Design and Implementation of UAV's Autopilot Controller

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Abstract: Unmanned Aerial Vehicles (UAVs) are remotely piloted or self-piloted aircraft by inputted program in advance or artificial intelligence. In this study Aileron and Elevator are used to control the movement of airplane for horizontal and vertical flights about its longitudinal and lateral axis. In an introduction, the drone was linearly modeled by extracting aerodynamic parameter through flight test and simulation, lift and drag coefficient corresponding to angle of attack, changes of pitching moment coefficient. In the main subject, the flight simulation was performed after constructing hardware using TMS320F2812 from TI company and PID with lateral and longitudinal controller for horizontal and vertical flights. Flying characteristics of two system were estimated and compared through real flight test with hardware equipped algorithm and adaptive algorithm that was applied to consider external factors such as turbulence. In conclusion the control performance of the controller with proposed algorithm was streamlined at lateral and longitudinal controller respectively, we will discuss guidance command to pass way point.

Keywords: Autopilot, Adaptive Control, UAV(Unmanned Aerial Vehicle), MRAC

1. INTRODUCTION

Unmanned Aerial Vehicles(UAVs) are greatly consist of three parts. The first part of UAVs is a navigation system to mainly measure location, attitude, velocity, acceleration and angular velocity of an aircraft. UAVs with low price measure attitude and angle of direction with an AHRS(Attitude Heading Reference System) and get location information using GPS(Global Positioning System). The second part is an autopilot controller to automatically control attitude, altitude and direction change. The third part is guidance control. UAVs complete a mission at target position along with the constant path by command that was transmitted from ground station. It is necessary to calculate the exact location in navigation system and the systematic combination of autopilot controller and guidance algorism are required.

This paper mainly describes an autopilot of UAVs. The law of control based on traditional control principle is variously applied in many fields such as an airplane, a helicopter and a combat plane. The traditional control as a linear controller is required to be linear according to flight condition at the overall flight envelop before design. However linear model based on traditional control theory is only approximate value and is not specialized complicate information included in non linear model. Moreover an acceptable performance with extensive scheduling during whole flight condition is required because the performance substantially decrease when aircraft is out of design trim point and a linear model is only effective very small perturbation under equilibrium condition and performance specialization used in linear controller design is directly not related to nonlinear system. Recently ,the adaptive control technology has been used to overcome these disadvantages as mentioned above. An adaptive control technology improve model uncertainty according to enhancement of mobility and flight envelop.

In this paper, we investigated whether a designed system equipped navigation system performs a given mission or not after applying autopilot and adaptive algorithm in micro-controller . In particular, we estimated the characteristic of a longitudinal and lateral controller designed with adaptive controller to improve model uncertainty and compared each controller with a traditional linear controller.

2. CONTROLLER DESIGN

2.1 Configuration of UAV

The aspect of UAVs used in modeling is shown as below.

Figure.1 Aspect of UAV

The configuration of used UAVs is a 230cm full length, 250cm full width, 7Kg weight itself, 5Kg loading weight and the engine(YW48cc) was remodeled inversely.

2.2 Design of Linear controller

Longitudinal controller and lateral controller assumes a few things as follows.[4]

Assumption:
① No elastic transform due to rigid body
② Aircraft is symmetric on a X-Y plane of a body coordinates system.
③ The earth is flat and is fixed.
④ A flight was carried out in specific envelop.

The state space dynamic equation for a longitudinal model is represented as follows within above assumption.

$$\begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \\ \dot{h} \end{bmatrix} = \begin{bmatrix} X_u & X_w & X_q & -g\cos\Theta_0 & 0 \\ Z_u & Z_w & U_0+Z_q & -g\sin\Theta_0 & 0 \\ M_u & M_w & M_q & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \\ \sin\Theta_0 & -\cos\Theta_0 & 0 & U_0\cos\Theta_0 & 0 \end{bmatrix} \begin{bmatrix} u \\ w \\ q \\ \theta \\ h \end{bmatrix} + \begin{bmatrix} X_{\delta e} & \cos\Theta_0/m \\ Z_{\delta e} & -\sin\Theta_0/m \\ M_{\delta e} & e_T/I_{YY} \\ 0 & 0 \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \delta_e \\ \delta_T \end{bmatrix} \tag{1}$$

