long-ver}} & A_{(\text{lat-dir})to(\text{long-ver})} \\
A_{(\text{long-ver})to(\text{lat-dir})} & A_{\text{lat-dir}}
\end{bmatrix}
\]  

4-7
Table 4.2: Characteristics for each eigenvalue at hover

<table>
<thead>
<tr>
<th>( \lambda )</th>
<th>(-4.9456 \pm 25.3104i)</th>
<th>(-4.4227 \pm 12.5906)</th>
<th>(-3.173)</th>
<th>(-0.4497 \pm 0.4285i)</th>
<th>(0.3754 \pm 0.3929i)</th>
<th>(-0.0089)</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>M</strong></td>
<td>Mag</td>
<td>Phase</td>
<td>Mag</td>
<td>Phase</td>
<td>Mag</td>
<td>Phase</td>
</tr>
<tr>
<td>( \text{u} )</td>
<td>0.0001</td>
<td>42.02</td>
<td>0.0416</td>
<td>-109.616</td>
<td>0.0862</td>
<td>0.8489</td>
</tr>
<tr>
<td>( \text{w} )</td>
<td>0.0051</td>
<td>-77.8867</td>
<td>0.0082</td>
<td>85.3641</td>
<td>0.7537</td>
<td>0</td>
</tr>
<tr>
<td>( \text{q} )</td>
<td>0.0025</td>
<td>-101.915</td>
<td>0.993</td>
<td>0</td>
<td>0.0042</td>
<td>0</td>
</tr>
<tr>
<td>( \text{( \theta )} )</td>
<td>0.0001</td>
<td>-20.2591</td>
<td>0.0740</td>
<td>-109.551</td>
<td>0.0188</td>
<td>0</td>
</tr>
<tr>
<td>( \text{( \beta_{1c} )} )</td>
<td>0.0001</td>
<td>-3.1469</td>
<td>0.0744</td>
<td>109.3715</td>
<td>0.0007</td>
<td>180</td>
</tr>
<tr>
<td>( \text{( \nu )} )</td>
<td>0.0084</td>
<td>80.0239</td>
<td>0.0013</td>
<td>-8.8088</td>
<td>0.1488</td>
<td>180</td>
</tr>
<tr>
<td>( \text{( \rho )} )</td>
<td>0.9984</td>
<td>0</td>
<td>0.0028</td>
<td>65.8675</td>
<td>0.0015</td>
<td>0</td>
</tr>
<tr>
<td>( \text{( \rho )} )</td>
<td>0.0006</td>
<td>-0.7323</td>
<td>0.0345</td>
<td>73.3537</td>
<td>0.634</td>
<td>0</td>
</tr>
<tr>
<td>( \text{( \phi )} )</td>
<td>0.0387</td>
<td>-101.057</td>
<td>0.0006</td>
<td>141.8374</td>
<td>0.001</td>
<td>180</td>
</tr>
<tr>
<td>( \text{( \beta_{1s} )} )</td>
<td>0.0387</td>
<td>101.0561</td>
<td>0.0002</td>
<td>179.3442</td>
<td>0.0002</td>
<td>180</td>
</tr>
</tbody>
</table>
4.4 Longitudinal-Vertical Motion

The longitudinal-vertical motion of a helicopter consists of the main rotor longitudinal flapping $\beta_{1c}$, pitching ($\theta$), pitching rate ($q$), forward velocity ($u$), and heave motion ($w$). The longitudinal stability is determined by the main rotor. The characteristic equation resolves into pairs of complex conjugate roots demonstrating
two oscillatory dynamics, one of short period and high damping and the other of long period (phugoid) which is fairly damped. Even though helicopters do not exhibit the short period and phugoid motions defined for fixed wing aircraft, there are certain similarities that can be drawn [27].

### 4.5 Lateral-Directional Motion

Yaw and yaw rate ($\psi, r$), roll and roll rate ($\phi, p$), lateral flapping ($\beta_1$), and lateral velocity ($v$) create the Lateral-Directional Motion of the helicopter. Lateral-Directional dynamics is largely influenced by the main rotor and the tail rotor aerodynamic because of helicopter geometrical asymmetry.

