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Development of Altitude Control System for Autonomous Flight Vehicle

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***Abstract*— The purpose of doing this research is to develop and implement altitude control system for holding of aerial vehicle. This paper emphasizes on the longitudinal stability control system. Experimental autonomous flight vehicle developed by Tamkang University is used as the application airframe as it gives coefficients to get the very accurate model of aircraft flight behaviour. An altitude hold control structure is constructed classical designs. The structure contains two control loops. The inner loop is for altitude rate regulation. In this control loop, the combination of altitude rate and pitch angle are used as the feedback signals. The simulation results of the inner loop are used to design the outer altitude control loop. The inner control loop for this research work is considered in this paper.**

***Keywords*—autonomous flight vehicle system, altitude holding system, elevator control, longitudinal stability, bode plot.**

I. INTRODUCTION

Autopilot is one of the essential functions for aircraft controls, especially for an unmanned aerial vehicle. The performance of the autopilot system directly affects the performance and mission success of the aircraft. The motion of an airplane in free flight is extremely complicated [1]. It contains three translation motions (vertical, horizontal, and transverse), three rotational motions (pitch, yaw, and roll) and numerous structural coupled elastic motions. To reduce the complexity of analysis, it is assumed that the aircraft is rigid-body and the aircraft's motion consists of small deviation from its equilibrium (trimmed) flight condition. In addition, the aircraft equations of motion can be separated into two groups, namely, longitudinal equations and lateral equations.

The longitudinal dynamics is characterized by two oscillatory modes of motion. One mode is lightly damped and has a long period, called long period or phugoid mode. Another mode is heavily damped and has a very short period, called short-period mode. The low phugoid damping is very objectionable under instrument flights for lightly damped low frequency oscillation.

A new control structure for altitude hold autopilot design is proposed in this paper. The slow and fast models are used to synthesize an autopilot control law based on the proposed control structure. The control law is designed sequentially to satisfy the design requirements for low and high frequency regions. In the proposed control structure, the fast dynamics of the angle of attack signal is treated first then using the slow model to proceed to the altitude hold design utilizing the combination of altitude rate and pitch angle as feedback to meet the design requirements.

II. AERONAUTICAL CONCEPTS FOR LONGITUDINAL CONTROL SYSTEM

For controlling the movement of the aerial vehicle, the coordinate systems and the equation of motions needed to study first.

A. Coordinate Systems

The movement of an airplane is described by the equations of motion with respect to the body frame coordinate system. In general there are three coordinate systems that are important when describing the motion of an airplane.

1. Body-fixed coordinate system
2. Earth-fixed coordinate system
3. Atmosphere-fixed coordinate system

The body-fixed frame can be expressed in terms of the Earth-fixed frame by the use of Euler angles (Φ, θ and ψ).

1. Body-fixed coordinate system

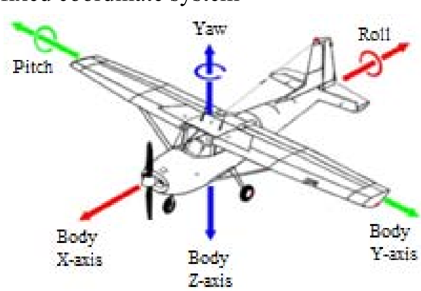

Fig. 1 Body coordinates system and rotations [2]

This coordinate system's origin is located at the aircraft center of gravity. The x-axis points forward along some axis of the fuselage in the aircraft's plane of symmetry. The y-axis is normal to the plane of symmetry pointing in the direction of the right wing. The z-axis points downward, completing the right-handed Cartesian system. See also Fig. 1. Rotations about the x, y and z-axis are called roll (Φ), pitch (θ) and yaw (ψ) respectively [2].