Control inputs for a longitudinal control are elevator and throttle. A control law according to each control input is as below

$$\delta_{th} = K_u \cdot u + K_{du} \cdot \frac{du}{dt} \tag{2}$$

$$\delta_e = K_h \cdot (h_{command} - h) + K_{dh} \cdot \frac{dh}{dt} + K_q \cdot q \tag{3}$$

The state space dynamic equation for lateral is represented as equation(4).

$$\begin{bmatrix} \dot{v} \\ \dot{p} \\ \dot{r} \\ \dot{\phi} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} Y_u & Y_p & Y_r - U_0 & g\cos\Theta_0 & 0 \\ L_u & L_p & L_r & 0 & 0 \\ N_u & N_w & N_r & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \end{bmatrix} \begin{bmatrix} v \\ p \\ r \\ \phi \\ \psi \end{bmatrix} + \begin{bmatrix} Y_{\delta a} & Y_{\delta r} \\ L_{\delta a} & L_{\delta r} \\ N_{\delta a} & N_{\delta r} \\ 0 & 0 \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \delta_a \\ \delta_r \end{bmatrix} \tag{4}$$

The rudder and aileron were used as control inputs and control law is as follows.

$$\delta_a = K_r \cdot [K_{heading} \cdot (\psi_{command} - \psi) - r] + K_p \cdot p \tag{5}$$

$$\delta_r = W_{wash}(s) \cdot r(s) \tag{6}$$

Figure.2 and Figure.3 present the block diagram of a longitudinal and lateral controller based on above dynamic equation respectively.

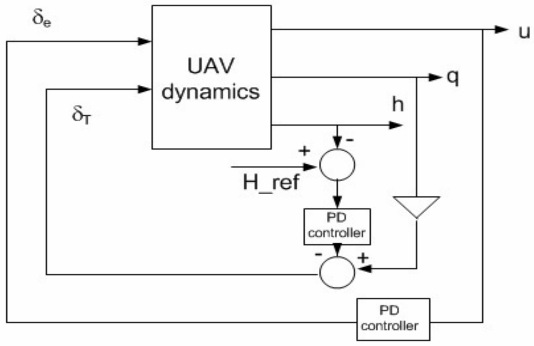

Figure.2 Altitude-hold controller using throttle and elevator

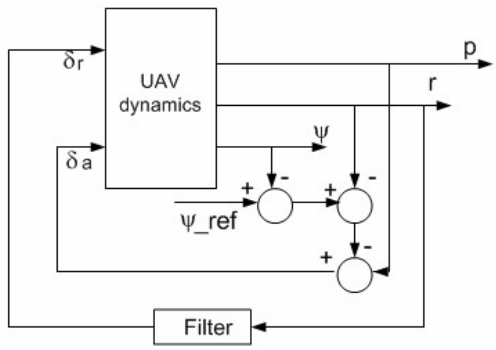

Figure.3 Heading-hold controller using aileron and rudder

2.3 Design of adaptive controller

The system equation of UAVs was defined as follows[1]

$$\dot{x}_p(t) = A_p x_p(t) + B_p u_p + v_d(t) \tag{7}$$

$$y_p = C_p x_p(t) \tag{8}$$

Where $x_p(t)$ is a state vector, $u_p(t)$ is a input vector, $y_p(t)$ is a output vector and $v_d(t)$ is a dither signal vector displaying the effect of wind and cross-coupling.

The state equation of reference model is as below.

$$\dot{x}_m(t) = A_m x_m(t) + B_m u_m(t) \tag{9}$$

$$y_m(t) = C_m x_m(t) \tag{10}$$

The correlation between ideal state model and reference model is obtained as follows.

$$x_p^*(t) = S_{11} x_m(t) + S_{12} u_m(t) \tag{11}$$

$$u_p^*(t) = S_{21} x_m(t) + S_{22} u_m(t) \tag{12}$$

The error $e(t)$ to flight attitude between real plant and reference model is as below.

$$e(t) = x_p^*(t) - x_p(t) \tag{13}$$

The adaptive algorithm is represented as equation(14).

$$u_p(t) = K_x(t)x_m(t) + K_e(t)e_y(t) + K_u(t)u_m(t) \tag{14}$$

Where $e_y(t)$ is a output error of aircraft and reference model defined as follows.

$$e_y(t) = y_m(t) - y_p(t) \tag{15}$$

$$r(t) = [e_y(t), x_m^T(t), u_m^T(t)]^T \tag{16}$$

Adaptive gain matrix $K_e(t), K_x(t), K_y(t)$ is substituted by matrix form as equation (17).

