System identification and control system design for a model helicopter in hover

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SYSTEM IDENTIFICATION AND CONTROL SYSTEM DESIGN FOR A MODEL HELICOPTER IN HOVER

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2005

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SCHOOL OF MECHANICAL AND AEROSPACE ENGINEERING

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Abstract

Mini-size UAV development has garnered much research interest in the past few years as they can be easily deployed in the battlefield and the cost is much lower as compared to the bigger size UAV. The aim of this research project is to gain a comprehensive insight into the development of a small UAV system by going through a developmental cycle involving the hardware and software implementation, integration and testing. This research was done using a small radio-controlled model helicopter because it is one of the more challenging aerial vehicle platforms due to its complex flight dynamics.

The design and operation of the helicopter used were studied to establish a better understanding on the platform and this was correlated to the study of the flight dynamics of the helicopter. The 6 DOF state-space equation of the helicopter flight dynamics was subsequently derived. This dynamic model is essential for the system identification process and flight control system design in the later phase.

The hardware system, consisting of inertia sensors, GPS, wireless communication devices and industrial computers, were integrated to the helicopter. Software programs were written to the onboard computers so that the system can collect real-time data for both the system identification process as well as sensor feedback data for controller implementation.

Flight tests were conducted to collect data for the system identification. Time domain parametric identification was carried out to identify the parameters in the state-space equation. The identified model was validated using different sets of flight test data that were not used in the identification process. After obtaining the identified model, flight controllers were designed to attain a stabilized hover flight mode. A classical/modern hybrid controller was used to meet the hover flight requirements.

Acknowledgements

I would like to express my deepest gratitude to my company, DSO National Laboratories, for starting this research initiative back in 1999 when I started working on it as a pioneer member as part of my final year project during my undergraduate study. Over the years, the company has given great financial support for this project as well as given me the opportunity to pursue this research as a part-time candidate in NTU. Special thanks to the project manager from DSO National Laboratories, Peter Seah, who has been very supportive in this project over the years.

I am grateful to my project supervisor, Associate Professor Eicher Low for his guidance, help and advice since my undergraduate days. His help and support over the years has been great.

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I would also like to thank my colleague, Zhou Min from DSO National Laboratories who has been helping me to implement the software coding required for the onboard computer system.

Lastly, I am grateful to the test pilot, Walter Lee, for his valuable time, advice and experience shared in this project. Thanks to his professional flying skill that save the helicopter from crashing from time to time during the flight test.

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List of Symbols

ϕ	: roll angle
Ψ	: blade azimuth
γ	: blade lock number
Θ	: blade pitch angle
$\mathbf{a}$	: longitudinal rotor blade flapping angle
A	:state-space system matrix
$\mathbf{a}_0$	: blade coning angle
A_{lon}	: longitudinal stick to cyclic pitch ratio
$\mathbf{b}$	: lateral rotor blade flapping angle
B	:state-space input matrix
B_{lat}	: lateral stick to cyclic pitch ratio
$\mathbf{F}$	: total external force vector acting on helicopter C.G.
g	: gravitational acceleration
h	:distance between rotor hub and fuselage C.G.
$\mathbf{I}$	: moment of inertia of helicopter
I_β	: blade moment of inertia about the flapping hinge
I_{xx}, I_{yy}, I_{zz}	: mass moment of inertia in x, y and z direction
K	:LQR gain matrix
K_β	: flapping hinge restraint
K_D	: derivative gain
K_I	: integral gain
K_P	: proportional gain
L, M, N	: moment in x, y and z direction
$\mathbf{M}$	: external moment vector acting on helicopter C.G.
m	: mass of helicopter

List of Symbols

p	: roll rate
q	: pitch rate
Q	: weighting matrix of system states
R	: weighting matrix of input states
r	: yaw rate
T	: main rotor thrust
$\boldsymbol{u}$	: control input vector
u, v, w	: body velocity in x, y and z direction
δ_{col}	: collective control input
δ_{lat}	: cyclic lateral control input
δ_{lon}	: cyclic longitudinal control input
δ_{ped}	: pedal control input
$\boldsymbol{x}$	: states vector
X, Y, Z	: body force in x, y and z direction
β	: rotor blade flapping angle
θ	: pitch angle
τ_f	: rotor time constant
ψ	: heading angle
Ω	: angular velocity of rotor
$\boldsymbol{\Omega}$	: angular velocity vector
$\boldsymbol{v}^I$	: linear velocity vector in inertia reference frame
$\boldsymbol{\omega}^I$	: angular velocity vector in inertia reference frame
$\boldsymbol{v}$	: linear velocity vector in body-fixed reference frame
$\boldsymbol{\omega}$	: angular velocity vector in body-fixed reference frame

Chapter 1

Introduction

1.1 Introduction

Unmanned Aerial Vehicle (UAV) is widely used worldwide for a broad range of applications. The UAV can be employed in military missions including ground, air and sea surveillance, target acquisition, target designation, communications relay and surface ordnance survey.

Small-size UAV system development has garnered much research interest in the past few years as they can be easily deployed in the battlefield and the cost is much lower as compared to the bigger size UAV. Although micro-UAV has been the focus of many researches, the prototypes developed are still far from being deployed as an operational UAV due to the lack of reliable micro components that are used in these vehicles. In addition, the payload offered by the micro-UAV is in the range of not more than 100 grams and this may not be useful for most real-life applications. With these constraints in the current technology, it is predicted that the micro-UAV will not be in the UAV market within the next 5 years. With the gap between the full-size UAV and the micro-

UAV, there exists a niche market for the mini-size UAV to fill into this gap before the emergence of a useful micro-UAV.

A number of mini-size UAVs has been deployed in recent battlefields such as the Pointer UAV in the Afghanistan war. However, these UAV systems are limited to fixed-wing aircrafts as they are simple in structure, efficient, easy to build and maintain as compared with the rotary-wing aircraft. The most vital reason is that the autopilot design for a fixed-wing UAV is much easier than a rotary-wing aircraft as it has a rather simple symmetric de-coupled flight dynamics.

The rotary-wing aircraft has been desired for certain applications where the unique flight capability of the rotorcraft is required. The rotorcraft can take off and land within a limited space, hover and cruise at a very low speed. This will come in useful for operation of the UAV in the urban environment where it can maneuver between buildings which a fixed-wing aircraft cannot. However, little progress has been made in the automatic control of small-scale unmanned helicopter with a few academic institutes demonstrating simple autonomous flight capabilities such as hover or slow forward flight. The main reason for this limitation is the absence of an accurate mathematical model of small-size helicopter vehicle dynamics that can be used for the analysis and design of flight control system.

1.2 Objective

The project aims to gain a comprehensive insight into the development of a model helicopter for an autonomous flying vehicle. Identification of the helicopter system dynamics will be done through flight test data collected using the system identification

techniques which will help to explore a more thorough understanding of the system dynamics. This will allow for the implementation of a flight control system on the helicopter so that it can perform hover flight autonomously.

1.3 Scope

The scope of the project covers the development process of an autonomous small-scale model helicopter with the use of a commercially off the shelf (COTS) radio-controlled (RC) model helicopter, measurement sensors, industrial computers and communication devices.

The basics of the RC helicopter design and operation are studied so that a better understanding of the model helicopter can be established before the helicopter dynamics can be studied. With an understanding of a simplified helicopter dynamics that is derived and adapted from full-size helicopter theory, parameter model development for the state-space dynamics model is done to describe the small-scale helicopter dynamics. This parametric model developed is used for the system identification process.

The hardware integration of the model helicopter includes measurement sensors, communication devices and an industrial computer. At the same time, customized software programs have been developed for the onboard flight computer and for ground monitoring station so as to fulfill the task of flight data collection for the system identification process and to support the synthesis of the flight controller for autonomous hovering of the helicopter.

System identification process is carried out with the flight test data collected. The system identification process identifies the parameters in the state-space model that describes the helicopter dynamics. Time domain system identification method is used and a framework is developed. The dynamic model is then validated with flight test data.

After the identification of the dynamic model, controller is designed using Matlab Toolboxes for the helicopter to perform autonomous hover. Software simulation is used to test the designed controller.

1.4 Project History and Contribution

This project, funded by DSO National Laboratories, was initiated in June 1999 with a collaboration between DSO National Laboratories, Aeronautical System Program Lab and Nanyang Technological University (NTU), School of Mechanical & Production Engineering. The author started working on an autonomous helicopter for his B.Eng thesis in July 1999 dealing with the implementation of simple sensors and real time data acquisition system for a model helicopter. Contributions to this autonomous helicopter development are made by various students from NTU respectively over the years as follows:

- System identification and modeling of a model helicopter by Roger Lim (B.Eng thesis, 2001)
- System identification and modeling of a model helicopter by Gaurav Tholia (B.Eng thesis, 2002)

- Low cost 3 axis IMU for unmanned helicopter by Wong Chong Kum (B.Eng thesis, 2002)
- Stabilised platform for a pan, tilt video camera for unmanned helicopter by Goh Toh Siang (B.Eng thesis, 2002)
- Sensor for unmanned helicopter landing by Marcus Fong (B.Eng thesis 2002)
- Implementation of RTOS on an unmanned vehicle system by Lim Wei Cheng (Industrial attachment, 2002)
- Fuel monitoring system for autonomous model helicopter by Chng Chen Keong (B.Eng thesis, 2003)
- Undercarriage design for RC model helicopter by Aw Yong Tze Ping (B.Eng thesis, 2003)
- Pan, tilt and zoom camera system for robotics air vehicle by Lim Jui Jing (B.Eng thesis, 2003)
- GUI development for monitoring of Unmanned Robotics System Using Matlab by Huang Bing Jie (Industrial attachment, 2003)

1.5 Organization of Report

This dissertation is organized as follows. Chapter 1 gives the introduction, objective and the scope of the project. Chapter 2 covers literature review of the relevant research that has been done. In Chapter 3, helicopter dynamics, modeling and parameter model development for system identification are addressed. Chapter 4 introduces the hardware, software and system integration works that are carried out on the model helicopter. Chapter 5 covers the detail of system identification and gives the result of

Chapter One

the parametric model. Subsequently in Chapter 6, flight controller design is covered and simulation to test the controller is presented. The conclusion, recommendation and future works are addressed in Chapter 7.

Chapter 2

Literature Review

The development of autonomous helicopter covers a number of important fields of disciplines. As the field covered is very wide, it is grouped into various areas so that relevant literature reviews can be covered to assist the development of the project.

The areas of disciplines that are important to this project are as follows:

1. Modeling of small-scale helicopter dynamics
2. System identification of rotary-wing UAV
3. Hardware and software system integration of small-scale helicopter
4. Flight Controller design of rotary-wing UAV

2.1 Modeling of small-scale helicopter dynamics

The field of helicopter dynamics is very much established for full-size helicopter [6]-[8]. However, in the area of small scale helicopter dynamics, there are very few studies and research that have been done.

The general approach for modeling is to use the first principle of modeling. However, this process is tedious and the model obtained is not very accurate. Weilenmann [9] performed a limited condition first principle modeling from the full-scale helicopter

theory using a rigged small-scale helicopter. A nonlinear differential equation model was obtained under this approach and the model was subsequently linearized about the hovering operating point for linear controller design. This linear model has about 70 parameters in which some of it can be measured easily while others had to be determined using complicated experimental setup, such as carrying out wind tunnel measurement.

Bernard Mettler [10] has done a great deal of work to precisely model and identify the system characteristics of a small-scale unmanned rotorcraft for advanced flight control design. He has developed a simple but effective linear parameterized model for the Yamaha R-50 small-scale helicopter using the system identification tool CIPHER (Comprehensive Identification from Frequency Responses). His current work is on a much smaller-scale model helicopter that performs aggressive maneuver [11].

A novel modeling technique that integrates first-principles and system identification modeling techniques was proposed by La Civita [12]. The result from this method shows a very accurate non-linear model suitable for flight simulations and a linear model adequate for control design.

2.2 System Identification of rotary-wing UAV

System identification is a procedure where the mathematical representation of the dynamics of a system can be extracted from input to output data. The classical theory of system identification can be obtained from Ljung [13]. System identification provides a more accurate and easier way to obtain the dynamic model of rotorcraft for control design application as compared to first principle modeling and this is a strong

motivating factor for the interest in using system identification modeling technique [14].

System identification modeling can be done either in the time-domain or in the frequency-domain. In the time domain system identification approach, software such as the Matlab[®] System Identification Toolbox can be used. There is a variety of estimating models that can be used for the modeling process, such as the ARX (autoregressive) and ARMAX (autoregressive moving average) model. The use of the estimating model is dependent on the suitability of the model structure for a particular modeling problem [15]. Various research projects had been carried out using time-domain system identification techniques to model small-scale model helicopters. In California Institute of Technology (USA), a model helicopter was fitted onto a test stand and was used as the test bed for controller design [16]. The test stand restricts the helicopter to have only angular motion. Data were collected with small input perturbations. Rigid body equations of motion were used to develop the parameterized identification model. The prediction error method (PEM) using discrete time domain was used for the identification of the parameters of a state-space model. In the University of California at Berkeley, the Matlab[®] System Identification Toolbox was used for time-domain system identification of various small-scale model helicopters for controller design [17]. PEM algorithm was used to identify the parameters in the parameterized state-space model proposed by B.Mettler in [10]. In Switzerland, Wecontrol GmbH develops flight computer system for model helicopter using time-domain system identification technique to obtain the dynamics of the helicopter before designing the flight controller for the helicopter [18]. The parameters of the state-space

model are identified using the ARMAX model using the Matlab[®] System Identification Toolbox.

The frequency-domain system identification for rotorcraft creates an impact and awareness when the Army-NASA Rotorcraft Division derived a frequency-domain identification approach in rotorcraft modeling [19]. A special numerical tool was developed for this identification approach, known as the Comprehensive Identification from FrEQUENCY Response (CIFER) and is currently available as a commercial product offered by Symvionics Inc. This approach has been applied to identification of various full-scale helicopters such as Black Hawk [20] and B0-105 [21]. For the small-scale helicopter, Mettler uses the frequency-domain CIFER system identification approach to develop the parametric model [10].

2.3 Hardware and software system integration of small-scale helicopter

UAVs had been developed as early as the first world war and were mostly limited to fixed-wing UAV. However, the first rotary-wing UAV prototype only came out as early as the 1950s due to the complexity of rotary-wing UAV as compared to the fixed-wing UAV. One such early day prototype of the rotary-wing UAV developed is the Gyrodyne QH-50, a co-axial configuration rotor design helicopter that was used in the naval ship as drone target and surveillance.

