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Real-Time Helicopter Flight Control Test Bed

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Abstract

The goal of this paper is to describe a test bed for real-time helicopter flight control systems and some initial control experiments utilizing the test bed. The developed test bed consists of an off-the-shelf radio control helicopter which has been modified so that it can be computer controlled. The helicopter is mounted on a sensor equipped test fixture which provides information on the helicopter's position and attitude while allowing it to move freely within some limited area. In this paper we describe the hardware and software developed to allow real-time helicopter flight control and plant identification. In addition we propose a dynamic model for the miniature helicopter plant and use this model to design a linearized feedback controller for hovering.

1. Introduction

The control of helicopters has been recognized as being an important nonlinear control problem. As an ongoing project at the real-time robot control laboratory our goal is to develop a miniature flying vehicle with on board computers and navigational instruments capable of autonomous
flight. A number of applications have been identified in commercial and military surveillance which require a "stationary eye in the sky" for high-quality imaging. Applications include traffic watch, border surveillance, police suspect pursuit, and military target identification, and tracking. In these applications helicopters are able to provide continuous images of fixed or moving targets using accurate, narrow field of view cameras. This is a significant advantage over fixed wing surveillance aircraft which must make complicated maneuvers to observe stationary targets, and use wide-angle, or gimballed cameras to keep the target continuously in sight. It also provides a significant advantage over balloons or dirigibles which generally have insufficient speed to pursue moving targets, and must also use wide-angle cameras. The primary advantage of using a helicopter is that it enables the effective use of a narrow field of view camera which provides more detailed image resolution, and faster image processing than wide angle counterparts; thus, enabling the helicopter to fulfill a role in both high speed pursuit, as well as stationary target recognition.

This paper is organized into six sections; section 2 gives a brief summary of past work on helicopter control, section 3 describes our real time control system test bed, and section 4 describes the dynamic model of our laboratory helicopter. Control design and experimental results are given section 5 and a summary of the paper and a brief discussion can be found in section 6.

2. Past Work on Helicopter Flight Control

There has been tremendous interest in the control of military helicopters, particularly for stability augmentation control (SAC) in high performance piloted vehicles. Requirements for stability augmentation
have arisen because of the need to reduce pilot workload and to improve flying qualities during poor weather conditions, low level, and night time flight. A number of modern linear feedback control schemes have been applied to the stability augmentation problem. These include \( H_\infty \) design techniques, linear quadratic regulator designs, eigenstructure assignment techniques and feedback linearization techniques (see [4] and [5] for a review of these design techniques applied to helicopter SAC). Most of the past work involves obtaining a model based on linearizing about an operating point and then designing a suitable controller. Work done by Meyer, Hunt, and Su [6] is an exception to this. Their control design involves transforming the full nonlinear model into a constant, decoupled linear model from which classical control design methods can be applied. The resulting control law is then transformed back in terms of the available control variables. Nonlinear adaptive control techniques have also been applied to the control of helicopters by Prasad, et al (see [7] [8]).

In the area of miniature helicopter flight control there is hardly any published work with the exception of the work by Furuta, et al [9] and the control of a constrained helicopter-like vehicle by Kienitz, et al [10]. In addition, most of this research has been done on electrically powered vehicles, and under dynamic restrictions that are unrealistic for a free-flight scenario (such as fixing the collective pitch angle). Although electric motors make the dynamic model easier to derive, they are unsuitable for most reconnaissance-type applications because they severely limit the helicopter's range, lifting capability, and maneuverability due to substantial battery weight.

Our work has so far uncovered that it is difficult to obtain exact parameters for many of the nonlinear terms. Further there are other
unmodelled dynamics arising from the dynamics of the propulsion system, flexibilities and backlash in the mechanical linkages, dynamics of the servo actuators and sensor electronics. Under these circumstances, robust design techniques such as variable-structure and adapting neural network control techniques might be necessary to obtain robust flight control [12] - [14]. These control techniques are currently being investigated for flight control experiments using our test bed, with the eventual goal being to obtain an autonomous flying vehicle.

3. Real-Time Experimental Hardware System Development

The current system represents the first step towards achieving the goal of an autonomous helicopter flight control system. Figure 1 shows the initial system organization which consists of a microcomputer connected through a set of interface electronics to the helicopter control surface servo system. The helicopter is mounted on an instrumented flight stand which through various sensors measures the position, attitude, and rotational speed of the helicopter rotor.