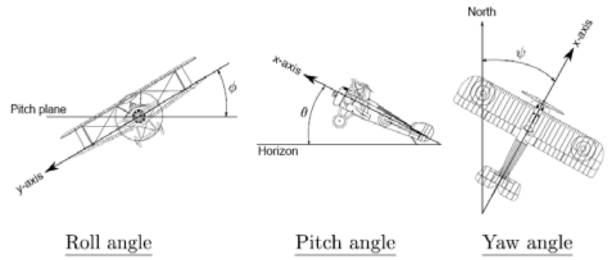

Fig. 2 Euler attitude description angle [3]

2. Earth-fixed coordinate system

The Earth-fixed coordinate system points in the directions North, East and Down, the NED frame. The x; y-plane is normal to the local gravitational vector with the x-axis pointing north and the y-axis pointing east. The z-axis points down. For navigation on a small piece of the Earth, the curvature of the Earth can be neglected for simplification. The NED frame is then fixed and does not rotate.

3. Atmosphere-fixed coordinate system

The atmosphere-fixed coordinate system is defined such that all three axes are always parallel to those of the Earth-fixed coordinate system. However, the atmospheric coordinate system moves at a constant velocity relative to the Earth-fixed coordinate system. The atmosphere coordinate system is only used when deriving the equations of motion. It is also used in wind tunnel testing of aircraft models.

B. Equations of Motions

When deriving the equations of motion, the aircraft is considered as a rigid body. To switch between the body-fixed coordinate frame and Earth-fixed coordinate frame, a matrix multiplication is used, consisting of three rotations:

a) Rotation about the z-axis, positive nose right (yaw ψ)
b) Rotation about the new y-axis, positive nose up (pitch θ)
c) Rotation about the new x-axis, positive right wing down (bank Φ)

The yaw, pitch and bank are known as Euler’s angles. The transformation matrix B is given as:

$$\begin{bmatrix} x \\ y \\ z \end{bmatrix}_{\text{body-fixed}} = B \begin{bmatrix} x \\ y \\ z \end{bmatrix}_{\text{earth-fixed}} \quad (1)$$

with

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

The inertia properties of the rigid airplane body can be defined by:

$$c_1 = \frac{(J_y - J_z)J_z - J^2_{zz}}{\tau} \quad c_2 = \frac{(J_x - J_y + J_z)J_{xz}}{\tau} \quad c_3 = \frac{J_z}{\tau} \quad c_4 = \frac{J_{zz}}{\tau}$$

$$c_5 = \frac{J_z - J_x}{J_y} \quad c_6 = \frac{J_{xz}}{J_y} \quad c_7 = \frac{1}{J_y} \quad c_8 = \frac{J_z(J_x - J_y) + J^2_{xz}}{\tau}$$

$$c_9 = \frac{J_x}{\tau} \quad \tau = J_x J_z - J^2_{xz} \quad (2)$$

And the aerodynamic and propulsive forces and torques will be split into the body axes:

$$F_B = \begin{bmatrix} F_x \\ F_y \\ F_z \end{bmatrix} \qquad T_B = \begin{bmatrix} L \\ M \\ N \end{bmatrix} \quad (3)$$

With these definitions, the equations of motion of the aircraft can be described by [2]:

$$\dot{U} = RV - QW - g\sin\theta + F_x / m$$

$$\dot{V} = -RU + PW + g\sin\Phi\cos\theta + F_y / m$$

$$\dot{W} = QU - PV + g\cos\Phi\cos\theta + F_z / m$$

$$\dot{\phi} = P + \tan\theta\,(Q\sin\theta - R\cos\theta)$$

$$\dot{\theta} = Q\cos\theta - R\sin\Phi$$

$$\dot{\psi} = (Q\sin\Phi + R\cos\Phi) / \cos\theta$$

$$\dot{P} = (c_1 R + c_2 P)\,Q + c_3 L + c_4 N$$

$$\dot{Q} = c_5 P R - c_6 (P^2 - R^2) + c_7 M$$

$$\dot{R} = (c_8 P - c_2 R)\,Q + c_4 L + c_9 N$$

$$\dot{p}_N = U\cos\theta\cos\psi + V(\sin\Phi\sin\theta\cos\psi - \cos\Phi\sin\psi) + W(\sin\Phi\sin\theta\cos\psi + \sin\Phi\sin\psi)$$

$$\dot{p}_E = U\cos\theta\sin\psi + V(\sin\Phi\sin\theta\sin\psi + \cos\Phi\cos\psi) + W(\cos\Phi\sin\theta\sin\psi - \sin\Phi\cos\psi)$$