The field of small-scale rotary wing UAV development started in the academic and research institutions in the early 1990's where most of the helicopters used are adapted from the hobby helicopters. However, to convert these commercially available hobby

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helicopters to autonomous helicopter need navigation sensors and flight computers. These autonomous helicopters developed represent a significant achievement in system design and integration, demonstrating the ability to conceive design and prototype complex autonomous aerospace system.

Draper [22], an autonomous helicopter built by a research team from Massachusetts Institute of Technology and Boston University, demonstrated a hardware and software avionics architecture that support the autonomous air vehicle operation and cooperation. The Draper was built from COTS radio-controlled helicopter with a Novatel RT-20 Differential Global Positioning System (DGPS) to provide navigation and velocity measurements. Systron Donner's Motion Pak inertia measurement unit (IMU) was used to provide 3 angular rates and 3 linear acceleration measurements and a sonar altimeter incorporated with Basic Stamp chips provided the altitude information. A PC104 stack was used as the main processing module with a Proxim's Proxlink2 modem performing the communication between the ground computer and the helicopter.

HummingBird [23], developed by Stanford Aerospace Robotics Laboratory, is another autonomous helicopter modified from radio-controlled helicopter with 5 GPS as its primary sensor for navigation and stabilization. It use the Carrier Phase Differential Technique to compute the altitude and position of the helicopter for the stabilization of the helicopter dynamics. An onboard 486 computer received all information from the GPS and complete the calculation of the helicopter position, velocity, attitude and attitude rate and then determined the appropriate control outputs which were fed to the helicopter servos through two 68HC11 microprocessors.

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Georgia Tech [24] developed an autonomous helicopter using Intel Pentium 133 as the flight computer. The sensors used include Attitude and Heading Reference System (AHRS) from Watson Industries which provided information on the helicopter attitude angles and rates. In addition, a 3-axis accelerometer was used to supplement the angular rate with linear acceleration. Two Polaroid ultrasonic range finders were used as ultrasonic altimeter to calculate the range to the ground. Novatel RT-20 DGPS was used to receive and update absolute earth coordinate position. An engine governor was also used to sustain constant rotor head speed.

US San Diego autonomous helicopter [25] was built based on a modified .60 scale radio controlled helicopter. This helicopter was equipped with sensors, an embedded micro-controller board and bi-directional communications links to a base station consisting of several networked PC computers. Sensors on the helicopter included 3 mechanical angular rate gyros from Futaba, a 2-axis inclinometer, a 3-axis magnetometer, a DGPS and an ultrasound altimeter. A Motorola 68332-based micro-controller board was used to carry out onboard computation.

Berkeley unmanned aerial vehicle system [26] was built on commercially available radio-controlled helicopter with a 233MHz Pentium MMX computer system to handle the computation task and control onboard of the helicopter. For navigation, the computer integrated the navigational sensor information from the inertia sensor, GPS, electronic compass and ultrasonic altimeter by Kalman filtering algorithm. A number of flight control algorithms were also used to control the servo system of the helicopter which helped to trim itself during the flight.

The USA helicopter [27] also uses commercially available model helicopter with two 486 industrial computers for its onboard computational tasks. One computer was used for communication purposes while the other one was used for navigation, control and mission management. The navigation algorithm provided attitude angles, velocity and position based on an inertia measurement unit with six strapped down sensors (3 gyros and 3 accelerometers) and a DGPS.

Rose-Hulman Aerial Robotics Vehicle [28] had a flight control system consisting of 7 fuzzy logic control loops on their autonomous helicopter. Fuzzy logic has the ability to interpret linguistic variable describing the air vehicle's states. The fuzzy logic control loops made use of the measurements from the AHRS, DGPS and a hall effect sensor to give the 13 state variables: x , y , z and their rates, roll, pitch, yaw and their rates and the main rotor speed.

2.4 Flight Controller design of rotary wing UAV

The flight control system design of the small-scale helicopter has a different objective as compared to the full-size helicopter. In the traditional full-sized helicopter control system design, the control system is designed primarily to improve the handling qualities of the aircraft for the pilot. This control system consists mainly of stability augmentation system (SAS) or simple autopilot system such as attitude hold autopilot. For fully autonomous flight operation of small-scale helicopter, more challenging and stringent requirements are imposed on the flight control system design as the smaller helicopter is much more agile, unstable, sensitive to wind disturbance and the dynamics are not as well understood as the bigger size counterpart.

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Various controller design methodologies have been applied to the controller design for the small-scale helicopter, from the classical controller design of PID controllers to the modern controller design of robust, nonlinear, fuzzy and adaptive controllers.

The classical controller design is the simplest method to implement based on decoupled SISO PID feedback loops. An example of such a controller scheme used is the Berkeley Yamaha R-50 model helicopter [22]. In this research, the MIMO helicopter dynamics is decoupled into four SISO subsystems with a substantial amount of coupling among the system being ignored. The four SISO subsystems consist of roll, pitch, yaw and heave channels and are stabilized by the proportional-differential (PD) controllers. Another application of classical controller design is by Carnegie Mellon's RUAV [10]. The control system used in this research consists of a PD position command system built around a proportional attitude control system. The positional controller regulates the attitude setpoint of the attitude controller. The vertical position and the heading are controlled by two separate PD control loops.

Modern control methodologies for robust controllers designs are quite popular. These include the H_∞ method [42] and LQR method [37]. In [18], the Wepilot autopilot system from the Measurement and Control Laboratory at ETH Zurich uses a H_∞ controller to provide for state feedback control of position and heading. In [16], an LQR controller with setpoint tracking is designed and implemented. The LQR controller implemented shows a faster response with less overshoot than the corresponding H_∞ controller implemented. However, there is noticeable steady-state tracking error with the LQR controller.

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Fuzzy logic controller has been applied in [36]. The fuzzy logic controller is based on a self organizing process that learns the appropriate relationship between control input and output. This autopilot is composed of four separate modules which correspond to the control actuators of the helicopter and is regulated by a PID controller with multiple fuzzy controller organized in a hierarchical manner.

Chapter 3

Helicopter Dynamics, Modeling and Parameter Model Development for System Identification

An accurate model that described the helicopter dynamics is essential to the flight controller design of the system. However, rotorcraft dynamics are complex and high fidelity nonlinear model is very difficult for controller implementation. Hence there is a need for trade-off between the fidelity of the dynamic model being developed and the type of controller design that is implemented.

In general, modeling can be done from the first principle modeling where a general nonlinear system model can be developed using the laws of aerodynamics and mechanics. This analytical model developed is very useful for the construction of simulation model provided that accurate knowledge of various system parameters such as rotor forces and moments are available. The major difficulty of this approach is that accurate knowledge of the aerodynamic parameters and other mechanical parameters are hard to obtain for small-size helicopter. Hence many engineering assumptions are adopted in the formulation of the model equations. As such, the model obtained may not correctly describe the actual helicopter dynamics. In addition, it cannot correctly predict the off-axis response of the helicopter.

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In contrast, system identification methodology begins with measured data and uses these data to extract a mathematical model that reflects the aircraft motion and this is usually sufficiently accurate. The model obtained is usually linear and low order model and is primary used for the design and implementation of flight controller. Besides using the flight test data for system identification, model validation and refinement processes are done using the system identification method. However, as the models obtained from the system identification are usually linear, they are valid in the vicinity of an operation range such as hover, slow forward speed flight or fast forward speed flight. This operational range is limited by the range of the flight test data used in the system identification process and the linear low order model constraint.

As the flight dynamics of the small-scale helicopter has not been well published in the helicopter community, there is still a lack of understanding on the flight dynamics to derive the helicopter dynamic model and its limitations by using different modeling techniques. As such, the approach to the modeling of the small-scale helicopter in this project will have to encompass both the empirical method and system identification technique in complementary so as to gain a better understanding of the dynamics of small-scale helicopter and its limitations. This will help in the flight controller design and implementation in the later part. In this chapter, the basic operation, working principles and dynamics of the small-scale helicopter will be covered. Subsequently, modeling of the helicopter is done based on its hover dynamics and this is extended to the development of the parameterized model that is used for system identification in the subsequent chapter.

3.1 Small scale helicopter design and operation

The small-scale helicopter that is used in this project is a COTS radio-controlled model helicopter as shown in Figure 3.1.

Figure 3.1 Commercial off the shelf model helicopter

The radio-controlled helicopter is probably the most complex type of radio-controlled model flight vehicle due to its agility and cross-coupled dynamics. Flying these helicopters required 100% concentration as most of these model helicopters are operated in open loop mode, with a pilot providing the feedback to control the helicopter. The model works on the same principles as the full-size helicopter and controlling the helicopter is just as difficult, if not more so due to size and orientation. It is not simply a matter of pushing one button for up, and another for forward flight

3.1.1 Mechanical Description of helicopter

The main parts of the model helicopter are covered briefly to have a better understanding of the model helicopter system.

Main Rotor Head Assembly

The main rotor head assembly consists of mixing arm, flybar, flybar control arm and the blade grip for the main rotor blade. The function of the main rotor assembly is to hold the main rotor blades and the flybar assembly together.

Swashplate Assembly

The swashplate assembly, as shown in figure 3.2, is made up of essentially a large ball bearing with the rotating inner race going up to the rotor head and the non-rotating outer race going to the push rod and servos. The function of the swashplate is to simply transfer the non-rotating actions of the servo pushrods to the rotating rotor head which allows for the cyclic and collective control of helicopter through the variation of the blade pitch angle.

Figure 3.2 Helicopter Swashplate

Stabilizer Bar Assembly

The stabilizer bar assembly is made up of a rod carrying small aerofoils (paddles) mounted at the extreme end of the rod and is pivoted at main rotor shaft so that it can rock freely. Mechanical mixing linkages are connected from the swashplate to the stabilizer bar and to the main rotor blade pitch link. The purpose of the stabilizer bar is to provide damping to the helicopter system as the dynamics of the small size helicopter is very fast.

Figure 3.3 Stabilizer bar assembly

Tail rotor assembly

The tail rotor provides the helicopter with a counter torque against the torque generated by the main rotor so that the heading direction of the helicopter can be controlled. The pitch angle of the tail rotor blade can be changed so that the tail boom can be swung either to the left or right. The tail rotor blade pitch is controlled by a rudder servo which is mounted at the rear of the main frame of the helicopter.

3.1.2 Control of the helicopter

The helicopter is directly controlled by using a remote radio control system which includes the radio transmitter, receiver, servos, battery and gyro. The servos are installed on the main body assembly while the receiver, battery and gyro are fixed onto the transmitter tray of the model helicopter. The motion of the model helicopter is controlled by five servos which are the throttle, lateral cyclic pitch, longitudinal cyclic pitch, collective pitch and the rudder servos. Servo horns and universal links are used to transmit the angular motion from the servos to the various control inputs of the helicopter.

In general, there are 4 basic controls that can be applied to the control of the model helicopter. They are:

- (i) engine throttle control
- (ii) collective pitch control
- (iii) longitudinal and lateral cyclic pitch control
- (iv) rudder control

Engine throttle control

The engine throttle control changes the amount of fuel and air that enters the carburettor of the engine in order to control the engine speed. This is achieved by the throttle servo rotation, rotating clockwise to close the throttle and counter-clockwise to open the throttle. This control is typically coupled together on the same stick as the collective pitch control. In order to keep the main rotor blades rotation at a constant

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RPM, an engine governor is added to the system to provide a close loop feedback to control the throttle servo.

Collective pitch control

The collective pitch control is used to increase the lift of the main rotor blades so that the helicopter can climb or descend through the movement of the swashplate when it is pushed up vertically or down without tilting the swashplate. When the swashplate is pushed vertically upward, the main rotor blade's pitch will increase and this will increase the lift from the rotor blades due to a higher angle of attack from the blades. With the increase in the vertical lift, the helicopter will climb. Vice versa, when swashplate is pushed down, the lift from the main rotor blades will decrease and this will cause the helicopter to descent.

Longitudinal and Lateral Cyclic pitch control

The cyclic pitch control of the helicopter comes in two parts, the longitudinal cyclic and the lateral cyclic control. The longitudinal cyclic control results in the pitching motion of the helicopter. The helicopter can either pitch forward or backward with the control input from the longitudinal cyclic servo pushing or pulling the swashplate that it is either being tilted forward or backward, pivoting about the center of the swashplate at the main rotor shaft. If the longitudinal cyclic pushrod is pushed up by the longitudinal cyclic servo, the swashplate will tilt forward.

Figure 3.4 Port view of swashplate motion for forward and backward pitch

Figure 3.5 Plan view of rotor disc

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Since the pitch arm of the blade is attached to the swashplate 90° ahead, the blade (shown in figure 3.4), which rotates in the clockwise direction, has its pitch angle increased when it is on the right-hand side (Retreating blade) and pitch angle decreased on the left-hand side (Advancing blade)^[13]. The forward tilt of the swashplate has no effect on the blade pitch angle when the blade is over the nose or the tail of the helicopter. The resultant of the unbalanced lift from the pitch angle change accelerates the left-hand blade down as it moves towards the nose and the right-hand blade up on its way to the tail resulting to the rotor flapping. The rotor flaps down over the nose and up over the tail and the effect of this flapping is translated as a moment about the center of gravity of the helicopter. In this case, the moment tilts the helicopter forward, resulting to a pitch forward of the helicopter.

The lateral cyclic results to the rolling motion of the helicopter, either a roll to the left or right. The lateral cyclic control will tilt the swashplate to the left or right. To have a right roll for the helicopter, the lateral cyclic servo will tilt the swashplate to the right while pivoting about the centerline along the center of the swashplate. This results in the change of the blade pitch angle of the rotor blade at the nose and tail positions. The blade at the nose will have a decrease in pitch while the blade at the tail will have an increase in the pitch angle. The blade pitch angle remains unchanged when the blade is at the left or right-hand side of the helicopter. Again, due to rotor flapping as the rotor flaps downward to the right-hand side and upward to the left-hand side of the helicopter, the helicopter will roll to the right.

Figure 3.6 Rear view of swashplate motion for left and right roll.