In our test bed the Central Processing Unit (CPU), a Motorola 68000 microprocessor, controls a set of counter/timers which generate Pulse Width Modulated (PWM) control signals. These control signals are buffered by an interface and sent out to the helicopter servo system. Any resulting change in the helicopter state due to servo action will be measured by the sensors and fed back through the interface to the computer where analog feedback signals are converted to digital signals which are processed by the CPU according to the selected control law.

There are five servo mechanisms on the helicopter which allow attitude and throttle input control. Each of the servo mechanisms operates
as an independent, closed-loop positioning system for a particular control surface. Each servo is controlled by PWM input signal generated by a counter/timer. The CPU sends the desired servo position encoded as the width (in microseconds) of the PWM servo signal. The actual pulse is generated by one of the five counter/timer latches which automatically generate the pulses whose widths correspond to the digital word loaded in the timer latches. The pulses generated by the counters with a pulse frequency of 20 ms (50Hz) are buffered through the interface electronics and run through an umbilical cord out to the appropriate helicopter servo. The servo modules, themselves, are quite complicated and each contains its own, independent, closed-loop positioning system (see Figure 2).

The PWM inputs to the servo trigger an internal one-shot which generates a pulse of known width. The difference between the input pulse width and an internally generated pulse width becomes the command signal which is then subtracted from a feedback signal from a position potentiometer on the servo gear train. This error signal after compensation drives the servo motor. The servo output drive shaft is coupled to a helicopter control surface through (sometimes highly complicated) mechanical linkages. The servo system is capable of accepting commands at any time from the CPU, and in the absence of a CPU update, will self-generate commands to hold its current position. The positioning system has approximately 10 bit positional accuracy over the range of helicopter control surface motion.

System Software Organization

An interrupt is generated every 20 ms which initiates the data acquisition sequence. The interrupt service routine (isr) initiates the data
retrieval process by calling a series of routines that read the A/D converter and scales the inputs into appropriate state units (e.g. radians, radians/sec, etc). These converted values are stored in global variables for use in other parts of the software, or in data structures that can be uploaded to a mainframe for further analysis.

Once valid data has been updated, the control signals are generated based on the desired output state of the helicopter and the actual states. This control signal is then translated into the required servo commands and the appropriate pulse width counts are sent out to the timers. Currently, we have the ability to close control loops about rotor speed and hover altitude. From the system terminal menu we can also select different types of controllers for each loop (linearized state feedback, PID, neural network controller, etc), and adjust various control parameters. We also have the capability to select various test modes, which include manual control, sinusoidal responses, and open or closed loop step responses.

The helicopter chassis is an X-Cell model 50 radio control aircraft manufactured by Miniature Aircraft, Florida, USA. It is powered by a 0.5 in³ displacement two-cycle combustion engine made by Webra Model-Building Inc (Germany). The helicopter has five servo mechanisms which control the throttle, rotor collective pitch, cyclic pitch, and tail rotor pitch (cyclic and tail rotor pitch correspond to body pitch, roll, and yaw motions).

Figure 3 shows the helicopter mounted on a commercially available flight stand. There are three sensors mounted on the test bed setup. The first two are potentiometers which measure the helicopter altitude, and the rotor collective pitch angle, respectively. The third sensor is a magnetic tachometer used for measuring the rotor speed. A single magnet has been
mounted on the motor shaft and a hall effect sensor on the chassis so that it senses the magnet and generates a pulse each time the magnet goes by. The output signal from the hall-effect sensor is then run to a frequency to voltage (F/V) converter. The output of the F/V converter is then filtered and monitored on one of the analog input channels of the analog board.

4. Helicopter Plant Dynamics

Although miniature helicopters are functionally similar to their full-scale counterparts, there are several differences (mainly in rotor construction) which require modifications to the normal thrust equations used to model full scale helicopters. For example, our rotor blades are straight and do not have a linear twist as is the case for real helicopters (see [1] [2]). Another significant difference between our helicopter and a full scale helicopter is that our helicopter compensates for the lack of flapping and lead-lag hinges ([2] [3]) by using a teetering hinge which produces the same effect [3]. In the work described here we model thrust as a nonlinear function of the throttle, rotational speed, and collective pitch position. Our aim here will be to model the helicopter in vertical flight.