$$K(t) = [K_e(t), K_x(t), K_y(t)] \tag{17}$$

Adaptive gain $K(t)$ is defined as a sum of proportional and integrative gain as follows.[2]

$$K(t) = K_p(t) + K_I(t) \tag{18}$$

$$K_p(t) = e(t)r^T(t)\bar{T} \tag{19}$$

$$\dot{K}_I(t) = e(t)r^T(t)T - \sigma K(t) \tag{20}$$

$$K_I(0) = K_{I0} \tag{21}$$

$$\sigma = \begin{cases} 0 & \|K\| \langle M_0 \\ \sigma_0(\|K\|/M_0 - 1) & M\langle \|K\| \langle 2M_0 \\ \sigma_0 & \|K\| \rangle M_0 \end{cases} \tag{22}$$

Block diagram about adaptive controller is as following Figure.4.

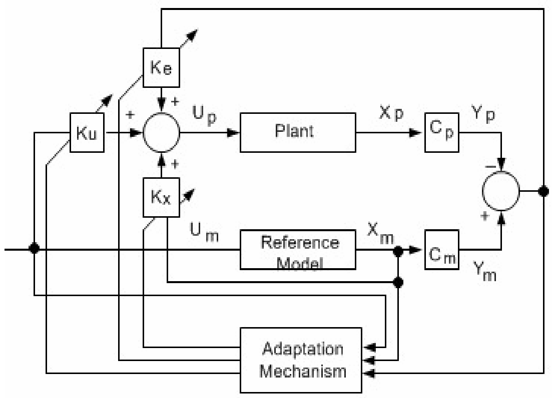

Figure.4 Block diagram of adaptive autopilot controller

3. SIMULATION AND EXPERIMENT

3.1 Estimation of parameters

A linear model is necessary to design the controller. A linear model is confirmed by flight test and empirical equation. The Modified Likelihood Estimation(MMLE) has been used to estimate the parameters from flight motion data.[6]

$$\dot{x}(t) = [A]\{x(t)\} + [B]\{u(t)\} + \{P\} \tag{23}$$

$$\{y(t)\} = [I]\{x(t)\} + \{Q\} \tag{24}$$

{y(t) is a output vector z(t) obtained by the computed output vector and in case a perfect model and parameters are identified z(t) is expressed as follow i

$$\{z(t)\} = \{y(t)\} + \{\eta(t)\} \tag{25}$$

Vector {c} to minimize cost function including all unknown parameter is obtained using MMLE and in case of discontinuous mesurement, the cost function is approximately written as below

$$J = \frac{1}{N-1}\sum_{i=1}^{N}[z_i - y_i]^T[D][z_i - y_i] \tag{26}$$

where i is a time-discontinuous point and N is a number of time point. The weighting matrix [D] is used to focus on measured state in many case. The value of cost function J is minimized using Newton-Raphson method. Newton-Raphson method is an iterative process using the first and second slope for an unknown vector {c} and approximated value of an unknown vector {c}

$$\{C_L\} = \{C_{L-1}\} - \{\nabla_c^2 J\}_L^{-1}\{\nabla_c J\}_L^T \tag{27}$$

L is a iteration number. $\{\nabla_c J\}$ and $\{\nabla_c^2 J\}$ can be calculated by setting J as a first and second slope.

$$[\nabla_c J] = \frac{2}{N-1}\sum_{i=1}^{N}[z_i - y_i]^T[D]\nabla_c[z_i - y_i] \tag{28}$$

$$[\nabla_c^2 J] = \frac{2}{N-1}\sum_{i=1}^{N}\nabla_c[z_i - y_i]^T[D]\nabla_c[z_i - y_i] + \frac{2}{N-1}\sum_{i=1}^{N}[z_i - y_i]^T[D]\nabla_c^2[z_i - y_i] \tag{29}$$

A cost function is as below

$$J = \frac{1}{N-1}\sum_{i=1}^{N}[z_i - y_i]^T[D_1][z_i - y_i] + [c - c_0]^T[D_2][c - c_0] \tag{30}$$

where $[c_0]$ is a deductive estimation, $[D_2]$ is a weighting matrix denoting sufficiency to a deductive value