Rudder control

The rudder control of the helicopter changes the blade pitch angle of the tail rotor blades and causes the tail boom to swing either left or right. The rudder control is controlled by a rudder servo which is mounted at the rear of the main frame of the helicopter. The servo motion is transmitted through the tail rod to the rudder lever at the tail as shown in Figure 3.6. The speed of the tail rotor rotation is dependent on the main rotor speed since it is directly driven by the belt through the gear that is coupled to the main rotor shaft.

Figure 3.7 Tail rotor control

3.2 Helicopter Dynamics and modeling

The helicopter dynamics can be understood through equations developed from first principles modeling. However, explicit details of the helicopter dynamics modeling from the first principle modeling will not be carried out as the final model obtained from the modeling is desired to be a simple and low order linear model for the subsequent system identification and flight controller synthesis. Hence, the approach for the modeling will be done in a way that is adequate for basic helicopter dynamics to

be understood and the result of the model developed be applicable in the subsequent system identification and controller design works.

3.2.1 Rigid Body Equation of Motion of a Helicopter

The fundamental of helicopter modeling starts from the formation of the equations of motion for a rigid body system.

The helicopter is a system that is capable of rotating and translating in six degrees of freedom (6 DOF). The helicopter dynamics can be studied by employing a lumped parameter approach where the helicopter can be seen as a composition of the main rotor, tail rotor, fuselage, horizontal stabilizer and vertical stabilizer. For a constant mass m and moment of inertia $\mathbf{I}$, using Newton-Euler equations expressed in the inertial reference frame,

$$m \frac{d \mathbf{v}^I}{dt} = \mathbf{F} \quad (3.1)$$

$$\mathbf{I} \frac{d \boldsymbol{\omega}^I}{dt} = \mathbf{M} \quad (3.2)$$

where $\mathbf{F} = [X \ Y \ Z]^T$ is the vector of external forces acting on the vehicle's center of gravity and $\mathbf{M} = [L \ M \ N]^T$ is the vector of external moments. The external forces and moments are produced by the main and tail rotor, gravitational forces and the aerodynamics forces from the fuselage components.

For analysis of dynamics system equations, the equations of motion are expressed in the body-fixed reference frame as follows:

$$m \frac{d\mathbf{v}}{dt} + m (\boldsymbol{\omega} \times \mathbf{v}) = \mathbf{F} \quad (3.3)$$

$$\mathbf{I} \frac{d\boldsymbol{\omega}}{dt} + (\boldsymbol{\omega} \times \mathbf{I} \boldsymbol{\omega}) = \mathbf{M} \quad (3.4)$$

where $\mathbf{v} = [u \ v \ w]^T$ and $\boldsymbol{\omega} = [p \ q \ r]^T$ are the helicopter velocities and angular rates respectively in the body-fixed frame.

Figure 3.8 Body fixed reference system with displacement variables

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Hence, the 6 DOF rigid body equations of motion for the helicopter are given by the three differential equations (Eq 3.5, Eq 3.6 & Eq 3.7) for the translational motion from equation 3.3 and three differential equations (Eq 3.8, Eq 3.9 & Eq 3.10) for the rotational motion from equation 3.4. These six differential equations expressed as a function of external forces and moments are as follows:

$$\dot{u} = (-wq + vr) + \frac{X}{m} \quad (3.5)$$

$$\dot{v} = (-ur + wp) + \frac{Y}{m} \quad (3.6)$$

$$\dot{w} = (-vp + uq) + \frac{Z}{m} \quad (3.7)$$

$$\dot{p} = -\frac{qr(I_{yy} - I_{zz})}{I_{xx}} + \frac{L}{I_{xx}} \quad (3.8)$$

$$\dot{q} = -\frac{pr(I_{zz} - I_{xx})}{I_{yy}} + \frac{M}{I_{yy}} \quad (3.9)$$

$$\dot{r} = -\frac{pq(I_{xx} - I_{yy})}{I_{zz}} + \frac{N}{I_{zz}} \quad (3.10)$$

3.2.2 Linearized Model

The application of Newton's law of motion to a helicopter in flight leads to a set of nonlinear differential equations (Eq 3.5 to Eq 3.10) for the evolution of the aircraft response trajectory and attitude with time. To simplify these nonlinear equations to a linear time-invariant dynamic model for the design of linear flight controller, linearization approach is used.

In the general form, the quasi-steady 6 DOF helicopter equations of motion can be written in the form of nonlinear differential equations in the first order vector form

$$\dot{\mathbf{x}} = \mathbf{f}(\mathbf{x}, \mathbf{u}) \quad (3.11)$$

where $\mathbf{x}$ is the column vector of helicopter states, $\mathbf{u}$ is the vector of control input vector and $\mathbf{f}$ is a nonlinear function of the helicopter motion. For this rigid-body dynamics described by Eq 3.5 to Eq 3.10, the state vector is

$$\mathbf{x} = [u, v, w, p, q, r, \phi, \theta]^T$$

and the control input vector is

$$\mathbf{u} = [\delta_{lat}, \delta_{lon}, \delta_{col}, \delta_{ped}]^T$$

where $[\phi, \theta]$ are the roll and pitch angles of the body and $[\delta_{lat}, \delta_{lon}, \delta_{col}, \delta_{ped}]$ are the control inputs for the lateral cyclic, longitudinal cyclic, collective and tail pedal respectively.

The nonlinear differential equations can be linearized about a trim state $(\mathbf{x}_0, \mathbf{u}_0)$ to form a linearized form of the equations of motion. This model is commonly known as the stability derivatives model where the external forces and moments are represented in terms of the stability and control derivatives. Using small perturbation theory, the

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disturbed motion of the helicopter behavior can be described as a linear perturbation from trim condition where $x = x_0 + \delta x$ and $u = u_0 + \delta u$

Hence the linearized model is given by

$$\delta \dot{x} = \left[\frac{\partial f}{\partial x} \right]_{(x_0, u_0)} \delta x + \left[\frac{\partial f}{\partial u} \right]_{(x_0, u_0)} \delta u \quad (3.12)$$

The linearized 6 DOF equations of motion are

$$\delta \dot{u} = (-\omega_0 \delta q + \delta \omega q_0 + v_0 \delta r + \delta v r_0) + \frac{\Delta X}{m} \quad (3.13)$$

$$\delta \dot{v} = (-u_0 \delta r + \delta u r_0 + \omega_0 \delta p + \delta \omega p_0) + \frac{\Delta Y}{m} \quad (3.14)$$

$$\delta \dot{w} = (-v_0 \delta p + \delta v p_0 + u_0 \delta q + \delta u q_0) + \frac{\Delta Z}{m} \quad (3.15)$$

$$\delta \dot{p} = \frac{(-q_0 \delta r - \delta q r_0)(I_{yy} - I_{zz})}{I_{xx}} + \frac{\Delta L}{I_{xx}} \quad (3.16)$$

$$\delta \dot{q} = \frac{(-p_0 \delta r - \delta p r_0)(I_{zz} - I_{xx})}{I_{yy}} + \frac{\Delta M}{I_{yy}} \quad (3.17)$$

$$\delta \dot{r} = \frac{(-p_0 \delta q - \delta p q_0)(I_{xx} - I_{yy})}{I_{zz}} + \frac{\Delta N}{I_{zz}} \quad (3.18)$$

For the equilibrium point in which the helicopter is in hover flight condition, the linearized 6 DOF equations can be further simplified as the linear and angular velocities are zero ($v_0 = u_0 = w_0 = p_0 = q_0 = r_0 = 0$).

Hence, for a hover flight condition, the linearized 6 DOF equations of motion is reduced to

$$\delta \dot{u} = \frac{\Delta X}{m} \quad (3.19)$$

$$\delta \dot{v} = \frac{\Delta Y}{m} \quad (3.20)$$

$$\delta \dot{w} = \frac{\Delta Z}{m} \quad (3.21)$$

$$\delta \dot{p} = \frac{\Delta L}{I_{xx}} \quad (3.22)$$

$$\delta \dot{q} = \frac{\Delta M}{I_{yy}} \quad (3.23)$$

$$\delta \dot{r} = \frac{\Delta N}{I_{zz}} \quad (3.24)$$

With the linearized 6 DOF equations of the motion of the rigid body system, the next step is to formulate the external forces and moments that act on the body. This is the main problem and the most difficult task in the development of the helicopter model. This will involve the formulation of each force and moment term and the measurement of the geometric constant specific to the location of the center of mass and location of the main rotor, tail rotor and stabilizer fins. In addition, detail analysis of the rotor and fuselage dynamics are required and this is not a trivial task. However, if system identification approach is used, the construction of the model can be formulated using a highly simplified expression.

3.2.2 Forces and Moments

A fundamental assumption of linearization is that the external forces X , Y , Z and moments L , M , N can be represented as an analytical function of the disturbed motion variables and their derivatives.

Using Taylor's series expansion, if the forces and moments functions and all its derivatives are known at any point (at trim condition), then the behavior of that function anywhere in its analytical range can be estimated from an expansion of the function in a series about the known point. Linearization amounts to neglecting all terms except the linear terms in the equation. The validity of linearization depends on the behavior of the forces at small amplitude. This means that the dominant effect should be linear only as the control and disturbances become very small. The forces can be written in the approximate form with the first order terms in the series:

$$\begin{aligned} \Delta X = & \frac{\partial X}{\partial u} \delta u + \frac{\partial X}{\partial v} \delta v + \frac{\partial X}{\partial w} \delta w + \frac{\partial X}{\partial p} \delta p + \frac{\partial X}{\partial q} \delta q + \frac{\partial X}{\partial r} \delta r + \frac{\partial X}{\partial \phi} \delta \phi + \frac{\partial X}{\partial \theta} \delta \theta + \frac{\partial X}{\partial \psi} \delta \psi \\ & + \frac{\partial X}{\partial \delta_{lat}} \delta_{lat} + \frac{\partial X}{\partial \delta_{lon}} \delta_{lon} + \frac{\partial X}{\partial \delta_{col}} \delta_{col} + \frac{\partial X}{\partial \delta_{ped}} \delta_{ped} \end{aligned} \quad (3.25)$$

All the forces and moments can be expanded in this manner. The partial derivatives of the forces or moments with respect to the vehicle states are called the stability derivatives and with respect to the vehicle control inputs are called control derivatives.

In simple representation, these derivatives will be represented in the form

$$\frac{\partial X}{\partial u} = X_u \quad \frac{\partial X}{\partial \delta_{lat}} = X_{lat}$$

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In addition, the deltas (δ and Δ) notation will be drop from all the variables in the equation except from the control inputs. Hence, Eq 3.25 can be rewritten as

$$\begin{aligned}\Delta X = & X_u u + X_v v + X_w w + X_p p + X_q q + X_r r + X_\phi \phi + X_\theta \theta + X_\psi \psi \\ & + X_{lat} \delta_{lat} + X_{lon} \delta_{lon} + X_{col} \delta_{col} + X_{ped} \delta_{ped}\end{aligned}\quad (3.26)$$

From the linearized force and moment components, not all the stability or control derivatives will be relevant. As proposed by Mettler[10], to simplify the 6 DOF helicopter model, the terms with negligible contribution to the forces and moments are discarded and this results to the simplified forces and moments equations as follows:

$$\frac{\Delta X}{m} = X_u u + X_{lon} \delta_{lon} \quad (3.27)$$

$$\frac{\Delta Y}{m} = Y_v v + Y_{lat} \delta_{lat} \quad (3.28)$$

$$\frac{\Delta Z}{m} = Z_w w + Z_{col} \delta_{col} \quad (3.29)$$

$$\frac{\Delta L}{I_{xx}} = L_u u + L_v v + L_{lat} \delta_{lat} \quad (3.30)$$

$$\frac{\Delta M}{I_{yy}} = M_u u + M_v v + M_{lon} \delta_{lon} \quad (3.31)$$

$$\frac{\Delta N}{I_{zz}} = N_r r + N_{ped} \delta_{ped} + N_{col} \delta_{col} \quad (3.32)$$

Note that the derivatives for the force equations are normalized by the mass of the helicopter and the derivatives for the moment equations are normalized by the respective moment of inertia.

3.2.3 Gravity Force

The gravitational force is one of the force components that is acting on the body of the helicopter. The gravitational force vector in the x, y and z body inertial reference frame can be written as $[0 \ 0 \ g]^T$. In the body reference frame, it is transformed using the Euler angle orientation. The Euler angle transformation matrix from the inertia frame to the body reference frame is given by:

$$\begin{bmatrix} \cos \psi \cos \theta & \sin \psi \cos \theta & \sin \theta \\ \cos \psi \sin \theta \sin \phi - \sin \psi \cos \phi & \sin \psi \sin \theta \sin \phi + \cos \psi \cos \phi & \cos \theta \sin \phi \\ \cos \psi \sin \theta \cos \phi + \sin \psi \cos \phi & \sin \psi \sin \theta \cos \phi - \cos \psi \sin \phi & \cos \psi \cos \phi \end{bmatrix}$$

This results in the gravity vector of $[-g \sin \theta \ g \cos \theta \sin \phi \ g \cos \theta \cos \phi]^T$ in the body axis.

Using small angle approximation, the gravity vector becomes $[-g\theta \ g\phi \ g]^T$. Hence, the forces due to gravity are

$$X_g = -g\theta \quad (3.33)$$

$$Y_g = g\phi \quad (3.34)$$

$$Z_g = g \quad (3.35)$$

In addition, to improve the rigid body model fidelity, other critical dynamics that is not captured in the rigid body model can be coupled to this model, such as the coupled rotor-fuselage dynamics [21].

3.2.4 Hybrid Model Formulation

The quasi-steady model, which represents the helicopter as a rigid body system has been formulated in the previous section. This quasi-steady model is commonly used in the rotorcraft system identification for the application to simulations that do not require high frequency validity. This 6 DOF equation formulation account for the rotor dynamics as simple time delays, absorbing the steady-state effects of the rotor flapping into the conventional stability and control derivatives [21]. Hence, this can only adequately describe the low and mid-frequency dynamics.

However, for flight control system application, the model must be accurate in the frequency range of about 0.3 to 3 times of the crossover frequency [21]. The dynamic model within this frequency range will have a significant influence on the magnitude and phase characteristics near the crossover frequency and will thus be important in predicting the close-loop behavior. This range is beyond the quasi-steady model. Therefore a higher-order model is needed to capture the low frequency body dynamics and the high frequency rotor flapping dynamics. The hybrid model was developed to address this limitation.