Modelling of Helicopter Dynamics in Vertical Flight

The following equations of motion describe our miniature helicopter in vertical flight:

\[ \ddot{z} = K_1(1+G_{eff}(z))C_T\omega^2 - g - K_2\dot{z} - K_3\dot{z}^2 - K_4 \]

(1)

where, \[ C_T = (-0.032592 + \sqrt{0.001062 + 0.061456\theta_c})^2 \]

(2)

\[ \dot{\omega} = -K_5\omega - K_6\omega^2 - K_7\omega^2\sin\theta_c + K_8u_{th} + K_9 \]

(3)
\[ \dot{\theta}_c = K_{10}(-0.00031746u_{\theta_c} + 0.5436 - \theta_c) - K_{11} \dot{\theta}_c \]  

where \( z \) represents the height above the ground in meters, \( \omega \) the rotational speed of the rotor blades in rad/sec, \( g \) the gravitational force in m/s^2, \( \theta_c \) the collective pitch angle of the rotor blades in radians and \( u_{th} \) and \( u_{\theta c} \) represent the input to the throttle and collective servo mechanisms, respectively. The first term on the right hand side of (1) is the main thrust term which is based on the blade element and momentum theories of vertical flight (see [1]-[3]). Here, \( G_{eff}(z) \) is the ground effect term. For our experimentations, this term is zero since the stand placed the helicopter rotor blades more than one diameter above the ground which takes the helicopter out of ground effect (see [1]-[3]). The second term is the force due to gravity, the third term takes into account damping in the flight stand especially due to the piston mounted to offset the weight of the stand, the fourth term represents the parasitic drag, and finally the last term represents constant drag. Equation (2) has primarily been derived from the work of Johnson [1] which relates the thrust constant, \( C_T \), to the collective pitch angle, \( \theta_c \). The exact coefficients were derived from rotor blade characteristics such as blade radius, cord length, number of blades, etc. The third equation represents the dynamic model of the two stroke combustion engine and the rotational velocity of the rotor blades. The form of this equation was determined through a number of experiments. The first term on the right is a damping term. The next two terms represent air foil drag losses. Finally the last two terms represent a linear approximation of the two stroke combustion engine and the effect of the throttle servo input, \( u_{th} \), to the rotational speed, \( \omega \). We should mention
that the characteristics of the combustion engine vary from day to day with changes in the air/fuel mixture, weather conditions, etc. Although we tune the engine each time experiments are performed, there is no way to guarantee that it will produce the same output for a given set of inputs (throttle position and collective pitch angle). As a result, we were only able to bound parameters $K_8$ and $K_9$ to within a range of values for which nominal values are given below. Equation (4) represents the collective pitch servo response to the input, $u_\theta_c$. The first group of terms represent a linear approximation of the relationship between the servo input and the resulting collective pitch in steady state and the last term represents the damping of the servo system due to the servo motor and linkages. Based on several parameter estimation experiments and least-squares error curve fitting techniques, we were able to come up with the following values for the parameters of our helicopter model for vertical flight:

$$
\begin{align*}
K_1 &= 0.25 \text{ m} \\
K_2 &= 0.10 \text{ s}^{-1} \\
K_3 &= 0.10 \text{ m}^{-1} \\
K_4 &= 7.86 \text{ m/s}^2 \\
K_5 &= 0.70 \text{ s}^{-1} \\
K_6 &= 0.0028 \\
K_7 &= 0.005 \\
K_8 &= 0.1088 \text{ s}^{-2} \\
K_9 &= -13.92 \text{ s}^{-2} \\
K_{10} &= 800.00 \text{ s}^{-2} \\
K_{11} &= 65.00 \text{ s}^{-1} 
\end{align*}
$$

**Linear Approximation of Helicopter in Vertical Flight**

Our initial flight control experiments involved the investigation of several linear controllers based on a linearized approximation of the nonlinear dynamics. In order to do this, we will find the linearized model of the nonlinear system about an operating point. This operating point will be determined by writing the above equations in state space form and setting all of the derivatives to zero. If we use the state space assignment
\[
\mathbf{x} = (x_1, x_2, x_3, x_4, x_5)^T = (\dot{z}, \dot{\omega}, \dot{\theta}_c, \ddot{\theta}_c)^T \quad \text{and} \quad \mathbf{u} = (u_{th}, u_{\theta_c})^T = (u_1, u_2)^T,
\]