The Dutch roll (Lateral / Directional Oscillation) is an oscillation in roll and yaw. In hover the helicopter oscillation becomes more like the Dutch roll of a fixed wing aircraft while in forward flight the directional stability of the helicopter increases. Table 4.3 summarizes all eigenmode motion with its corresponding eigenvalue.

<table>
<thead>
<tr>
<th>Eigenvalue</th>
<th>Damping</th>
<th>Freq. (rad/s)</th>
<th>Mode description for hover condition</th>
</tr>
</thead>
<tbody>
<tr>
<td>-4.9456±25.3104i</td>
<td>0.19</td>
<td>25.80</td>
<td>Dutch roll</td>
</tr>
<tr>
<td>-4.4227 ± 12.5906</td>
<td>0.31</td>
<td>13.30</td>
<td>Short period pitching oscillation</td>
</tr>
<tr>
<td>-3.17</td>
<td>1</td>
<td>3.17</td>
<td>Damped yaw heave mode</td>
</tr>
<tr>
<td>-0.4497 ± 0.4285i</td>
<td>0.72</td>
<td>0.621</td>
<td>Stable phugoid</td>
</tr>
<tr>
<td>0.3754 ± 0.3929i</td>
<td>-0.69</td>
<td>0.543</td>
<td>Unstable phugoid</td>
</tr>
<tr>
<td>-0.0089</td>
<td>1</td>
<td>0.0089</td>
<td>Forward velocity mode</td>
</tr>
</tbody>
</table>

### 4.6 The Differences between a Flybar Helicopter and a Flybarless Helicopter

It is important to study the differences between a flybar model helicopter and a flybarless model helicopter. First of all, a model helicopter has a fast response because of its small size. Hence, without engaging an additional stability augmentation device (the flybar), it would be really hard for a human pilot to control it. A flybar (Bell-Hiller paddles) with an airfoil is not used currently, however, it is replaced with electronic stabilizer to improve the stability characteristic around the pitch and roll axes.

A control augmentation to the main rotor cyclic input can be created by using the stabilizer bar to control the flapping motion. This augmentation is employed by the Bell mixing mechanism; the aerodynamic paddles of the flybar give the gyro damping
and perform as a servomechanism for the control input (Figure 4.2). From a control viewpoint, this can be understood as a rate damper feedback in the pitch and roll loops. The rate damper feedback also reduces the response of the aircraft to wind gusts.

![Swash Plate Position](image)

**Figure 4.2: Flybar Mechanism [12]**

The Lateral/Directional Oscillation (Dutch Roll) for a flybar helicopter is well damped and stable (Table 4.4). For a flybarless helicopter the eigenvalues are less damped and therefore less stable as shown in Figure 4.3 (b). Figure 4.3 shows comparison between flybar and flybarless helicopters. It can be seen that both systems are unstable but the flybar system (Figure 4.3 (a)) shows more stable poles; far away from the origin. Therefore, it is more stable and easier to control.

**Table 4.4: Modes descriptions at hover (flybar helicopter)**

<table>
<thead>
<tr>
<th>Eigenvalue</th>
<th>Damping</th>
<th>Freq. (rad/s)</th>
<th>Mode description for hover condition</th>
</tr>
</thead>
<tbody>
<tr>
<td>-30.5±15.5i</td>
<td>0.891</td>
<td>34.2</td>
<td>Dutch roll</td>
</tr>
<tr>
<td>-15.6±11.5i</td>
<td>0.803</td>
<td>19.4</td>
<td>Short period pitching oscillation</td>
</tr>
<tr>
<td>-3.24</td>
<td>1</td>
<td>3.24</td>
<td>Damped yaw heave mode</td>
</tr>
<tr>
<td>-0.351±0.357i</td>
<td>0.701</td>
<td>0.501</td>
<td>Stable phugoid</td>
</tr>
<tr>
<td>0.311±0.337i</td>
<td>-0.678</td>
<td>0.458</td>
<td>Unstable phugoid</td>
</tr>
<tr>
<td>-0.00846</td>
<td>1</td>
<td>0.00846</td>
<td>Forward velocity mode</td>
</tr>
</tbody>
</table>
Figure 4.3: Eigenvalues Plots: (a) Flybar Helicopter, (b) Flybarless Helicopter
Figure 4.4: Eigenvalues polar plots at hover flight regime (flybar helicopter)