The hybrid model formulation adds additional dynamics to the system by introducing new states to improve the model validity. Highly simplified rotor dynamic equations that capture the on and off axis flapping response for cyclic control and angular body rates are input for the high frequency dynamics. All the quasi-steady derivatives are retained to account for coupling terms and low frequency dynamics. The rotor and fuselage dynamics are coupled through effective rotor spring terms.

The hybrid model formulation will have the rotor motion modeled through a simple tip-path plane model and the rotor forces and moments expressed in terms of the rotor states. Therefore, the rotor dynamics need to model explicitly so that it can then be coupled to the fuselage equations of motion.

3.2.5 Simplified Rotor Model

Main Rotor dynamics

The main rotor is the most crucial and complicated part of the helicopter dynamics. It generates the vertical thrust for the helicopter to lift off the ground. The lift generated by the main rotor blade is the function of many factors, such as the relative speed, air density, airfoil shape, angle of attack of the blade and so on. Beside rotation motion about the main shaft (measured by Ψ), the rotor blade also feathers to change the blade pitch angle (measured by Θ) and flaps in a normal direction to the rotor disc (measured by β).

The main rotor system also has a swashplate mechanism that changes the blade pitch angle simultaneously or as a function of the angular position of the main rotor shaft. The thrust and rotor moment are produced by changing the blade pitch angle. When the blade pitch angle change simultaneously, it is called a collective pitch. Collective pitch changes the blade pitch angle to both of the main rotor blades to give an average blade pitch angle and this control the vertical lift. When the blade pitch angle changes as a function of the angular position about the hub, it is called the cyclic pitch. The cyclic pitch changes the distribution of the lift force over the disc so that the direction of the thrust vector can be tilt from the upright direction. It generates rolling or pitching

moment to cause the fuselage to roll or pitch, depending on whether a longitudinal or a lateral cyclic pitch has been applied.

From Prouty [8], the local blade pitch angle $\Theta(\Psi)$, as a function of its positional azimuth angle Ψ around the hub, is given by the equation

$$\Theta(\Psi) = \Theta_0 - B_{\text{lat}} \delta_{\text{lat}} \cos \Psi - A_{\text{lon}} \delta_{\text{lon}} \sin \Psi \quad (3.36)$$

where

Θ_0 = Average blade pitch set by collective control, δ_{col}

$B_{\text{lat}} \delta_{\text{lat}}$ = Lateral cyclic pitch set by lateral cyclic control, δ_{lat}

$A_{\text{lon}} \delta_{\text{lon}}$ = Longitudinal cyclic pitch set by longitudinal cyclic control, δ_{lon}

A_{lon} and B_{lat} are the linear constant coefficients used to normalize the longitudinal and lateral cyclic control from range of +/- 100% to angle input of +/- 1 radian. The unit of measurement is rad / %.

The azimuth angle Ψ definition is shown in Figure 3.9.

Figure 3.9. Definition of Azimuth angle. The blade positional angle Ψ is zero when blade is over at the tail.

Rotor Flapping Dynamics

The flapping motion is an important characteristic in the helicopter. Flapping is an oscillatory motion of the main rotor blades about the hinges in which the blades flap perpendicular to the rotor disc. The flapping motion is a result of the fluctuating thrust caused by the change of the angle of attack of the blades, the velocity and direction of local flow of air into the main rotor. Since the lift is perpendicular to the blade surface, if the blade is flapping along the flapping hinge, the overall lift over the blade has a vertical and a horizontal component. The horizontal component acts as the moment in rolling and pitching as well as the forces (X and Y) in x and y directions.

When the cyclic control is applied to the swashplate while the blades are rotating, the blades experience a periodic change in the angle of attack and velocity. This periodic changes in the angle of attack of the blade results in the periodic changes in the blade lift. The end result will be the periodic flapping motion of the blade about the hinge, which will produce forces and moments on the rotor hub. In a simplified rotor model, the blade is assumed to be rigid and its motion can be described by the motion of the blade tip, which is commonly known as the tip-path plane motion. The blade flapping motion equations are derived from the balance of moments about the flapping hinge, in which the moment are contributed by the blade aerodynamics force, centrifugal force, inertia force and flapping restraint of the rotor hub. Details of the derivation of the blade flapping equation will not be covered and can be found in reference [29]. With an alternating blade pitch angle Θ , the blade flaps up and down during its revolution with angle β to the plane perpendicular to the main rotor shaft. This angle β can be expressed by Fourier series [30]

$$\beta(\Psi) = a_0 - a_1 \cos \Psi - b_1 \sin \Psi - a_2 \cos 2\Psi - b_2 \sin 2\Psi - \dots \quad (3.37)$$

As the magnitude of higher harmonics are negligible as compared to the first harmonic, the second and higher harmonics can be truncated and ignored.

Hence the first harmonic representation of the blade flapping motion is

$$\beta(\Psi) = a_0 - a \cos \Psi - b \sin \Psi \quad (3.38)$$

Equation 3.38 is also known as the tip-path plane equation. a_0 describes the coning angle, a describes the longitudinal rotor flapping angle and b describes the lateral rotor flapping angle.

Figure 3.10 Side view of rotor system showing rotor flapping angles

From Mettler's work [10], equation 3.28 can be transformed from the rotating variable Ψ to non-rotating variables involving a and b . The coupled first order rotor flapping equations proposed by Mettler, which capture the key tip-path plane responses due to control inputs and vehicle motion, are given by

$$\tau_f \dot{\mathbf{b}} = -\mathbf{b} - \tau_f p - \frac{q}{\Omega} - B_a \mathbf{a} + B_{lat} \delta_{lat} \quad (3.39)$$

$$\tau_f \dot{\mathbf{a}} = -\mathbf{a} - \tau_f q - \frac{p}{\Omega} - A_b \mathbf{b} + A_{lon} \delta_{lon} \quad (3.40)$$

The term τ_f is the rotor time constant for the rotor together with the stabilizer bar and is given by $\frac{16}{\gamma\Omega}$ while $-\tau_f q$ and $-\tau_f p$ are the longitudinal and lateral flapping produced by a body pitching rate q and rolling rate p and this corresponds to the pitch and roll rotor damping. The cross-coupling effects, $-\frac{q}{\Omega}$ and $+\frac{p}{\Omega}$ are the lateral and longitudinal flapping resulting from the change of blade angle of attack produced by a body rolling rate p and pitching rate q . The stability derivatives, B_a and A_b , are another cross-coupling effect due to the flapping restraint. B_a and A_b are the lateral and longitudinal flapping derivatives and are given by $\frac{\tau_f K_\beta}{2\Omega I_\beta}$ where K_β is the flapping hinge restraint and I_β is the rotor blade moment of inertia about the flapping hinge.

Rotor and fuselage coupling dynamics

The rotor and fuselage coupling dynamics are achieved by having the rotor forces and moments expressed in term of the rotor states. The forces and moments are produced by the main rotor blades exerting on the rotor hub. The rotor forces and moments analysis are based on the tip-path plane rotor model. Figure 3.11 shows the forces and moments that are exerted on the rotor head.

Figure 3.11 Schematic of forces and moments on the rotor hub.

The forces along the body axis contributed by the rotor thrust vector, T , using small angle approximation, are as follows

$$X_R = -T \sin a \cos b \approx -T a \quad (3.41)$$

$$Y_R = T \sin b \cos a \approx T b \quad (3.42)$$

$$Z_R = -T \cos a \cos b \approx -T \quad (3.43)$$

The moments acting on the body are produced by the rotor flapping and the tilting of the rotor thrust vector.

For moment produced by rotor flapping, to calculate the hub torsion moment, the restraint at the blade attachment to the rotor hub is approximated using a linear torsional spring with a constant spring rate K_β . Hence, the lateral hub torsional moment

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(roll moment) L_k and longitudinal hub torsional moment (pitch moment), M_k are given by

$$L_k = K_\beta \mathbf{b} \quad (3.44)$$

$$M_k = K_\beta \mathbf{a} \quad (3.45)$$

For a distance h between the center of gravity and the rotor thrust vector (shown in Figure 3.8), the moments produced by the tilting of the rotor thrust vector, lateral moment from force Y (roll moment), L_T and longitudinal moment from force X (pitch moment), M_T are

$$L_T = hT\mathbf{b} \quad (3.46)$$

$$M_T = hT\mathbf{a} \quad (3.47)$$

Hence, the total roll (L_R) and pitch (M_R) moments are

$$\begin{aligned} L_R &= K_\beta \mathbf{b} + hT\mathbf{b} \\ &= (K_\beta + hT)\mathbf{b} \end{aligned} \quad (3.48)$$

$$\begin{aligned} M_R &= K_\beta \mathbf{a} + hT\mathbf{a} \\ &= (K_\beta + hT)\mathbf{a} \end{aligned} \quad (3.49)$$

The forces and moment produced by the main rotor can be expressed in term of the stability derivatives so that it can be coupled to the fuselage rigid body equation of motion (equation 3.19 to 3.24) in terms of rotor states $\mathbf{a}$ and $\mathbf{b}$.

From equation 3.41 to 3.43, it can be seen that only the lateral and longitudinal forces produced by the main rotor, X_R and Y_R are related to the rotor states. Therefore, the lateral force derivative (X_a) and the longitudinal force derivative (Y_b) are used to

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couple to rigid body equations. This will replace the derivatives X_{lon} and Y_{lat} in equations 3.27 and 3.28. As for the moments, both L_R and M_R are related to the rotor states and hence they are coupled to the rigid body equations by the roll and pitch moment derivatives, M_a and L_b . This will replace the derivatives M_{lon} and L_{lat} in equations 3.30 and 3.31. With the replacement of the rotor forces and moments terms X_{lon} , Y_{lat} , M_{lon} and L_{lat} by the flapping derivatives X_a , Y_b , M_a and L_b , the cyclic commands δ_{lat} and δ_{lon} will be input directly to the rotor dynamic equations 3.39 and 3.40. The cyclic commands will indirectly affect the rigid body equations through the rotor flapping angles **a** and **b** obtained from the rotor dynamic equations 3.39 and 3.40. Hence the coupled rotor fuselage equations of motion, with the combination of equations 3.19 to 3.24 (rigid body equations on motion), equations 3.27 to 3.32 (forces and moments components) and equations 3.33 to 3.35 (gravity force) are as follows:

$$\dot{u} = X_u u - g\theta + X_a \mathbf{a} \quad (3.50)$$

$$\dot{v} = Y_v v + g\phi + Y_b \mathbf{b} \quad (3.51)$$

$$\dot{w} = Z_w w + Z_{col} \delta_{col} \quad (3.52)$$

$$\dot{p} = L_u u + L_v v + L_a \mathbf{a} + L_b \mathbf{b} \quad (3.53)$$

$$\dot{q} = M_u u + M_v v + M_a \mathbf{a} + M_b \mathbf{b} \quad (3.54)$$

$$\dot{r} = N_r r + N_{ped} \delta_{ped} + N_{col} \delta_{col} \quad (3.55)$$

Parameterized state space model

The vehicle dynamic equations of motion will be combined to form the parameterized state space model to be used in the system identification in the later phase. The differential equations for the entire vehicle dynamics can be written as 10 first order differential equations as suggested by Mettler[10] to be put into the state space representation.

Coupled longitudinal and lateral dynamics

The coupled longitudinal and lateral dynamics are made up of the fuselage lateral and longitudinal motion from equations 3.50 and 3.51, fuselage roll and pitch motion from equations 3.53 and 3.54 and main rotor dynamics of rotor lateral and longitudinal flapping from equations 3.39 and 3.40. The roll and pitch Euler angles, θ and ϕ , are added into state space model as well.

Heave dynamics

The heave dynamics equation is given by equation 3.52 where the control derivative Z_{col} is the control derivative that accounts for the change in the main rotor thrust due to the change in the collective blade pitch input δ_{col} .

Yaw dynamics

The yaw dynamics equation is given by equation 3.55. The yaw moment, resulted from the moment of the tail rotor thrust, is controlled by the control input δ_{ped} . The moment produced by the vertical tail aerodynamic force is neglected as the effect is negligible

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in hover flight. The presence of a tail gyro helps to provide damping to the yaw dynamics. The dynamics of the yaw gyro is contained in the overall yaw dynamics and hence there is no necessity to have a component wise dynamics for the yaw gyro and the bare tail dynamics.

The state space model can be put into the general form

$$\dot{\mathbf{x}} = \mathbf{A}\mathbf{x} + \mathbf{B}\mathbf{u} \quad (3.56)$$

$$\mathbf{y} = \mathbf{C}\mathbf{x} \quad (3.57)$$

where $\mathbf{x}$ is the state vector, $\mathbf{A}$ is the matrix that contains the stability derivatives, $\mathbf{B}$ is the matrix that contains the control derivatives, $\mathbf{u}$ is the control vector and $\mathbf{y}$ is the measurement vector. The $\mathbf{A}$, $\mathbf{B}$ and $\mathbf{C}$ matrices and $\mathbf{x}$, $\mathbf{u}$ and $\mathbf{y}$ vectors are given by

$$\mathbf{A} = \begin{bmatrix} \frac{1}{\tau_f} & \frac{B_s}{\tau_f} & -1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ \frac{A_b}{\tau_f} & \frac{1}{\tau_f} & 0 & -1 & 0 & 0 & 0 & 0 & 0 & 0 \\ L_b & L_a & 0 & 0 & 0 & 0 & L_u & L_v & 0 & 0 \\ M_b & M_a & 0 & 0 & 0 & 0 & M_u & M_v & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & X_a & 0 & 0 & 0 & -g & X_u & 0 & 0 & 0 \\ Y_b & 0 & 0 & 0 & g & 0 & 0 & Y_v & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & Z_w & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & N_r \end{bmatrix}$$

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$$B = \begin{bmatrix} B_{lat} & 0 & 0 & 0 \\ 0 & A_{lon} & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & Z_{col} & 0 \\ 0 & 0 & N_{col} & N_{ped} \end{bmatrix}$$

$$C = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 \end{bmatrix}$$

$$\mathbf{x} = [\mathbf{a}, \mathbf{b}, u, v, \Phi, \theta, p, q, w, r]^T$$

$$\mathbf{u} = [\delta_{lat}, \delta_{lon}, \delta_{col}, \delta_{ped}]^T$$

$$\mathbf{y} = [\mathbf{a}, \mathbf{b}, u, v, \Phi, \theta, p, q, w, r]^T$$

Chapter 4

Hardware, Software and System Integration

A conventional UAV consists of the air vehicle platform, avionics, flight computer, flight software, telemetry and the ground control system. The successful operation of the UAV system depends very much on the instrumentation and system integration of the UAV system before a flight controller can be designed to control the UAV system. Without a well instrumented system, even the best designed flight controller cannot do the job of controlling the UAV system. The integration of the helicopter system is not a trivial task as many mechanical and electronic components are required to be integrated into the small helicopter platform which has a limited payload. In addition, the integration of various components into the small platform will affect the inertial, center of gravity and aerodynamics behavior of the platform. Besides having a constrained mounting space and payload, the onboard components that are to be mounted is subjected to harsh environmental factors such as vibration from the engine and moving mechanical parts, heating from the engine and exhaust of the helicopter and oil rich air/fuel mixture from the helicopter exhaust. To house all the electronics within the constraint space and to minimize electromagnetic interference (EMI) is indeed a challenge. The wireless modem in the GHz frequency range that radiates strong radio

signal, GPS receiver that needs to receive the GPS signal, on board industrial computer system that needs to communicate with various input/outputs devices and remote control system that needs to control the helicopter. Interference of any of the mentioned system will have a great impact on the operation of the whole helicopter platform and it may result in the crashing of the helicopter.