(5)

we will get the following state space model

\[
\begin{bmatrix}
\dot{x}_1 \\
\dot{x}_2 \\
\dot{x}_3 \\
\dot{x}_4 \\
\dot{x}_5
\end{bmatrix} =
\begin{bmatrix}
x_2 \\
-K_1 C_T x_3^2 - g - K_2 x_2 - K_3 x_2^2 - K_4 \\
-K_5 x_3 - K_6 x_3^2 \sin x_4 - K_7 x_3^2 - K_8 x_1 - K_9 \\
K_{10}(-0.0003174 u_2 + 0.5436 - x_4) - K_{11} x_5
\end{bmatrix}
\]

(6)

where,

\[
C_T = (-0.032592 + \sqrt{0.001062 + 0.061456 x_4})^2.
\]

(7)

It can be seen that by setting the derivatives to zero and selecting the collective pitch about which we desire the operating point, the remaining operating conditions can be easily solved. In our case we chose a collective pitch angle of \(\theta_{co} = 7.16\) degrees = 0.125 radians which we know from experience is a good choice to obtain liftoff and hovering. The perturbation of the system from the operating point \((x_0, u_0)\) is such that \(x = x_0 + \Delta x\) and \(u = u_0 + \Delta u\). Then using a Taylor’s Series expansion about the operating point, we get

\[
\frac{dx_i}{dt} = \frac{dx_{i0}}{dt} + \frac{\Delta x_i}{dt} = f_i(x_0 + \Delta x, u_0 + \Delta u) =
\]

\[
= f_i(x_0, u_0) + \frac{\partial f_i}{\partial x_1} \Delta x_1 + \cdots + \frac{\partial f_i}{\partial x_n} \Delta x_n + \frac{\partial f_i}{\partial u_1} \Delta u_1 + \cdots + \frac{\partial f_i}{\partial u_m} \Delta u_m + \text{H.O.T.}
\]

(8)
where the partial derivatives are evaluated at the operating point \((x_0, u_0)\).

If we stay relatively close to the operating point, the higher order terms (H.O.T.) can be dropped to give us the following linearized model.

\[
\Delta \dot{x} = \left. \frac{\partial f}{\partial x} \right|_{(x_0, u_0)} \Delta x + \left. \frac{\partial f}{\partial u} \right|_{(x_0, u_0)} \Delta u = A\Delta x + B\Delta u
\]

(9)

where \(\Delta x\) and \(\Delta u\) are the perturbation from the operating states \(x_0\) and \(u_0\), respectively. Note, that \(A\) and \(B\) are constant matrices since the partial derivatives are evaluated at the operating point. The following linear model is a linearized approximation for our nonlinear system about the operating point, \((x_0, u_0) = (x_01, x_02, x_03, x_04, x_05, u_01, u_02) = (z_0, \dot{z}_0, \omega_0, \theta_0c, \dot{\theta}_0c, u_{0th}, u_{0\theta}c) = (z, m, 0 \text{ m/s}, 138.01 \text{ rad/s}, 0.125 \text{ rad}, 0 \text{ rad/s}, 1615.18, 1318.84)\).

\[
\begin{bmatrix}
\Delta \dot{x}_1 \\
\Delta \dot{x}_2 \\
\Delta \dot{x}_3 \\
\Delta \dot{x}_4 \\
\Delta \dot{x}_5
\end{bmatrix} =
\begin{bmatrix}
0 & 1.0 & 0 & 0 & 0 \\
0 & -0.1 & 0.256 & 190.64 & 0 \\
0 & 0 & -1.645 & -94.49 & 0 \\
0 & 0 & 0 & 0 & 1.0 \\
0 & 0 & 0 & -800.00 & -65.00
\end{bmatrix}
\begin{bmatrix}
\Delta x_1 \\
\Delta x_2 \\
\Delta x_3 \\
\Delta x_4 \\
\Delta x_5
\end{bmatrix} +
\begin{bmatrix}
0 & 0 \\
0 & 0 \\
0.1088 & 0 \\
0 & 0 \\
0 & -0.2539
\end{bmatrix}
\begin{bmatrix}
\Delta u_1 \\
\Delta u_2
\end{bmatrix}
\]

(10)

The resulting eigenvalues for the linearized system are the following: 0, -0.10, -1.65, -16.49 and -48.51. Although these poles are not in the right-half plane, the response of this system will be very slow. Also when looking at the eigenstructure, we see that the zero eigenvalue corresponds to the response of the vertical position, \(x_1\), which implies the vertical position will be affected by external disturbances such as wind, etc. As a result of the poor natural response, we have investigated several linear
control design techniques in order to obtain a well controlled response for the states of the system with particular interest being paid to the vertical position of the helicopter during hover.