4.7 Controller Design

Once the helicopter model is established, control system design is made using multiple-input multiple-output (MIMO) system to simplify the analysis. Then, it is important to control the helicopter’s Euler angles and to establish a stability augmentation system. Controlling the helicopter’s motion in the x and y-directions is achieved through the control of the helicopter’s angular orientation (attitude angles).
Therefore, hover condition control design of the decoupled systems, i.e., longitudinal and lateral models will be discussed in more details in the next sections.

4.7.1 Longitudinal-Vertical Controller Design. The longitudinal-vertical decoupled state-space model is shown in Figure 4.5. It is used in this chapter for the gain tuning based on the linear helicopter model.

![Diagram of longitudinal-vertical state-space model](image)

Figure 4.5: Longitudinal-vertical state-space model

The flowing longitudinal-vertical state-space model is used to control the pitch $(\theta, q)$ using pitch damper for the inner loop and pitch hold for the outer loop.

\[
\begin{bmatrix}
\dot{u} \\
\dot{w} \\
\dot{q} \\
\dot{\theta}
\end{bmatrix} = \begin{bmatrix}
-0.019 & 0.0005 & 0.0736 & -9.81 & -9.732 \\
0.0009 & -3.283 & -0.1038 & 0.0129 & -0.0014 \\
0 & 0.002 & 0 & 0 & 178 \\
0 & 0 & 0.9949 & 0 & 0
\end{bmatrix}
\begin{bmatrix}
u \\
w \\
q \\
\theta
\end{bmatrix} + \begin{bmatrix}
0.04215 & 0 \\
-294.6 & 0 \\
0.1764 & 0 \\
0 & 0
\end{bmatrix}
\begin{bmatrix}
u_{col} \\
u_{long}
\end{bmatrix}

Figure 4.6 shows the step response of pitch damper loop. The performance of the control loop shows 48% overshoot, settling time after 1.76 seconds, and no steady state error. This performance guarantees the inner loop is faster five times than the outer loop.
After tuning the Pitch damper loop, the outer hold loop is created as shown in Figure 4.5. The step response of this loop is shown in Figure 4.7 with 14.3% overshoot, 1.58 seconds settling time, and no steady state error.

Pitch Hold Loop

After tuning the pitch loop, heave dynamics controller is started, by controlling the velocity in z-direction and then adding the altitude loop to the decoupled system. The step response of the z-velocity loop shows good performance of 12.9% overshoot, 1.37 seconds settling time, and no steady state error.
The altitude hold loop response is shown Figure 4.9 with 15.4 % overshoot, 4.51 seconds settling time, and no steady state error.

Table 4.5 shows the summary of the tuned gains for longitudinal-vertical controller design system.
Table 4.5: Longitudinal-Vertical Controller Gain Values

<table>
<thead>
<tr>
<th>Loop</th>
<th>Kp</th>
<th>Ki</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pitch Damper</td>
<td>0.0096</td>
<td>0.208</td>
</tr>
<tr>
<td>Pitch Hold</td>
<td>2.83</td>
<td>0.126</td>
</tr>
<tr>
<td>Z – Velocity</td>
<td>-0.059</td>
<td>-0.108</td>
</tr>
<tr>
<td>Altitude hold</td>
<td>2.28</td>
<td>0.942</td>
</tr>
</tbody>
</table>

4.7.2 Lateral-directional Model Controller Design.

Applying the same approach to design the lateral-directional controller:

Figure 4.10: Lateral-directional state-space model

Lateral-directional state space model can be written as follow:

\[
\begin{bmatrix}
\dot{\phi} \\
\dot{p} \\
\dot{r} \\
\dot{\beta}_{ls}
\end{bmatrix} =
\begin{bmatrix}
-0.025 & -0.0736 & 0.1038 & 9.77 & 9.732 \\
0 & 0 & 0 & 0 & 665.1 \\
0 & 0 & 0 & 0 & 0 \\
0 & 1 & -0.0013 & 0 & 0
\end{bmatrix}
\begin{bmatrix}
\dot{v} \\
p \\
r \\
\phi
\end{bmatrix} +
\begin{bmatrix}
-17.33 & 0 \\
-60.27 & 0 \\
444.5 & 0 \\
0 & 0
\end{bmatrix}
\begin{bmatrix}
\nu_{Ped} \\
\nu_{Lat}
\end{bmatrix}
\]
As shown in Figure 4.11, the tuned roll damper response demonstrates 32.9% overshoot, 1.23 seconds settling time, and no steady state error. The inner loop’s frequency is 6.28 rad/sec which is five times faster than the roll hold loop.

Roll Damper Loop

![Roll Damper Response Graph](image)

Figure 4.11: Roll damper response

Figure 4.12 shows the roll hold loop with 12.2% overshoot, 4.15 seconds settling time, and frequency of 1.21 rad/sec.

Roll Hold Loop

![Roll Hold Response Graph](image)

Figure 4.12: Roll hold response

Heading hold and yaw damper loops are tuned where the innerloop frequency is 18.80 rad/sec and the outerloop is 2.7739 rad/sec. Figure 4.13 shows the yaw damper step response with 34.8% overshoot, and 0.5 sec settling time. The inner loop is tuned
in such a way that it has to be faster than the outer loop by five times which explains why it has an excessive overshoot.

**Figure 4.13: Yaw damper response**

Figure 4.14 shows the heading hold step response with 21.6% overshoot, 2.71 sec settling time, and no steady state error.

**Figure 4.14: Heading hold response**
Table 4.6 shows the summary of the final gain values for lateral-directional controller.

<table>
<thead>
<tr>
<th>Loop</th>
<th>Kp</th>
<th>Ki</th>
</tr>
</thead>
<tbody>
<tr>
<td>Roll Damper</td>
<td>0.00241</td>
<td>0.241</td>
</tr>
<tr>
<td>Roll Hold</td>
<td>2.167</td>
<td>0.520</td>
</tr>
<tr>
<td>Yaw Damper</td>
<td>0.037</td>
<td>0.011</td>
</tr>
<tr>
<td>Heading Hold</td>
<td>3.99</td>
<td>0.75</td>
</tr>
</tbody>
</table>

**4.7.3 Full System Controller Design.** After designing the controller for the decoupled systems, the same gains are applied to the full system resulting with the same performance, as shown in Figure 4.15.

![Figure 4.15: Full System](image-url)
Chapter 5: Experimental Results and Discussion

Hardware-in-the-loop simulation (HILS) is an important tool in the development of a real-time control system. It permits the engineers to develop the flight control system software in conditions close to a real flight.

Figure 5.1 presents a schematic diagram of the HILS. In the HILS system setup the dSpace dsp processor performs real time a simulation of Helicopter 6-DOF UAV Model. The navigation INS/IMU and GPS sensors and the control surfaces- actuators are also modeled and simulated. The control commands provided by the pilot are delivered via the servo board. The control logic, on the MPC555 microcontroller, generates servo commands based on the pilot commands and states inputs from the dSpace computer, and sends them back to the servo board. The ground station computer also sends commands, and receives the helicopters attitudes and displays the helicopter’s positions on the map.

5.1 Hardware in the Loop Simulation Results

The simulated helicopter model, which has been developed using Matlab/Simulink toolbox, is run on the dSpace real time system. This device provides compatibility to download the helicopter model. The hover controller, implemented in an MPC555 embedded controller, is linked with the dSpace performing the helicopter UAV model to send the aircraft stats, and receive the servo commands. An MPC 555
embedded system is utilized as the flight control computer, in which the autopilot control laws and gains are embedded.

Figure 5.2 shows the Euler angles and rates received form the dSPACE platform, when commanding the pitch angle 10 deg the pitch rate changes accordingly. The inner pitch loop returns it to the steady state and stabilizing the helicopter while the outer pitch loop controls the pitch angle. It can be shown that the roll angle does not change when introducing pitch command proving that there is a weak coupling between pitch and roll at hover flight.