With such a demanding factor on the system integration of the software and hardware system, utmost care is taken during the system integration process to increase the reliability and the robustness of the overall helicopter system.

4.1 Helicopter Platform

The helicopter used for this research is a commercially off-the-shelve (COTS) remote model helicopter, Raptor 60 from ThunderTiger company. The helicopter is made mostly from ABS composite and carbon fiber plates. As a higher payload and reliability are needed for this test platform, the helicopter is modified with a larger capacity engine, longer tail boom, longer main rotor blades and a stiffer swashplate system. Figure 4.1 shows the Raptor 60 model helicopter and the specification of the modified helicopter is given in Table 4.1

Figure 4.1 Raptor 60 model helicopter

Length	1.2m
Rotor diameter	1.5 m
Dryweight	6.0kg
Maximum Takeoff weight	10.0kg
Engine	OS-91SX Engine

Table 4.1 Specification of modified Raptor 60 helicopter

The model helicopter consists of a main rotor that is being driven by a glow engine which has a clutch that will engage the main rotor shaft upon throttling up the engine. The tail rotor is driven through a belt drive system to provide the necessary counter torque and is driven by the pinion that is geared to the main rotor shaft with a specific gear ratio. The control of the helicopter is actuated via 5 servos which control the main rotor blade collective, the longitudinal cyclic pitch, the lateral cyclic pitch, the tail rotor collective pitch and the throttle of the engine. Each of these servos is used to control each independent control surface and this makes the control of the helicopter easier unlike in some helicopters which use collective cyclic pitch mixing and have a more complex way of controlling the swashplate mechanism. Two commonly used sensors, the gyro and engine governor, are also installed on the helicopter to help the ease of

flying the helicopter. The Futaba GY601 gyro is used to help to stabilize the yaw dynamics of the helicopter by providing a negative feedback of the yaw rate to the tail rotor as shown in Figure 4.2.

Figure 4.2 Block diagram of feedback of tail gyro on yaw dynamics

This gyro is capable of Heading Lock function, in which it is able to lock the heading of the helicopter to a fix heading. This helps to ease the task of controlling the helicopter as the yaw motion is very fast due to the counter torque produce from the change of the main rotor RPM and the blade pitch angle (as the induced drag changes). Figure 4.3 shows the Futaba tail gyro that is used in the helicopter

Figure 4.3 Futaba tail gyro used in the helicopter

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The Futaba GV-1 engine governor unit is used to maintain the RPM of the main rotor to a constant value. This is done by having a hall effect sensor and a magnet to monitor the engine RPM. It performs a closed loop control of the engine RPM with the control of the throttle servo. Figure 4.4 shows the engine governor controller unit and Figure 4.5 shows the mounting of the hall effect sensor and the magnet.

Figure 4.4 Engine governor controller unit

Figure 4.5 Mounting of hall effect sensor and magnet.

4.2 Sensor System

The sensors that are needed onboard of the helicopter are responsible for collecting data needed for the system identification experimentation as well as for the flight controller design. According to the state space model that has been developed in Chapter 2, the data that are required to determine the helicopters states are the body linear velocities (u, v, w), the body attitude angles (θ, ϕ, Ψ), the body angular rates (p, q, r) and the body linear accelerations (a_x, a_y, a_z). Sensors are installed to collect these required data.

4.2.1 Inertia Measurement Unit.

The inertia measurement unit used onboard is a 0.7kg Crossbow DMU-HDX-AHRS as shown in Figure 4.6. This sensor is a strapdown type of inertia sensor that makes use of accelerometers, angular gyros and magnetometer to give a nine states reading of the platform: 3 angular rates (roll, yaw & pitch angular rates), 3 attitude angles (roll, yaw & pitch angles) and 3 linear acceleration measurements (X, Y & Z acceleration).

Figure 4.6 Crossbow DMU-HDX-AHRS sensor unit

The sensor data output is transmitted through a RS232 port. The data is output in a string of two byte values and customized C++ codes have been written to obtain these values. The data obtained are measured in the body coordinate system.

As the DMU is very sensitive to the vibration, care is taken when choosing the mounting location of the sensor. The vibration source is mainly from the moving parts such as the engine piston movement, gear transmission, belt drive system and both main and tail rotors rotation. Therefore, the mounting location chosen should be far from these vibration sources. In addition, the sensor is sensitive to magnetic field interference since there is a magnetometer inside the sensor. This will restrict the mounting location of the sensor to places that has low magnetic field interference and also places where it is surrounded by iron, which will shield the sensor from picking up the weak magnetic field of the earth to give accurate heading readings.

As the accelerometers and the gyros are made of solid state Micro Electro-Mechanical System, it is rather sensitive to temperature changes. To prevent the sensor data from drifting due to the drastic change in the temperature, the sensor cannot be placed near to the engine, engine cooling fan or the engine muffler where there is a drastic

temperature change. With these considerations, the DMU is mounted on the center of the undercarriage as shown in Figure 4.7.

Figure 4.7 Mounting of the DMU sensor on the undercarriage.

The DMU is mounted on the undercarriage by four screws and. A passive rubber damper is added between the mounting of the sensor and the mounting base so as to reduce the high frequency vibration noise on the sensor. At the same time, the rubber damper acts as a shock absorber, just in case the helicopter has a crash landing. Thin balsa wood is added on the side of the DMU so as to deflect the warm air from the engine cooling fan from directly blowing at the sensor which will heat up the sensor, contributing to the drifting error in the data.

As the DMU is mounted at an offset position from the center of gravity (CG) of the helicopter, this results to an offset error in the sensor data. Mounting the sensor at the exact CG of helicopter is quite impossible and impractical. First, at the CG of the helicopter, there may not be a mounting space for the DMU. Second, when additional

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payload is added on the helicopter, the CG position will change and it is impractical to shift the mounting location of the sensor when there is a CG shift. Hence, kinematics equation is used to correct the offset error due to the offset of CG.

Consider the DMU is mounted in the offset location described by the position vector $\mathbf{r}$, where the origin of the vector $\mathbf{r}$ is the CG of the fully integrated helicopter system as shown in Figure 4.8.

Figure 4.8 Offset of the DMU from the CG of helicopter

Taking the DMU as a fixed rigid point moving relative to the inertia frame of the helicopter, the measured acceleration of the DMU is given by

$$\mathbf{a}_{meas} = \mathbf{a}_{cg} + \boldsymbol{\omega} \times (\boldsymbol{\omega} \times \mathbf{r}) + \dot{\boldsymbol{\omega}} \times \mathbf{r} \quad (4.1)$$

From Equation 4.1, the measured acceleration is biased by a centripetal acceleration $\boldsymbol{\omega} \times (\boldsymbol{\omega} \times \mathbf{r})$ and a tangential acceleration $\dot{\boldsymbol{\omega}} \times \mathbf{r}$. With the location of the DMU with respect to the CG of the helicopter known, the measured acceleration can be corrected

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for the effect of the offset in the mounting location of the DMU. This is simply achieved by subtracting the centripetal and tangential components from the measured acceleration value. As the centripetal acceleration component is usually very small as compared to the tangential acceleration component, it is neglected in this analysis. To further simplify the calculation of the tangential acceleration component, the mounting of the DMU is directly below the main rotor hub of the helicopter and the lateral and the longitudinal CG will always be balance back to the neutral axis about the main rotor hub. This practice has physical significance as it will balance the helicopter in static by the weight of the helicopter. This means that the helicopter will not be any nose heavy, tail heavy or laterally one side heavy and this makes the helicopter easier to fly.

With this set up, the positional vector $\mathbf{r}$ is reduce to a vertical offset vector $[0,0,h_{cg}]^T$.

Hence the tangential acceleration bias is given by

$$\begin{aligned} \mathbf{a}_T &= \dot{\boldsymbol{\omega}} \times \mathbf{r} \\ &= \begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{bmatrix} \times \begin{bmatrix} 0 \\ 0 \\ h_{cg} \end{bmatrix} = \begin{bmatrix} \dot{q}h_{cg} \\ -\dot{p}h_{cg} \\ 0 \end{bmatrix} \end{aligned} \quad (4.2)$$

Therefore, the data of the measured accelerations a_x and a_y will be de-trend with this bias of the tangential accelerations of $\dot{q}h_{cg}$ and $\dot{p}h_{cg}$ respectively.

4.2.2 Global Positioning system

Global Positioning system (GPS) provides three dimensional position and time in which the estimates of the velocity and heading are deduced. The GPS receiver used is

the Trimble SKII board. It uses an active antenna that operates in the L band frequency of 1575.42 Mhz.

Figure 4.9 Trimble SK II GPS board.

The data refresh rate of the GPS is 1 Hz, which is rather slow as compared to the DMU data rate. At such a low data rate, the GPS seem not to be useful for the overall data acquisition system. However, the role of the GPS is used as a complementary to the DMU in helping to overcome some of the limitations of the DMU and at the same time improves on the overall accuracy of the data measured.

The DMU measures the inertia motion of the helicopter and this can be developed to a full Inertia Navigation System (INS) through the numerical integration of the data provided by the DMU to give the data required for inertia navigation. The data required for inertia navigation includes the positional, velocity and attitude data. Currently, the DMU is providing the attitude data (roll, pitch and yaw angles). To get the positional and velocity data, the acceleration data from the DMU is required to go through

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numerical integration and then transformed to the spatial coordinates using Euler transformation as the data from the DMU is measured in the body fixed axis. However, as the data provided by the DMU is contaminated by noise, bias and drift error, the inertia estimates obtained from the numerical integration will diverge very quickly as the error will grow unbounded as time lapses get longer. Hence, external information is required to compensate and correct the result of the numerical integration through constant update of more accurate data so that it will keep the error bounded all the time. A simple approach is to have the INS information be constantly updated by the velocity and positional estimate from the GPS at a refresh rate of 1 Hz so that the error of the INS is bounded.

With the implementation of the INS /GPS coupling, it introduces a constraint to the mounting of the antenna of the GPS. If the GPS and the DMU is mounted at two different locations, there is always an offset between the two sensors and this will introduce additional error into the system. Hence, it is desired to locate the DMU and GPS antenna as close together to each other as possible. In addition, another constraint to the mounting of the GPS antenna is that it should be located as far away from noisy radio signal environment as the GPS signal is usually very weak. Figure 4.8 shows the mounting location of the GPS antenna.

Figure 4.10 Mounting of GPS antenna

4.3 Flight computer system

The flight computer system used onboard of the helicopter is PC104 standard computer, which has a footprint of 3.55" by 3.775". The flight computer system is made up of a CPU board, DC-DC power supply board, GPS board and Timer board. Solid-state chip disk is used on the flight computer as the mass storage device. These boards are interconnected together using the PC104 bus by stacking together on top of each other as shown in Figure 4.9.

Figure 4.11 Flight Computer System made up of PC104

The PC104 format follows the industrial standard computer and hence is chosen due to their smaller footprint, higher level of reliability and robustness as well as there is a wide variety of peripheral boards that can be added to it for other functions.

4.4 Ground Monitoring Station

The ground monitoring station is made up of a RC transmitter and a laptop with a wireless modem. The RC transmitter allows the RC pilot to control the helicopter from the ground and issue the commands for the flight test. The laptop with the wireless modem provides for the communication to the onboard flight computer for real-time downloading of the flight test data. Figure 4.10 shows the ground monitoring station.

Figure 4.12 Ground Monitoring station

4.5 Servo Actuator

The servo actuators that are used are servomotors which has a DC motor with a built in feedback circuit and the output of the shaft of the motor is connected to a high ratio gear box so as to produce high torque at the output of the gear box. The servomotor is driven by Pulse-width Modulate (PWM) signal so that it will move to a specific angular displacement to control the helicopter. The change of PWM signal from the reference point will indicate the amount of control input that is applied to each of the control. Hence the reading of PWM signal going to the servo will be used to determine the control input that is applied by the pilot during the system identification experimentation. In the later phase of the controller design, the PWM signal will be

generated to control the servomotor. Figure 4.11 shows the servomotor that is used in the helicopter.

Figure 4.13 Servomotor used in the helicopter.

4.6 Wireless Communication

The FreeWave wireless modem is used for the wireless communication between the helicopter and the ground monitoring station. The modem makes use of the 2.4 GHz frequency band for data transmission with a data throughput of up to 115.2kbps.

Figure 4.14 Freewave data modem used in the system

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During the system identification experimentation, the wireless communication helps to transmit real-time flight test data to the ground monitoring station so that the quality of the flight test data can be monitored while the helicopter is performing the required maneuvers. For the flight controller design testing, the real-time data transmission helps to monitor the health of the helicopter when it is being controlled by the onboard flight computer.

The modem is installed on the left side of the undercarriage. As the other onboard electronics are near to the modem, the antenna from the modem is placed at the tail boom of the helicopter through a RF cable extension so that the radio signal output will not cause interference to the onboard computer system, GPS antenna and the servo actuators.

Figure 4.15 Mounting of data modem antenna and modem.

4.7 Software

4.7.1 Ground Monitoring Station Software

In the system identification experimentation, flight test data are collected and transmitted to the ground monitoring station through the wireless data modem so that real-time data can be viewed from the ground monitoring station while the helicopter is flying. This will help to better monitor the flight test data collected and as well as better monitor the progress of the flight test. A real time Graphical User Interface (GUI) for the monitoring the flight test data was developed by Bing Jie [34].