Figure 5 is shown the procedure to estimate parameter using MMLE

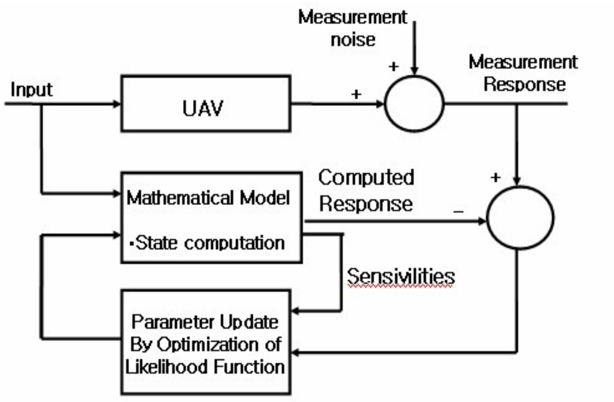

Figure.5 parameter estimation using MMLE

The navigation system equipped with airspeed meter and flow incidence angle sensor was used to obtain flight data. The sensors of the navigation system consist of 3 axis accelerometer, three rate gyro sensor, three magnetic compass and altimeter, the microcontroller was developed with Master using DSP2812 from TI company and slave using Atmega128 from Atmel company. The developed system was consisted of two PCB boards above and below. The real system is presented in Figure 6

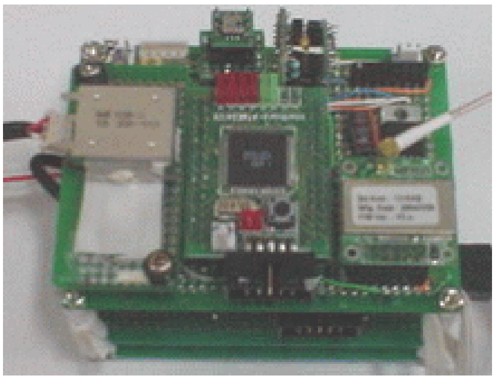

Figure.6 Navigation system

Results from linear modeling obtained by MMLE and empirical equation are as follows.

Longitudinal :

$$A=\begin{bmatrix} -0.056 & 5.802 & -9.804 & 0 & 0 \\ -0.04 & -4.24 & 0 & 0.920 & 0 \\ 0 & 0 & 0 & 1 & 0 \\ 0.029 & -57.301 & 0 & -5.433 & 0 \\ 0 & -19.044 & 19.044 & 0 & 0 \end{bmatrix}$$

$$B=\begin{bmatrix} 0 & -0.188 \\ 0 & -0.208 \\ 0 & 0 \\ 0 & -48.232 \\ 0 & 0 \end{bmatrix}, \quad x=[u,\alpha,\theta,q,h]$$

Lateral :

$$A=\begin{bmatrix} -4.887 & 0.481 & -0.002 & -0.980 & 0 \\ 0 & 0 & 1 & 0 & 0 \\ -50.16 & 0 & -17.72 & 5.8 & 0 \\ 32.17 & 0 & -1.501 & -2.023 & 0 \\ 0 & 0 & 0 & 1 & 0 \end{bmatrix}$$

$$B=\begin{bmatrix} 0 & 0.211 \\ 0 & 0 \\ 92.405 & 17.512 \\ 0 & -38.232 \\ 0 & 0 \end{bmatrix}, \quad x=[\beta,\phi,p,r,\psi]$$