![Pitch Data](image1)

![Pitch Rate Data](image2)

![Roll Data](image3)

![Roll Rate Data](image4)

Figure 5.2: Euler angles and rates from HILS

Figure 5.3 shows the yaw angle when introducing 20 deg. clockwise command, the actual yaw overshoots by around 25%. However, it overshoots more than 50% when commanding 20 deg. counterclockwise. This behavior explains the oscillations while introducing clockwise command to the tail rotor. That is because of the main rotor and
the heading directions are spinning together clockwise, which adds extra angular moment to the system. In contrast when commanding counterclockwise, the angular momentum is subtracted, therefore more stable system. The yaw rate response changes when commanding the rudder and returns to zero after reaching the desired heading value, see Figure 5.3.

![Figure 5.3: Yaw angle and rate response from HILS](image)

Figure 5.3 shows the altitude hold and the Z velocity response. It can be seen the actual altitude reaches its desired value while introducing ramp command to 20 m. The Z velocity inner loop tries to keep the Z velocity zero when reaching the desired altitude.
After tuning and verifying the nonlinear model in the HILS, the next step is to implement the hover controller on the Rotary Wing Mazari Autopilot. The testing procedure started on the ground by examining if the controller delivers the right command to the servo when pitching and rolling, then the platform has been tested on the test stand to check the flight control for proper operation. Finally, a flight test is arranged.

Figure 5.6 shows the pitch angle data and its corresponding rate, it can be shown that the time delay between the desired input and the actual output is 0.2 sec while the helicopters follows the pilot’s command exactly with almost zero steady state error. Also, the roll angle and rate are shown in Figure 5.6 in which it can be seen that three is 0.2 sec time delay and almost zero steady state error. The roll rate data reading can
be improved by modifying the IMU location which is 20 cm below the CG which means less vibration on the sensors and better performance.

Figure 5.5: Euler angles and rates from flight test

Figure 5.6 shows the real flight data test for the yaw loop, it can be seen that rudder overshoots when commanding it clockwise. Also, the yaw damper controller tries to keep the yaw rate zero.

Figure 5.7 shows the real flight data test for the altitude hold loop. The altitude is commanded from 10 m to 30 m at a rate of 1 m/s. The controller shows around 0.5 m steady state error, and around 1.2 m overshoot. The inner loop, which controls the Z body velocity, shows very good steady state and transient performance by keeping the velocity zero at the desired altitude.
Figure 5.6: Yaw angle and rate response from flight test

Figure 5.7: Altitude and Z Body Velocity data from flight test
5.3 MicroPilot Flight Test Results

MicroPilot autopilot has been integrated into the helicopter, configured, tuned and tested. Figure 5.8 shows the pitch data during autonomous take-off, it demonstrates 0.4 sec delay between the actual and the desired pitch.

Figure 5.8: MicroPilot pitch data

Figure 5.9 presents roll data during autonomous take off, it demonstrates 0.2 sec delay between the actual and the desired roll.

Figure 5.9: MicroPilot roll data
Figure 5.10 shows MicroPilot yaw data while turning to the next waypoint, the actual yaw follows the desired yaw but the helicopter is waggling slowly.

![Figure 5.10: MicroPilot yaw data]

Figure 5.10 : MicroPilot yaw data

Figure 5.11 represents the X body velocity data logged form the Micropilot autopilot which shows almost zero steady state error, and 15 % overshoot.