The GUI developed is able to record and display flight data on a real-time basis. A maximum of 6 channels can be selected for display and the graphs can be zoomed in and out so that a better visual check can be done at the data plots. In additional, the data plots can be paused at any point during real-time recording as well as continues real-time plotting at a click of a button. All the real-time data that are displayed on the GUI are recorded to a file and it can be retrieved at a later phase for analysis. This GUI can also be used offline for flight data analysis by retrieving the flight data from previous flight. Figure 4.14 shows a screenshot of the GUI developed.

Figure 4.16 Screenshot of real time GUI developed used in the ground monitoring station.

4.7.2 Onboard Flight Computer Software

The onboard flight computer runs on the DOS operating system. For the system identification experimentation, C++ program codes are written for the system initialization, sensor checks, data collection, data processing and the transmission of the data collected to the ground monitoring station.

The codes for data collection and interfacing of the hardware and software were written using Borland C++. The data packets can be read by the ground computer upon printing '#' to the serial port object, and data receiving is paused upon printing '\$'. The onboard program is catered to data acquisition of flight test data from the Timer/Counter board and the DMU sensor unit and also communicates with the wireless modem to transmit all the data to the ground modem.

4.8 Overall System Integration

A special undercarriage is designed and built so as to house all the electronics and sensors that is necessary on board of the helicopter. Passive vibration isolation is taken into the design consideration so as to reduce the vibration transmission from the upper airframe to the undercarriage. This is achieved through the use of rubber shock absorbers. In addition, a float absorber is place on the base of the undercarriage to reduce the impact on the platform when it touches the ground during landing. Copper shielding foil is used on the main compartment that houses the electronics to prevent Electro-Magnetic Interference (EMI). To prevent the DMU sensor from heating up, a housing made of good insulator material (a composite of wood and high heat resistance plastic) is used to shield the sensor unit from the cooling fan outlet of the engine. Figure 4.15 shows the fully integrated helicopter that is used for the system identification flight test.

Figure 4.17 Fully integrated helicopter system

Chapter 5

System Identification

Mathematical models of dynamic system are useful in many areas and applications. In this project, the mathematical model developed is used to describe the dynamics of the model helicopter system as well as to provide a basis for flight controller design. In general, there are two approaches in the construction of mathematical dynamic model. The first approach is to use first principle modeling where the dynamic behavior of the system is obtained through the analytic approach by using the basic laws of physics (such as Newton's Law). The second approach is to use system identification where experiments are performed on the system to collect data and a model is obtained through the model fitting of the recorded data by assigning suitable numerical values to the parameters.

System identification approach in modeling the system dynamics is useful when the system is very complex such that it is impossible to obtain a reasonable model using the first principle modeling. In addition, the model obtained from the first principle modeling often contains a number of unknown parameters which are impossible to be determined without experimentation. However, system identification does have its limitation. The identified model has a limited validity such that it is valid within the envelope of identification only. In addition, for parametric identification, an appropriate model structure is required prior to the system identification process. In

certain application, it is difficult or impossible to measure some state variables that are important to the dynamic model. Lastly, data measurement usually contains noise and this will degrade the quality of model identified since the model accuracy is dependent on the data measured.

5.1 System Identification Problem Statement

The objective of the system identification is to identify the parameters in the linear-time invariant model of the helicopter that has been derived in Chapter 3 (Equations 3.56 and 3.57). This model identified is used for the controller design in the later phase.

5.2 System Identification Procedure

The procedure for carrying out system identification is illustrated in the flowchart in Figure 5.1. In general, system identification process begins with some prior knowledge of the system to be identified. This knowledge on the system can be some basic characteristics of the system such as the linearity and bandwidth of the system and the specific order of the dynamic equation or the values of the associated coefficients may not be known.

The input signals used in the system identification experiment to excite the system have a significant influence on the model identified. Proper design of experiment needs to be done prior to the execution of the experiment so that essential data required for the system identification can be collected using the right methodology. The success of the system identification process is very dependent on the quality of the test data collected.

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Hence, the flight test procedures and data collection procedures will be covered in greater details in the subsequent section.

After identifying the parameters in the model, verification needs to be done to evaluate the accuracy of the model obtained. This is done by comparing the time histories of the test data collected with the output predicted by the identified model.

Figure 5.1 Schematic flowchart of system identification procedure

Figure 5.2 Helicopter dynamics

5.3 Design of experiment

It is important to define a sequence of excitation signals to be input to the helicopter model so that the dominant modes and cross-coupling effects in the helicopter dynamic can be identified. As the helicopter dynamics is non-linear and unstable in open-loop, the input signals chosen to excite the helicopter must be bounded within a region such that the helicopter will still stay in a stable hover mode. The design of experiment has to be carried out to ensure the flight tests are carried out within this bounded limit.

The helicopter dynamics, which is described by equations 3.56 and 3.57, can be grouped into 3 sub dynamics groups such that there is no cross-coupling effect between each subgroup response. The purpose of dividing into 3 decoupled dynamics subgroups is to facilitate the task of flight test data collection and system identification to be executed in 3 smaller subset tasks. This will help in the practical execution of flight testing and system identification process. The 3 sub dynamics groups are coupled roll and pitch, heave and yaw dynamics as illustrated in Figure 5.2.

5.4 Flight Test Procedures and Execution

In the collection of flight test data used for system identification, flight maneuvers were performed by an external pilot using a RC transmitter. The flight test was conducted in open loop except for the yaw axis where a yaw gyro is used to provide yaw damping to the system.

Special flight test maneuvers are performed to collect data used for system identification. Three different maneuver test points are used so that different dynamics of the helicopter can be identified in a decoupled manner, which has been discussed in Section 5.3. Table 5.1 shows the three flight test points.

Test Point	Operating Point	Control Stick Input
Roll & Pitch	Constant heading & height	$\delta_{lon}, \delta_{lat}$
Heave	Constant attitude	δ_{col}
Yaw	Constant roll and pitch angles & height	δ_{ped}

Table 5.1 Hover test points for system identification

To execute the roll and pitch maneuver, the pilot will pull the longitudinal and lateral sticks simultaneously for the helicopter to roll and pitch while the heading and the height of the helicopter remain unchanged. This is to capture the coupled roll and pitch dynamics. In the heave maneuver, the pilot will change the collective stick of the helicopter to allow the helicopter to ascent and descent while the heading and the roll and pitch angles remain constant. This will capture the heave dynamics. For the yaw maneuver, the tail rotor pitch is changed so that the heading of the helicopter will change while the height, roll and pitch angles of the helicopter remain unchanged.

5.5 Flight test data collection

In each of the flight maneuver, the external pilot applied a step input of 10 to 15 percents control authority to a particular control input stick for duration of 1.5 seconds. In order to keep the helicopter in the operating condition during each of the maneuver, the pilot used the other three control inputs to trim the helicopter. The same flight maneuver was repeated 3 times to collect enough data for the system identification.

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With the setup of the helicopter system and ground monitoring station as covered in Chapter 4, flight test data were collected. The data collected, which are the helicopter states and pilot stick inputs, were recorded. These helicopter states are the body linear velocities (u, v, w), the body attitude angles (θ, ϕ, Ψ), the body angular rates (p, q, r) and the body linear accelerations (a_x, a_y, a_z). The pilot control stick inputs are δ_{lat} , δ_{lon} , δ_{col} and δ_{ped} . The quality of the flight test data collected was evaluated at the ground monitoring station through the use of the real-time monitoring GUI as shown in Figure 5.3. Various data plots were used to ensure that the flight test data collected has a high coherence with the required flight maneuver.

Figure 5.3 Real-time monitoring of flight test data

Before the flight test data was used in the system identification, an offline pre-processing of the data was done. These include filtering the angular rates with zero-phase non-causal filter and removing the bias value from the data. The zero-phase non-causal digital filter filters out the high frequency noises without introducing phase delay. The bias value from

the attitude angles and control inputs were removed with the offset values due to the bias in the trim states during the flight test.

5.6 System Identification

The helicopter dynamics is a MIMO model. The state-space model structure is used to describe the helicopter dynamics. The Matlab System Identification Toolbox is used for identification of the unknown parameters in the state-space model. Two estimation methods are available in the toolbox, the PEM (Predictor-Error Method) and N4SID estimation. The PEM is a standard prediction error that seeks to minimize the quadratic error between the predicted model value and the flight test data based on iterative minimization of a criterion. The N4SID is a subspace-based method that does not use iterative search. Details of the N4SID algorithms used will not be covered but can be found in [4] and [15]. The PEM estimation is used in for the parameters estimation instead of the N4SID as it is simple and is a more general estimation method.

5.6.1 Identification Result

The system identification process was divided into 3 separated portions as discussed in the design of experiment. Different sets of flight test data were used for each of the dynamics identification as discussed in section 5.3 and 5.4.

5.6.1.1 Yaw Dynamics Identification

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The yaw dynamics was identified first where the parameters to be identified are the stability derivative, N_r and the control derivatives N_{col} and N_{ped} . Figure 5.4 shows the identified model plotted against the flight test data. The fitting between the identified data and the flight test data was computed using the Matlab's System Identification Toolbox '**compare**' command. This command computes the percentage fitting between the identified model and the flight test data fitting. A 100% fitting will indicate a perfect fitting between the identified model and the flight test data used. In this project, a fitting of 75% and above will be considered a good matching between identified model and the flight test data. A very good fitting of 91.92% was obtained for the yaw model.

The identified parameters are:

$$N_r = 0.30926$$

$$N_{col} = 0.043484$$

$$N_{ped} = -6.5782$$

Figure 5.4 Identified yaw model output

5.6.1.2 Heave Dynamics Identification

In the heave dynamics model identification, the parameters to be identified are stability derivative Z_w and control derivative Z_{col} . The heave velocity, w , is used as the input state for the identification process. This velocity data is provided by the onboard GPS with a refresh rate of 1 Hz. As the data sampling rate of the system was set at 33 Hz, the velocity data obtained from the GPS will display a zero-order hold equivalent when it was plotted with a sampling rate of 33 Hz as shown in Figure 5.5 since a zero-order hold equivalent was used. An IMU / GPS filtering algorithm using the acceleration data from the IMU to predict the velocity data between the intervals of GPS updates was attempted. However, due to the poor signal to noise ratio (SNR) of the Z acceleration data, there was

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a poor match between the velocity profile and the GPS data. The poor SNR of the Z acceleration data is due to the bias of 1g in the Z axis due to the inherent gravitational force as well as the vibration noise that the sensor picked up. With the small amplitude of acceleration in the Z axis and high vibration noise from the helicopter airframe due to engine and rotor vibration, the perturbation in the Z acceleration due to the flight maneuver can hardly be differentiated from the noise level that was picked up by the accelerometer. It was decided to use the GPS data directly without using the IMU / GPS coupling algorithm.

As the GPS is known to provide a more accurate data (up to 0.2 m/s accuracy) in X and Y axes as compared to the Z axis (0.5 m/s accuracy), the w velocity data obtained from the GPS is considered to have a poor accuracy as shown in Figure 5.5. In order to reduce the error due to the poor accuracy during the identification, the data was curve fitted to reduce the average error between the data samples. A fourth order polynomial equation was used to curve fit the data as shown in Figure 5.5.

Figure 5.5 Approximation of heave velocity using curve fitting

Subsequently, identification was carried out using the curve fitted data and a fitting of 82.7% was obtained for the heave model as shown in Figure 5.6.

The parameters identified are:

$$Z_w = 0.98902$$

$$Z_{col} = 0.3207$$

Figure 5.6 Identified heave model output

5.6.1.3 Roll and Pitch Dynamics Identification

The identification for the coupled roll and pitch dynamics is the most complicated as there are many coupled parameters to be identified and many state input variables are involved. From equation 3.57, 8 state and 2 control input variables are required. Of all these variables required, the rotor blade flapping angles, **a** and **b** cannot be measured. Since these 2 state variables are not measurable and are not a critical state variable in the 6 DOF equations to be controlled in the controller design phase, they will be treated as an internal state and identification of the stability derivatives related to these 2 state

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variables will not be critical. However, as the control inputs and the angular dynamics to the roll and pitch dynamics are directly affected by these 2 state variables, equation 3.57 has to be modified to accommodate the use of internal states without affecting the result of the identification on other variables.

From equations 3.39 and 3.40, the flapping angle states can be express as a function:

$$\mathbf{b} = f(p, q, \mathbf{a}, \delta_{\text{lat}}) \quad (5.1)$$

$$\mathbf{a} = f(p, q, \mathbf{b}, \delta_{\text{lon}}) \quad (5.2)$$

The flapping angle states, $\mathbf{b}$ and $\mathbf{a}$ are coupled together with each other.

Changes to stability derivatives

The stability derivatives L_b , L_a , M_b and M_a in the angular rate dynamics equations, which are linked to the flapping angle states $\mathbf{a}$ and $\mathbf{b}$, will be replaced by four new parameters that are linked with the angular rate state p and q as the flapping angle state.

The stability derivatives, X_a and Y_b that are linked to the flapping angle states for the velocity dynamics u and v , will be removed and the effect of the change of the flapping angle states on the velocity dynamics will be introduced through the control derivatives.