Above parameters are used in linear controller and also in adaptive controller

3.2 Results of simulation

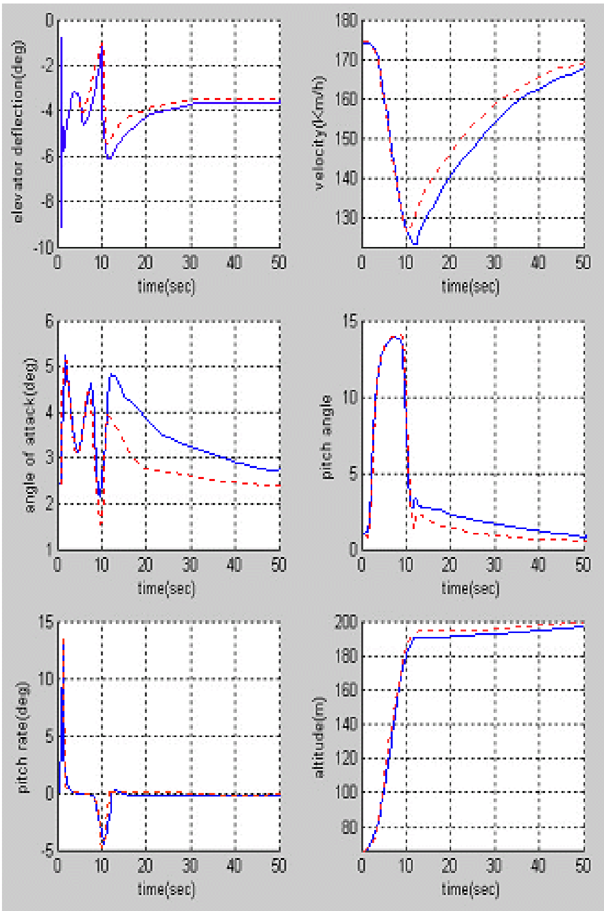

Figure 7 Performance analysis using longitudinal controller

A dotted line in Figure 7 is a result from simulation using classic control theory. The elevator deflection is relatively small with -6 degree of maximum. The aircraft is maintained about 13 degree of pitch angle at the beginning of flight test and the velocity of aircraft is decreased by increase of lift and drag which caused by increased the angle of attack according to altitude rising command. The altitude is close to required altitude. The solid line is denoted that an adaptive controller follows the reference model. An previously designed linear model is used as a reference model and the altitude control command according to altitude rising command rapidly follows the model.

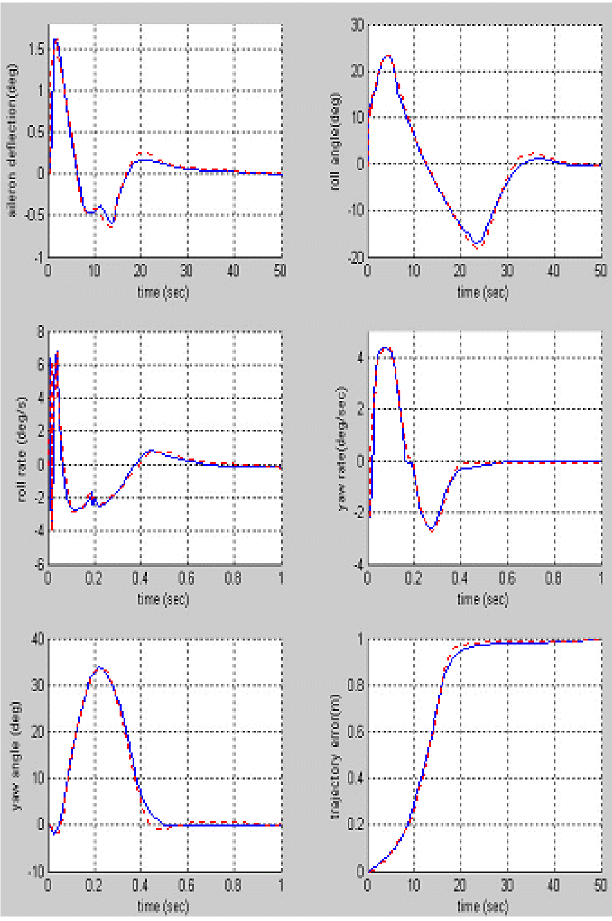

Figure 8 Performance analysis from simulation

The aileron deflection has to be below maximum 1.7 degree for changing the paths as presented in Figure 8. Also, this shows that the trajectory error is reduced due to changed yaw angle caused by sideslip and roll angle change with changing early aileron deflection.

4. CONCLUSION AND FUTURE WORK

In conclusion, both direct MRAC and classical controller designed through linear modeling of an aircraft show a good performance after simulation of each controller. However, when applied to real aircraft the gain obtained by simulation of each controller can not be directly used due to disturbance and noise and needs to be seriously tuned. In this regard, the designed adaptive controller is more effective than others. Our future work is to establish an algorithm using navigation system developed by us and TMS320F2812 Digital Signal Processor and optimize an algorithm by tuning the gain obtained the simulation.

5. UNITS AND SYMBOLS

5.1 Symbols

u : airspeed
w :acceleration of Z axis
q : pitch rate
θ : pitch angle
h : altitude
Θ_0 : initial pitch angle
m :weight
r : yaw rate
ϕ : roll angel
ψ : yaw angle
g : gravity
x_p^* : ideal state vector
u_p^* : ideal input vector
S : arbitrary matrix
α : angle of attack
β : sideslip angle

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