![Figure 5.11: MicroPilot X Body Velocity data]

Figure 5.11 : MicroPilot X Body Velocity data

Figure 5.12 demonstrates the Y body velocity, it can be seen that the controller is doing very well by keeping the UAV on its track with a maximum desired velocity of 0.25m/s.
After finishing the inner and outer PID loop tuning, autonomous take off for 30m and then landing was performed successfully. Finally, the last mission was planned for full autonomous test including auto takeoff and landing, and the waypoints navigation. Waypoints were selected to fly total distance of around 640 m. The mission was finished very successfully with a total fight time of around 5 minutes and 30 seconds, as shown in Figure 5.13.
Figure 5.13: Waypoints following experiment conducted using AUS-UAV platform in Hamreyya free zone, Sharjah, UAE (square pattern)
Chapter 6 : Conclusion and Recommendation

In this report we have presented the research and development of the first rotary wing UAV platform in the academic institutions in the region. The work included successful integration and flight testing of COTS autopilot (Micropilot system) to a model scale helicopter. During this work, numerous challenges and know-how issues were faced. Issues like vibration isolation, bugs in the system resulting in significant delays and the need to send the system back to the manufacturer three times during the course of integration. Finally, the system was made operational and a complete mission was flown covering more than 2km mission involving waypoint navigation and autonomous takeoff and landing.

The second part of this thesis which includes developing a mathematical model for the helicopter (flybarless Maxi-Jocker 3) was accomplished. This model was used to develop attitude flight control system using hardware in the loop simulation rapid prototyping system. The flight control laws were successfully coded onto a 32 bit microcontroller and appropriate modifications to the ground control station were made. Finally, the developed attitude autopilot was test flown and performed as expected.

Last part of this work is to benchmark the attitude controllers developed in this thesis against the COTS autopilot (Micropilot) performance. This demonstrates the successful achievement of the main goal of this thesis is the use of the Micropilot system which resulted in benchmarking our development activities.

6.1 Conclusion

- The MicroPilot autopilot MP2128\textsuperscript{LRC} has been integrated, test flown, and evaluated after solving many problems and issues, such as system failure during the flight due to MicroPilot’s processor bugs, also bad calibration for z-accelerometers which caused two major crashes. Micropilot autopilot is fully operational and is ready to be used for benchmarking future development of local research activities.
- After completion of the MicroPilot autopilot PID-gains tuning, autonomous takeoff and landing and waypoint navigation was performed successfully and consistently. Flight test procedures and operation were developed and important experience was gained from the COTS integration and operation.
- The flybarless nonlinear helicopter model has been developed. Attitude PID flight control laws were developed based on linearized models and established for the
hover flight condition. The differences between flybarless and flybar model have been discussed in details. The flight control laws were evaluated using HILS in order to speed up the helicopter autopilot development.

- The nonlinear model and the tuned PI gains have been tested in real flight test. The helicopter has shown good flight performance.
- A 6 degrees of freedom test stand was designed and built to help in indoor flight testing and evaluation for both Micropilot integration and own flight control laws development and testing. It proved to be an effective tool to tune the helicopter before the real flight test.
- The attitude autopilot developed in this thesis for the model scale helicopter has shown faster response than the MicroPilot in pitch and almost the same response in the roll. Also, it shows no steady state error in the attitudes while the MicroPilot has around 2 deg steady state errors in the pitch attitude and 0.5 deg in the roll attitude.
  The rudder loop for Mazari autopilot shows smooth and stable heading while the MicroPilot shows waggling rudder with about ± 5 deg.

6.2 Future Work and Recommendation

The development of Rotary Wing Mazari, capable of navigating and maneuvering through complicated surroundings, is an exciting prospect. The future work recommended here uses the present work as a springboard for achieving autonomous, robust takeoff and waypoint navigation.

First, the full model of the vehicle needs to be improved by modeling gust disturbances and forward flight model which would assist in developing algorithms for trajectory tracking. Also, the system identification test should be planned to get more accurate parameters for the aircraft.

Finally, the test stand needs to be improved by using frictionless roller bearings and using lighter material other than steel which is used for the rolling fork.
References


[34] "http://www.modelairplane.cadblog.net/helicopter_stability_control.htm." [Online].