Changes to control derivatives

The input to the lateral stick input (δ_{lat}) will change the flapping angle state $\mathbf{b}$ and will affect the flapping state $\mathbf{a}$ as they are coupled together. The same apply to the longitudinal stick input (δ_{lon}) where the flapping angle state $\mathbf{a}$ will be changed by this stick input and this will affect the flapping state $\mathbf{b}$. Therefore, it can be deduced that a

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change in either of the flapping angle state **b** or **a** can be represented by the coupled effect of lateral and longitudinal stick input, δ_{lat} and δ_{lon} .

$$\mathbf{b} = f(\delta_{lat}, \delta_{lon}) \quad (5.3)$$

$$\mathbf{a} = f(\delta_{lat}, \delta_{lon}) \quad (5.4)$$

Since the flapping angle states **b** and **a** are internal states that cannot be measured and identification of the rest of the parameters involved the helicopter states (p, q, u, v) that are perturbed by flapping angle states **b** and **a**, the flapping angle states will be replaced by a combination of the input stick signal, δ_{lat} and δ_{lon} . Hence, new parameters will be assigned to the identification of the control derivatives matrix, replacing the original 2 parameters B_{lat} and A_{lon} that are assigned to the **b** and **a** flapping angle states only. The modified roll and pitch state-space model to be identified is:

$$\begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{\phi} \\ \dot{\theta} \\ \dot{u} \\ \dot{v} \end{bmatrix} = \begin{bmatrix} L_p & L_q & 0 & 0 & L_u & L_v \\ M_p & M_q & 0 & 0 & M_u & M_v \\ 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & -g & X_u & 0 \\ 0 & 0 & g & 0 & 0 & Y_v \end{bmatrix} \begin{bmatrix} p \\ q \\ \phi \\ \theta \\ u \\ v \end{bmatrix} + \begin{bmatrix} B_1 & B_2 \\ B_3 & B_4 \\ 0 & 0 \\ 0 & 0 \\ B_5 & B_6 \\ B_7 & B_8 \end{bmatrix} \begin{bmatrix} \delta_{lat} \\ \delta_{lon} \end{bmatrix}$$

The identification of the coupled roll and pitch dynamics was carried out in 2 steps. The first step was to identify the parameters that are related to the angular dynamics p and q , which are L_b , L_a , M_b and M_a . This was done first as the angular rate dynamics are more stable and is a dominant dynamics. At the same time, the control derivatives, B_1 , B_2 , B_3

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and B_4 were identified as well. Subsequently, the parameters with the horizontal dynamics were identified next. These parameters are L_u , L_v , M_u , M_v , X_a , Y_b , X_u and Y_v .

Figure 5.7 shows the identified result of the roll and pitch model with the flight test data.

The fitting of the p , q , u and v are 77.33%, 67.91%, 77.46% and 63.78%.

The parameters identified are:

$$L_p = 0.7823$$

$$L_q = -0.0632$$

$$M_p = 0.0374$$

$$M_q = 0.6981$$

$$L_u = -3.582 \times 10^{-3}$$

$$L_v = -4.615 \times 10^{-3}$$

$$M_u = -5.439 \times 10^{-3}$$

$$M_v = -6.995 \times 10^{-3}$$

$$X_u = 0.9927$$

$$Y_v = 0.9926$$

$$B_1 = 1.380$$

$$B_2 = 0.2097$$

$$B_3 = 0.8653$$

$$B_4 = 1.603$$

$$B_5 = -4.116$$

$$B_6 = 3.273$$

$$B_7 = 6.849$$

$$B_8 = -6.921$$

Figure 5.7 Identified roll and pitch model output

5.6.2 Full identified model

The full identified model for the **A** and **B** matrices are:

$$A = \begin{bmatrix} 0.7823 & -0.0632 & 0 & 0 & -0.0035817 & -0.0046147 & 0 & 0 \\ 0.0374 & 0.6981 & 0 & 0 & -0.0054394 & -0.0069948 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & -9.81 & 0.99274 & 0 & 0 & 0 \\ 0 & 0 & 9.81 & 0 & 0 & 0.99262 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0.98902 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0.30926 \end{bmatrix}$$
$$B = \begin{bmatrix} 1.3795 & 0.20969 & 0 & 0 \\ 0.86529 & 1.6047 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ -4.1164 & 3.2734 & 0 & 0 \\ 6.8492 & -6.9214 & 0 & 0 \\ 0 & 0 & 0.3207 & 0 \\ 0 & 0 & 0.04348 & -6.5782 \end{bmatrix}$$

5.6.3 Model Validation

Time domain verification of the model obtained was conducted by comparing the identified model with the flight test data not used in the system identification process. The plots for the identified model were obtained using the identified 6 DOF model with inputs from the measured states and control variables from the same flight test data used for validation. These plots are shown in Figure 5.8 to 5.13 where the flight test data is shown in green solid line and the identified model response is shown in blue dotted line.

Figure 5.8 Angular rate dynamics p

Figure 5.9 Angular rate dynamics q

Figure 5.10 Yaw rate dynamics r

Figure 5.11 Lateral velocity dynamics u

Figure 5.12 Lateral velocity dynamics v

Figure 5.13 Vertical velocity dynamics w

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The identified model shows a good matching with the flight test data although there are minor deviations in some of the intervals of the response. In general, the identified model shows quite a good representation of the actual helicopter response.

The eigenvalues of the identified model are shown in Table 5.2. The system has mostly unstable eigenvalues except for one in the Roll & Pitch mode. This explains why the helicopter dynamics is unstable in the Roll & Pitch, Heave and Yaw modes. All the 3 modes are quite well damped with damping ratios near to 1.

Mode	Eigenvalue	Damping	Frequency (rad/s)
Roll & Pitch	-0.0009	1	0.0009
	0.0093	1	0.0093
	0.7148±0.0819I	0.7195	0.9935
	0.9910	1	0.9910
	1.0367	1	1.0367
Heave	0.9910	1	0.9910
Yaw	0.3093	1	0.3093

Table 5.2 Eigenvalues of the identified helicopter system

With the unstable dynamics in the Roll & Pitch, Heave and Yaw modes, a robust flight control system has to be designed to stabilize the unstable helicopter dynamics.

Chapter 6

Flight Control System Design

The controller design for the helicopter without any mathematical model is a difficult task because the helicopter dynamics is unstable. Manual tuning of the controller parameters during experimental flights is a dangerous and time-consuming process and this is limited to only classical SISO controller design method. In the previous chapter, the helicopter dynamics had been identified using system identification technique. Having a mathematical model permits the use of sophisticated design methods for multivariable systems. The controller can be easily parameterized and verified in simulation before any real flight test is conducted.

In order to provide a working autopilot system for the helicopter, a model based controller design approach was taken as what had been done in the previous chapters where the helicopter dynamic model had been defined and identified through system identification. Non-model based controller designs, such as the fuzzy logic controller [36] has been applied on the control of the helicopter. Although this is an attractive method as no helicopter dynamic model needs to be defined for the controller design, this approach does not guarantee a working controller while it is being tuned based on the fuzzy rules.

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The classical control method is attractive as it is simple to use and implement. However, the classical controller design has its limitation as it is applicable to SISO system only. The basic of classical control for the aircraft control system is achieved by the successive loops closure to stabilize the aircraft. Usually, inner rate feedback loops are used to reduce the state parameters variation while other compensators, such as derivatives and integral actions, are added on the outer loops to stabilize and reduce the steady state error of the system. This design procedure becomes increasingly difficult when more loops are added to the system in the multivariable system with multiple inputs and outputs. Applying classical control theory will require the successive closure of individual loops which will involve a significant amount of time in the trial and error process and may not guarantee a successful controller.

The modern control theory is directly based on the state variables model which contains the system input-output information [37]. The modern control system design will eliminate the needs of trial and error at each of the loop by having a matrix equation that contains the control gains for the controller. The matrix equation can be solved readily and has all the control gains computed simultaneously so that all the loops close at the same time unlike the classical control where the loop close individually in successive manner. This will bring a faster cycle time and easier task for controller design. However, the design and implementation of the modern control theory is much more complex and difficult.

In this chapter, a flight control system will be designed for the helicopter to achieve a stabilized hover flight. Both the classical and modern control theories will be used such

that the helicopter autopilot system will consist of a hybrid of classical and modern controllers.

6.1 Problem statement for Flight Control System Design

The objective of the flight control system design is to provide a simple and practical working autopilot for the helicopter to achieve a stabilized hover flight mode. As the autopilot is designed to perform hover flight, the emphasis of the design is on the specification on steady state error and disturbance rejection that the autopilot can achieve rather than the dynamic response by the autopilot with a command input. At the same time, the controller must be easily implemented and robust enough to handle light gust disturbance conditions.

6.2 Approach to controller design

To achieve a hover autopilot where the helicopter stays in hover, the roll, pitch and yaw attitude angles must be held constant so that the helicopter can stay in level flight position. At the same time, there should be no translational velocity movement in the X, Y and Z direction with respect to the body axis of the helicopter. To fulfill this condition, attitude and velocity controllers have to be designed to control the helicopter. A precise position hover is not required for this project and hence position controllers will not be implemented in this autopilot design.

As the helicopter system is a MIMO system, to use a classical controller design which is applicable to the SISO system only, the helicopter dynamics has to be fully

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decoupled into SISO subsystems. This can be done by ignoring the coupled dynamic effect among the subsystems. However, the coupling effect can be quite substantial, especially in the case of the roll and pitch dynamics. Ignoring this coupled effect will have a significance impact on the robustness of the controller that can be designed. On the other hand, having a full MIMO system controller design using modern control theory will be more complex and difficult to implement. Hence, the approach taken for the autopilot design is to incorporate a hybrid of classical control design as well as the modern control design. The motivation of this approach is to make the controller design and implementation as simple and easy as possible.

The MIMO helicopter dynamic model will be decoupled into subsystems consisting of SISO and MIMO models to facilitate the use of hybrid controller design. It is intuitive for the model to be decoupled into the 3 subsystems which is similar to the system identification process in the previous chapter.

The autopilot system design will be divided into 3 parts, consisting of the controller design for the 3 decoupled subsystem, roll & pitch, heave and yaw dynamics subsystem. Figure 6.1 shows the architecture of the proposed controller design.

Figure 6.1 Architecture of the proposed controller design.

6.3 SISO Controller Design

The SISO controller design can be easily achieved by using a classical Proportional-Integral-Derivative (PID) controller to obtain the desired system response through feedback system.

6.2.1 PID Controller Design Theory

Figure 6.2 PID Controller with unity feedback

The PID controller has the transfer function:

$$K_p + \frac{K_I}{S} + K_D S = \frac{K_D S^2 + K_p S + K_I}{S}$$

where K_p is the proportional gain, K_I is the integral gain and K_D is the derivative gain. As shown in Figure 6.2, the difference between the desired input value and the actual output Y is represented by the tracking error e . This tracking error is sent to the PID controller where the controller will compute the derivative and the integral of this error. Hence, the signal to the plant, u , after passing through the PID controller will be equal to the proportional gain times the magnitude of the error plus the integral gain times the integral of the error plus the derivative gain times the derivative of the error.

$$u = K_p e + K_I \int e dt + K_D \frac{de}{dt}$$

This signal u will be sent to the plant and a new output Y will be obtained. The new output Y will be sent back to through the feedback loop to generate a new error e and the controller will take this as new error and this process goes on and on.

The proportional controller (K_p) will have the effect of reducing the rise time but cannot eliminate the steady-state error of the system. An integral control (K_I) will have the effect of eliminating the steady-state error but it makes the transient response of the system worse with the increase of settling time. A derivative control (K_D) will have the effect of increasing the stability of the system with the reduction of overshoot and lowering the settling time. However, with the combination of PID terms, the effect of

each gain term on the system are dependent on each other and changing any of these terms will affect the other two terms.

6.2.2 Heave Controller Design

The heave control of the helicopter will be controlled by using a PID controller as the heave dynamics is a SISO system where the vertical velocity of the helicopter is controlled by the collective stick control. From the identified state-space equation in previous chapter, the transfer function of the heave dynamics is calculated to be:

$$\mathbf{G}_{\text{Heave}} = \frac{0.3207}{s - 0.98}$$

The Matlab SISO Design Toolbox is used to design the heave controller. This SISO design toolbox is an interactive graphical user interface that facilitates the design of compensators for SISO feedback loop, where the root locus and bode plot will help in the selecting the gains of the controller. Figure 6.3 shows the SISO design toolbox GUI interface. The heave controller design is desired to have a closed-loop response of steady-state error of less than 5 percents and a settling time of less than 2 seconds.

Figure 6.3 Matlab SISO design toolbox GUI interface

The step response test signal is used for tuning of the PID controller using the toolbox.

Figure 6.4 shows the closed-loop response of the selected controller that satisfied the design parameters.

Figure 6.4 Closed-loop response of heave dynamics with the designed controller

The designed PID controller for the heave dynamics is given by transfer function:

$$\frac{55.1S^2 + 661.2S + 1983.6}{S}$$

6.4 MIMO Controller Design

The modern control technique provides a direct way of designing multi-loop controllers for MIMO system by closing all the loops simultaneously. However the

fundamental of the modern control is that the system must be both controllable and observable. Therefore this criteria will be tested prior to the design of the controller using modern control theory. The Linear Quadratic Regulator (LQR) controller design is a popular approach in the design of robust controller for multivariable aircraft control systems where there is a full states information feedback of the system [37].

6.4.1 Controllability and Observability

6.4.1.1 Controllability

The state-space equation in equation 3.56 with the dynamic states and input matrices **A** and **B** are said to be controllable if any initial state $x(0) = x_0$ and any final state x_1 , there exist an input sequence of finite length transfers x_0 to x_1 in finite time[38].

For a n -dimensional pair, (**A,B**) is controllable if the $n \times np$ controllability matrix c has rank n , where $c = [B \quad AB \quad A^2B \quad \dots \quad A^{n-1}B]$.

6.4.1.2 Observability

The state and measurement matrices **A** and **C** are said to be observable if for any unknown initial state $x(0) = x_0$, there exist a finite integer $k_1 > 0$ such that the knowledge of the input sequence $u(k)$ and output sequence $y(k)$ from $k = 0$ to k_1 suffices to determine uniquely the initial state $x(0)$ [38].

For the n -dimensional pair, (**A,C**) is observable if the $nq \times n$ observability matrix O has rank n , where $O = [C \quad CA \quad \dots \quad CA^{n-1}]^T$.

6.4.2 LQR Controller Design Theory

The LQR method is a powerful technique for designing controllers for complex systems that have stringent performance requirements [39]. In the previous section, when the state-space system fulfills the conditions of controllability and observability, a stable and robust LQR controller can be designed on the system [40]. The objective of this controller design is to design an autopilot system that regulates certain states of the helicopter to zero while obtaining desirable closed-loop response characteristics. In this project, a state feedback LQR controller will be designed with the full state variables $\mathbf{x}$ available at the output measurement. Figure 6.5 shows the LQR with the state feedback.