5. Controller Design for Linearized System

With the linearized model determined, our next step was to design a state feedback controller, $K$, such that $A + BK$ has the desired stable eigenvalues [11] (see Figure 4 for organization of the control scheme). Our state feedback control law will be $\Delta u = K\Delta x$ where $K$ is the following:

$$
K = \begin{bmatrix}
K_{11} & K_{12} & K_{13} & K_{14} & K_{15} \\
K_{21} & K_{22} & K_{23} & K_{24} & K_{25}
\end{bmatrix}.
$$

If we design $K$ such that $A+BK$ results in a stable system, i.e. $\Delta x \rightarrow 0$, the system will be stable about the operating point $(x_0, u_0)$. Since helicopter vertical flight is typically handled by keeping the rotor speed constant using the throttle and adjusting the collective pitch to change the amount of lift developed, we decoupled our controller to reflect this. The throttle was controlled by the error in the rotational speed via $K_{13}$ and the collective pitch was controlled by the error in the height and the vertical velocity via $K_{21}$ and $K_{22}$, respectively. Using the following selection for our control matrix

$$
K = \begin{bmatrix}
0.0 & 0.0 & -20.0 & 0 & 0 \\
1000.00 & 200.00 & 0.0 & 0.0 & 0.0
\end{bmatrix},
$$

the eigenvalues for $A + BK$ become $-2.65 \pm 11.17j$, $-3.74$, $-6.84$, and
As all of the eigenvalues are sufficiently far in the left-half plane, we would expect this control to provide a desirable response to changes in the desired height.

**Flight Tests**

This controller was then tested using the helicopter test bed described in the previous sections. A desired height was selected and the hover control routine was initialized. The plots following in Figures 5a-d are the results of stepping the desired height from 0.75 to 1.20 meters while the linearized hover control is activated the entire time. Note that the desired height was stepped at approximately 0.84 seconds into the data record.

From Figure 5a, we can see that the response time is quick and that the helicopter exhibited only a small amount of overshoot. In figure 5b we see that as expected the collective pitch is increased when the step in the desired height is applied. This increase in collective pitch causes the rotational speed to decrease (see Figure 5c) due to the increased airfoil drag (see the third term of equation (2)) which in turn results in an increase in the throttle servo input to compensate for this drag loss as seen in Figure 5d. The only detraction from these results is the steady-state error present in the final height. This is a result of not having perfect parameter estimations and the fact that the controller is based on a linear approximation of the nonlinear system. As mentioned earlier, the engine dynamical behaviour varies from day to day and sometimes from minute to minute depending on the remaining fuel, fuel/lubricant mixture, the air/fuel ratio, etc. This makes it difficult to model the engine dynamics with precision. As a result, adaptive and various other nonlinear control
techniques that don't require perfect knowledge of the plant parameters should and are being investigated.

An important factor that we will need to take into account before attempting free flight will be the dynamic load of the stand. A gas cylinder/piston was installed to balance out the weight of the stand in an attempt to reduce this loading. However, the gas cylinder does not correct for the additional effective inertia added by the stand and in fact it will add to the damping of the system. As our current parameters include the effective loading of the stand, we will need to make some slight modifications to our dynamic model (in particular Equation (1)) to correct for this before free flight is attempted. We expect our control of hover in free flight to do as well if not better than it did in tethered flight. In free flight however, a major problem which we are currently encountering is finding inexpensive and small real time sensors for height and velocity measurements.

6. Summary and Conclusion

In this paper we have described a real-time control system test bed for helicopter flight control. The organization of the hardware and software system was also described. We then described the dynamic model of the system and how the parameters were measured through experimentation. Problems associated with the control of a gasoline engine was also described. A linearized controller for hovering and the control of the rotor angular velocity was also described. The response of the nonlinear plant to our control design was effective and was also presented in this paper. As a result of this work several nonlinear control designs are currently being investigated and implemented on our test bed and these will be reported in future publications.
References


13. Pallett, T., and Ahmad, S., "Neural Network Control of Miniature Helicopter in Vertical Flight," Working Paper at Purdue University,
Figure 2: Servo Module

Figure 3: Miniature Helicopter and Flight Stand

Figure 4: Linearized State Feedback Control Block Diagram
Figure 5a: Position Step Response During Hover Control

Figure 5b: Collective Pitch During Hover Control
Figure 5c: Rotational Speed During Hover Control

Figure 5d: Throttle Servo Input During Hover Control