Appendix A : Technical Data Sheets

- **Futaba S9251 servo**

  Modulation: Digital
  Torque: 4.8V: 51.0 oz-in (3.67 kg-cm)
  Speed: 4.8V: 0.07 sec/60°
  Weight: 2.01 oz (57.0 g)
  Dimensions:
  Length: 1.57 in (39.9 mm)
  Width: 0.79 in (20.1 mm)
  Height: 1.46 in (37.1 mm)
  Motor Type: Coreless
  Gear Type: Metal
  Rotation/Support: Dual Bearings

- **Futaba BLS452 servo**

  Modulation: Digital
  Torque: 4.8V: 156.0 oz-in (11.23 kg-cm)
  6.0V: 194.0 oz-in (13.97 kg-cm)
  Speed: 4.8V: 0.18 sec/60°
  6.0V: 0.14 sec/60°
  Weight: 2.05 oz (58.0 g)
  Dimensions:
  Length: 1.57 in (39.9 mm)
  Width: 0.79 in (20.1 mm)
  Height: 1.46 in (37.1 mm)
  Motor Type: Brushless
  Gear Type: Metal
### MICROBEAST Overview

### SETUP MENU

(Menu LED is steady ON)

<table>
<thead>
<tr>
<th>Status-LEDs</th>
<th>off</th>
<th>purple</th>
<th>red flashing</th>
<th>red</th>
<th>blue flashing</th>
<th>blue</th>
</tr>
</thead>
<tbody>
<tr>
<td>A Mounting orientation</td>
<td>upright (vertical)</td>
<td>flat (horizontal)*</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>B Swashplate servo - frequency</td>
<td>user defined</td>
<td>50 Hz*</td>
<td>65 Hz</td>
<td>120 Hz</td>
<td>165 Hz</td>
<td>200 Hz</td>
</tr>
<tr>
<td>C Tail servo - center position pulse-length</td>
<td>user defined</td>
<td>960 µs</td>
<td>760 µs</td>
<td>1520 µs*</td>
<td></td>
<td></td>
</tr>
<tr>
<td>D Tail servo - frequency</td>
<td>user defined</td>
<td>50 Hz*</td>
<td>165 Hz</td>
<td>270 Hz</td>
<td>333 Hz</td>
<td>560 Hz</td>
</tr>
<tr>
<td>E Tail servo - rotor endpoints</td>
<td>tail stick - move to right endpoint and wait / left endpoint and wait</td>
<td></td>
<td></td>
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<td></td>
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</tr>
<tr>
<td>F Tail - sensor direction</td>
<td>normal*</td>
<td>reversed</td>
<td></td>
<td></td>
<td></td>
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</tr>
<tr>
<td>G Swashplate - servo centering</td>
<td>reference position</td>
<td>CH1 center pos.</td>
<td>CH2 center pos.</td>
<td>CH3 center pos.</td>
<td></td>
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</tr>
<tr>
<td>H Swashplate - mixer</td>
<td>user defined</td>
<td>mechanical</td>
<td>90°</td>
<td>120°*</td>
<td>140°</td>
<td>140° (T=1)</td>
</tr>
<tr>
<td>I Swashplate - servo directions</td>
<td>nor</td>
<td>rev</td>
<td>nor</td>
<td>nor</td>
<td>rev</td>
<td>nor</td>
</tr>
<tr>
<td>J Swashplate - cyclic pitch geometry</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
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<tr>
<td>K Collective pitch range</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
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</tr>
<tr>
<td>L Swashplate - cyclic limit</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td></td>
</tr>
<tr>
<td>M Swashplate - sensor directions</td>
<td>rev</td>
<td>nor</td>
<td>rev</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
</tr>
<tr>
<td>N Pirouette optimization direction</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
<td>nor</td>
</tr>
</tbody>
</table>

- MICROBEAST 3 axis gyro
Vita

Ahmad Alshoubaki was born in Saudi Arabia. He grew up in Jordan and was educated in local public schools and graduated from 2\textsuperscript{nd} Senior High School in Irbid, Jordan 2005. He graduated from Jordan University of Science and Technology in 2009, with a Bachelor’s degree in Mechanical Engineering. After his graduation, he worked as a research intern at Technische Universität Bergakademie Freiberg in Germany.

Ahmad moved to the United Arab Emirates in 2010 and worked as a graduate teaching assistant for two years at the American University of Sharjah. During the same period, he began the Master’s program in Mechanical Engineering at the same university.