Figure 6.5 LQR with state feedback

The measured output $y(t)$ corresponds to the signal that can be measured and therefore be available for control. The control objective is to make this signal as small as possible in the shortest amount of time. The helicopter dynamics is described by the linear-time state-space model:

$$\dot{\mathbf{x}} = \mathbf{A} \mathbf{x} + \mathbf{B} \mathbf{u}$$

The control will feedback in the form $\mathbf{u} = \mathbf{K}\mathbf{x}$ where $\mathbf{K}$ is a matrix of constant feedback coefficients to be determined by the LQR design. To control the helicopter for a hover flight, the regulator only need to stabilize the helicopter and has a good closed-loop time response. In this case, $\mathbf{u}(t)$ will have only pure feedback inputs and no other auxiliary input. The objective of the state regulation for the helicopter is to drive any initial condition error to zero to achieve a stable system. This is achieved by finding an optimal gain matrix $\mathbf{K}$ through the minimization of the quadratic cost function. The quadratic cost function is given by:

$$\mathbf{J} = \frac{1}{2} \int_0^{\infty} (\mathbf{x}^T \mathbf{Q} \mathbf{x} + \mathbf{u}^T \mathbf{R} \mathbf{u}) dt$$

where $\mathbf{Q}$ and $\mathbf{R}$ are the weighting matrices on the state and input states. This minimization of the quadratic cost function can be regarded as a minimum energy problem where the controller seeks to minimize both energies. However, decreasing the energy of the controlled output from $\mathbf{y}(t)$ will require a large control input and a small control input will lead to large controlled output. The relative magnitudes of $\mathbf{Q}$ and $\mathbf{R}$ selected will have to balance these conflicting goals with the trade-off requirements on the smallness of the state against the smallest of the input. The choice of $\mathbf{Q}$ and $\mathbf{R}$ will affect the time response in the closed-loop system. Hence, the challenge of the LQR controllers lies in the selection of the weighting matrices $\mathbf{Q}$ and $\mathbf{R}$.

6.4.3 Yaw Controller Design

The function of the yaw controller of the helicopter is to control a constant heading angle (heading hold) during the hover flight. The yaw rate dynamics is controlled by 2 control stick inputs which are the collective and pedal sticks. Since this is a multi-input and single output system, the LQR controller is used for this multiple inputs system. The state-space equation of the yaw dynamics is given by:

$$\begin{bmatrix} \dot{r} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} 0.309 & 0 \\ 1 & 0 \end{bmatrix} \begin{bmatrix} r \\ \psi \end{bmatrix} + \begin{bmatrix} 0.0435 & -6.578 \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \delta_{col} \\ \delta_{ped} \end{bmatrix}$$

$$\mathbf{y} = \begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix} \begin{bmatrix} r \\ \psi \end{bmatrix}$$

For the yaw dynamics to be controllable and observable, the rank of the controllability and observability matrix must have a rank of 2. The rank of both the controllability and observability matrices were found to be 2. This concludes that the yaw dynamics is both controllable and observable. To find the feedback gain $\mathbf{K}$ matrix for the LQR controller, Matlab was used to compute this value. Before the feedback gain matrix $\mathbf{K}$ can be computed, an initial estimate of the weighting matrices $\mathbf{Q}$ and $\mathbf{R}$ are needed. Diagonal identity matrices were used for both $\mathbf{Q}$ and $\mathbf{R}$ matrices as the initial estimate. Subsequently, Simulink model was constructed to simulate the system which will help to evaluate the gain matrix $\mathbf{K}$ obtained. During the optimization of the controller, different values of $\mathbf{Q}$ and $\mathbf{R}$ were used till a satisfactory response for the system is obtained. In this case, a zero heading angle is used as reference. Therefore the yaw controller will have to keep the heading angle to zero when the autopilot mode is

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engaged. This is commonly known as the heading hold mode. The autopilot system has to be robust enough to handle external disturbances such as wind gust. In addition, during the switch over from manual flight mode to hover autopilot mode, the autopilot must be able to manage the transition of state from a non-hover flight condition to a hover flight condition. A simulink model was constructed for the analysis of these 2 conditions. Initial conditions were added to the dynamic model to simulate the transition of manual to autopilot mode. By using a different initial condition from that required for hover mode, the designed controller has to control the system and bring it to the desired state variables to meet the hover condition. In addition, disturbances to the dynamic model can be represented by the inclusion of initial condition. The initial condition set can be seen as the disturbance affecting the state variables where the controller needs to bring these state variables to the hover condition again. Figure 6.6 shows the Simulink model used to evaluate the response of the yaw dynamics model with the LQR controller.

Figure 6.6 Simulink model for yaw dynamics with LQR controller

Figure 6.7 Response of closed-loop yaw dynamics with LQR controller

Figure 6.7 shows the closed-loop response of the yaw dynamics with the LQR controller after several iterations of using difference weighing matrices $\mathbf{Q}$ and $\mathbf{R}$. The matrices for $\mathbf{Q}$, $\mathbf{R}$ and $\mathbf{K}$ are:

$$\mathbf{Q} = \begin{bmatrix} 1 & 0 \\ 0 & 100 \end{bmatrix}$$

$$\mathbf{R} = \begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix}$$

$$\mathbf{K} = \begin{bmatrix} 0.0136 & 0.0661 \\ -2.08 & -10 \end{bmatrix}$$

The initial condition set for this simulation is yaw rate, $r_0 = 1.0$ rad/s and the heading angle, $\psi_0 = 0.3$ rad. The result shows that the controller is able to bring the yaw rate and heading angle to zero within 1 second, thus simulating that the controller is capable of controlling the yaw dynamics to the desired hover condition.

6.4.4 Roll & Pitch Controller Design

The function of the roll and pitch controller is to keep the helicopter at a horizontal attitude angles (roll and pitch angle at zero degrees) and achieving a zero linear velocities (both u and v). The roll and pitch dynamics is controlled by 2 control stick inputs which are the lateral and longitudinal sticks. The state-space equation of the roll and pitch dynamics is given by:

$$\begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{\phi} \\ \dot{\theta} \\ \dot{u} \\ \dot{v} \end{bmatrix} = \begin{bmatrix} 0.782 & -0.0632 & 0 & 0 & -0.00358 & -0.00462 \\ 0.0374 & 0.698 & 0 & 0 & -0.00544 & -0.00700 \\ 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & -9.81 & 0.993 & 0 \\ 0 & 0 & 9.81 & 0 & 0 & 0.993 \end{bmatrix} \begin{bmatrix} p \\ q \\ \phi \\ \theta \\ u \\ v \end{bmatrix} + \begin{bmatrix} 1.38 & 0.210 \\ 0.865 & 1.61 \\ 0 & 0 \\ 0 & 0 \\ -4.11 & 3.27 \\ 6.85 & -6.92 \end{bmatrix} \begin{bmatrix} \delta_{lat} \\ \delta_{lon} \end{bmatrix}$$

The rank of the controllability and observability matrix were computed using Matlab they were found to have a rank of 6. This means that the roll and pitch dynamics are both controllable and observable.

The determination of the **K** matrix follows the same procedure as the controller design procedure for the yaw dynamics described in the previous section. The parameters for the controller after the iterations of **Q** and **R** are as follows:

$$\mathbf{R} = \begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix}$$

$$\mathbf{Q} = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 2 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \end{bmatrix}$$

$$\mathbf{K} = \begin{bmatrix} -38.312 & 31.807 & -48.756 & 84.394 & -21.346 & -0.62251 \\ 29.425 & -26.887 & -424.2 & 405.19 & -60.082 & -71.578 \end{bmatrix}$$

The same analysis procedure is applied to the controller that has been designed where the initial conditions of the state variables have been set to non-zero. The initial conditions used for the state variables are:

$$\begin{bmatrix} p_0 \\ q_0 \\ \theta_0 \\ \phi_0 \\ u_0 \\ v_0 \end{bmatrix} = \begin{bmatrix} 1.0 \\ 1.0 \\ 0.5 \\ 0.5 \\ 2 \\ 2 \end{bmatrix}$$

Figure 6.8 shows the response of the roll and pitch dynamic of the system with the initial condition for the state variables as described above. The LQR controller implemented is able to bring all the 6 state variables to the zero within 10 seconds.

Figure 6.8 Response of closed-loop roll & pitch dynamics with LQR controller

6.5 Simulation of Autopilot System

The overall autopilot system for the hover flight will be the combination of the heave mode, yaw mode and roll & pitch mode controllers. Full model simulation of the autopilot system designed with the hybrid of the classical and modern controller is performed using the Matlab’s Simulink Toolbox. Figure 6.9 shows the block diagram

of the simulation model and Figure 6.10 shows the simulink model that was built for the simulation.

Figure 6.9 Block diagram of the simulation model

Figure 6.10 Simulink model used for simulation

External disturbances are used to disturb the system at a hover trim position to determine the effectiveness of the hybrid autopilot system on the full helicopter model. The autopilot is supposed to bring the helicopter back to a trim hover state under the excitation of external disturbances. This full model simulation will help to check for the effectiveness of the autopilot system in canceling out any of the cross-coupling effects that might be present in the full helicopter dynamic model. Nine different external disturbances to the helicopter dynamic states are used. These external disturbances used are the maximum limits that the helicopter will be operating in. The results of the close loop response of the helicopter system with the autopilot system are attached in Appendix A.

External Disturbance	Magnitude
ϕ	0.5 rad
θ	0.5 rad
ψ	0.3 rad
p	1 rad/s
q	1 rad/s
r	1 rad/s
u	2 m/s
v	2 m/s
w	2 m/s

Table 6.1 External disturbances used for simulation

The result shows that the hybrid autopilot is able to bring the helicopter from the disturbed conditions back to the original trim position in a typical duration of about 8 seconds.

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Appendix B shows the simulation results of open loop system response of the helicopter without the autopilot (in green) as compared with the closed loop system response of the helicopter with autopilot (in blue). The same external disturbances conditions in Table 6.1 are applied to both system. The simulation results show that the designed hybrid autopilot is able to stabilize the helicopter to achieve a trim hover flight. If there is no autopilot system (as shown in the open loop system response), the helicopter system will be destabilized with its states diverge from the trim condition. Hence, it can be concluded that the designed autopilot meets the design requirements of stabilizing the helicopter to a trim hover flight.

Chapter 7

Conclusion, Recommendation and Future Works

7.1 Conclusion and Recommendation

This project has presented the work done on the development of a small rotary-wing VTOL UAV based on a small remote-controlled model helicopter. The aim of this research is to gain a comprehensive insight into the development of a small UAV system. The field of disciplines involved in this project is very wide, which includes modeling of helicopter dynamics, UAV hardware system integration, system identification and flight control system design. Extensive effort was spent for the past 3 years, from the hardware setup, system identification and subsequently the flight control algorithms design.

In the modeling of the helicopter dynamics, the model proposed is adapted from the full-size helicopter theory as well as the literature proposed by other research work that has been done. This model has been very much simplified to adapt to a linear system for the ease of system identification and the flight control system design and implementation. To improve on the fidelity of the model used, a more detailed study on

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the helicopter flight dynamics on the small-scale helicopter should be done to eliminate the need of many engineering assumptions and include more cross-coupling effects into the model. The model used is critical to the success of the entire project as the system identification work and the flight controller design is based on the proposed mathematical model.

As the payload and size of the helicopter is limited, care must be taken in the selection of the onboard avionics system to be used. In addition, the harsh operating environment within the system will affect the reliability of the onboard electronics and this is crucial to the operation of the system. If any of the onboard electronics fails, there is a high possibility of the helicopter going out of control and thus crashing the helicopter. Several improvements can be made to the current system in the future. First, the vibration damping for the sensors can be further improved so that less vibration noise will be picked up by the sensors. Improving the vibration damping will help to increase the reliability of the onboard avionics system as well. Secondly, the current GPS used has a refresh rate of 1 Hz. A higher refresh rate GPS can be used so that it can give a better resolution for the velocity and position data. Barometer can be added into the system to improve on the current vertical velocity measurement.

Flight test data are collected using the instrumented platform to perform the required flight test points. The pre-processing of flight test data before the system identification process is critical to the success of the system identification. The signal noise of the system has to be filtered out so that the data measured will be able to represent the dynamics of the vehicle accurately. In this project, digital filters are used to remove the high frequency noise due to vibration using simple filter design. To improve the quality

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of flight test data collected in the future, more comprehensive filters, both analogue and digital filters, can be used. This will require a more comprehensive study and implementation of data acquisition system on the helicopter system.

The IMU/GPS coupling scheme is not used to obtain a more accurate linear velocities due to the poor signal to noise ratio (SNR) of the acceleration data obtained. The poor SNR of the data is due to the accelerometers picking up the vibration noise from the helicopter moving parts such as the main rotor and engine rotation. A better filtering algorithm and improvement to the vibration damping isolation scheme discussed previously can help to eliminate this problem. An alternative approach is to get COTS sensor unit that has build in INS/GPS solution. An example is the GuideStar 111m integrated sensor [41]. A disadvantage of such system is the high cost and export license that ties to the purchase of the product.

The time domain system identification has been successfully performed on the derived 6 DOF state-space dynamics equation. Verification has been done using the flight test data not used on the identification process and the results show quite a good trend matching between the predicted model and the actual flight test data. The time domain identification is used in this project as it is easier to carry out. However, frequency domain system identification can be used in the future to capture the different frequency dynamics and this will improve the fidelity of the model.

The objective of the flight control system design is to design an autopilot that is able to perform stabilized hover flight mode. A hybrid of PID and LQR controllers has been successfully designed and full model simulation has been done. Flight test can be carried out in the future to evaluate the effectiveness and robustness of the autopilot.

7.2 Future Works

Once the autopilot system can perform the stabilized hover flight mode, the flight envelope of the autonomous helicopter can be expanded to perform slow forward flight mode to aggressive maneuvers. This will involve more works to be done on modeling of the helicopter dynamics as well as designing a more robust controller to handle these aggressive maneuvers.

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APPENDIX A

SIMULATION RESULTS

Appendix A

Appendix A

APPENDIX B

SIMULATION RESULTS

Appendix B

Appendix B

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GLOSSARY

AHRS	Attitude and Heading Reference System
CG	Center of Gravity
COTS	Commercially Off The Shelf
DOF	Degree Of Freedom
EMI	Electro Magnetic Interference
GPS	Global Positioning System
GUI	Graphical User Interface
GUIDE	Graphical User Interface Development Environment
IMU	Inertia Measurement Unit
INS	Inertia Navigation System
LQR	Linear Quadratic Regulator
MIMO	Multi Input Multi Output
MTOW	Maximum TakeOff Weight
PEM	Prediction Error Method
PID	Proportional Integral Derivative
PWM	Pulse Width Modulate
RC	Radio Controlled / Remote Controlled
RPM	Revolution Per Minute
RTOS	Real Time Operating System
SISO	Single Input Single Output
SNR	Signal to Noise Ratio
UAV	Unmanned Aerial Vehicle