$$\dot{p}_D = -U\sin\theta + V\sin\Phi\cos\theta + W\cos\Phi\cos\theta$$

With the definition of the state vector as:

$$X = [\; U \;\; V \;\; W \;\; \Phi \;\; \theta \;\; \psi \;\; P \;\; Q \;\; R \;\; p_N \;\; p_E \;\; p_D \;]^T$$

with

U, V and W - the velocity with respect to the body axes x, y and z respectively

Φ, θ and ψ - the three Euler angles

P, Q and R - the angular rates

p_N, p_E and p_D - the positions with respect to the NED frame.

The model is linearized using the so called 'flat Earth' approximation. In this approximation it is assumed that the Earth is flat, and this can be used when navigation is only required over a small area of the Earth [1].

III. Longitudinal Dynamics of an Autonomous Flight Vehicle

Before getting into the control system design, longitudinal flight characteristics should be analyzed. For this purpose, Equations of Motion (EoMs) governing the longitudinal flight, taken have been used for analysis where the first two are force equations in *x* and *z* directions, respectively, while M is the moment equation in *y* direction.

$$x:\ \left(\frac{mu}{Sq}s - C_{Xu}\right)'\ u(s) - C_{X\alpha}{}'\alpha(s) - C_w(\cos\Theta)\theta(s) = 0$$

$$z:\ -C_{Zu}{}'\,u(s) + \left[\left(\frac{mu}{Sq} - \frac{c.C_{Z_{\dot\alpha}}}{Sq}\right)s - C_{Z_\alpha}\right]'\alpha(s)$$

$$+\left[\left(-\frac{mu}{Sq} - \frac{c.C_{Z_q}}{2u}\right)s - C_w\sin\Theta\right]\theta(s) = 0$$

$$M:\ \left(-\frac{c.C_{m_{\dot\alpha}}}{2u}s - C_{M\alpha}\right)'\alpha(s) + \left(\frac{I_y}{Sqc}s^2 - \frac{c.C_{M_q}}{2u}s\right)\theta(s) = 0$$

where

u = change of velocity in longitudinal flight,
α = change of angle of attack in longitudinal flight,
Θ = pitch angle,
θ = change of pitch angle from equilibrium point, so that
$u = u / U_0$ and $\alpha = w / U_0$,

where

u is perturbation velocity in X direction,
w is perturbation velocity in Z direction and
U_0 is the steady state velocity in longitudinal flight.

IV. System Implementation

A. Airframe

Since this research is aimed at fixed wing autonomous flight vehicle research, a suitable airframe is required. Experimental autonomous flight vehicle developed by Tamkang University is used as the application airframe as it gives coefficients to get the very accurate model of aircraft flight behaviour. The principal specifications of this airframe are as follows:

TABLE I [4]
Principal Specification of Autonomous Flight Vehicle

Weight	7 kg
Cruise speed	15.264 m/s
Wing span	2.1 m
Wing area	0.802 m^2
Horizontal tail area	0.1332 m^2

Fig. 3 Photograph of airframe used in this research [4]

Using the aircraft data and coefficients, the calculated transfer function of change in pitch angle to elevator deflection in frequency domain is

$$\frac{\Delta\theta}{\Delta\delta_e} = \frac{-2.804(s+0.1217)}{s^2+0.1496s+0.2134}$$

Transfer function of change in angle of attack to elevator deflection in frequency domain is

$$\frac{\Delta\alpha}{\Delta\delta_e} = \frac{-0.5045(s^2-0.0234s+0.0.8401)}{s^2+0.1496s+0.2134}$$

The purpose of an altitude hold control structure is to automatically control the aircraft to flight at a selected altitude. Assume the aircraft's velocity vector is V_T, then the vertical component of V_T is $V_T \sin\theta_F = \dot{h}$, where θ_F is the flight path angle (the inclination of the velocity vector V_T to the horizontal), and h is the altitude of aircraft.

The flight path angle θ_F is the difference between the aircraft pitch angle θ and the angle of attack α, that is $\theta_F = \theta - \alpha$. In level flight, the flight path angle θ_F is small and $V_T \approx u$, the forward velocity. Hence

$$\Delta\dot{h} \approx u(\Delta\theta - \Delta\alpha)$$

Thus

$$\Delta h = \int u(\Delta\theta - \Delta\alpha)dt$$

At the considered trimmed flight condition, u_0= 15.264 m/s, the transfer function of altitude rate to elevator deflection is

$$\frac{\Delta\dot{h}}{\Delta\delta_e} = \frac{4.016s^3-14.78s^2-1544s-33.49}{s^4+13.62s^3+37.96s^2+8.227s+7.666}$$

The complete diagram of altitude and altitude hold control loop for experimental UAV is shown in Fig. 4.

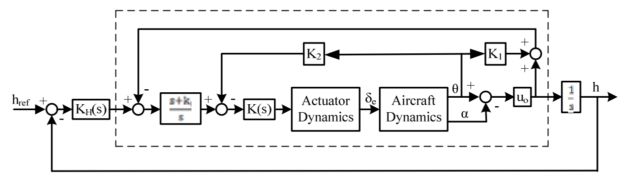

Fig. 4 Complete diagram of altitude and altitude hold control loop

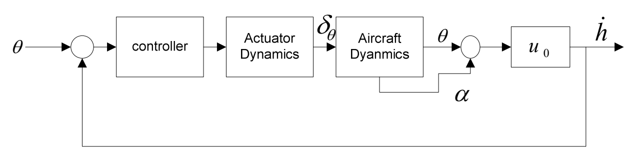

Fig. 5 Altitude rate regulation control loop (inner loop)

Fig. 5 is the diagram of the altitude rate regulation control loop. The actuator dynamics, A(s), is represented as a simple first order model with D.C. gain A (0) equals to 1.

$$A(s) = \frac{10}{s+10}$$

V. Simulation Results

The root locus design tool is an interactive design toolbox in MATLAB software. Using this design tool (''rltool'' command in MATLAB environment), we can add pole(s), zero(s), and adjust gain for the controller, and evaluate the Bode plot and performance of the closed-loop system immediately. A controller to satisfy certain design requirement can be determined easily by using this design tool.

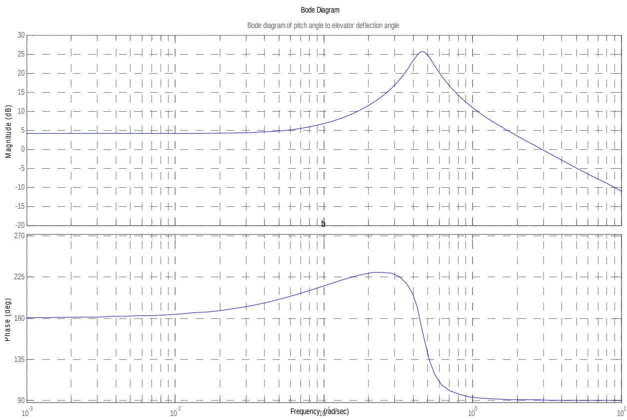

Fig. 6 Frequency response of the transfer function; $\Delta\theta/\Delta\delta_e$

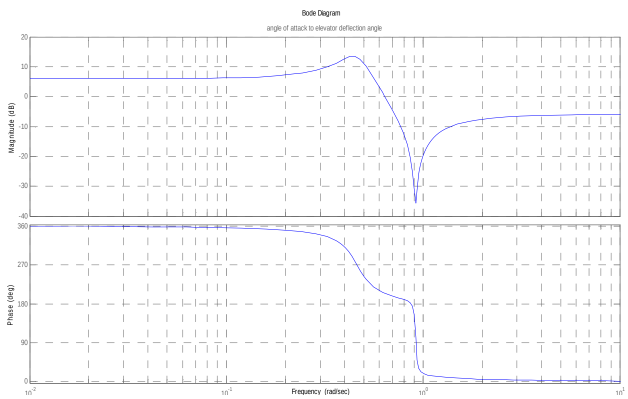

Fig. 7 Frequency response of the transfer function; $\Delta\alpha/\Delta\delta_e$

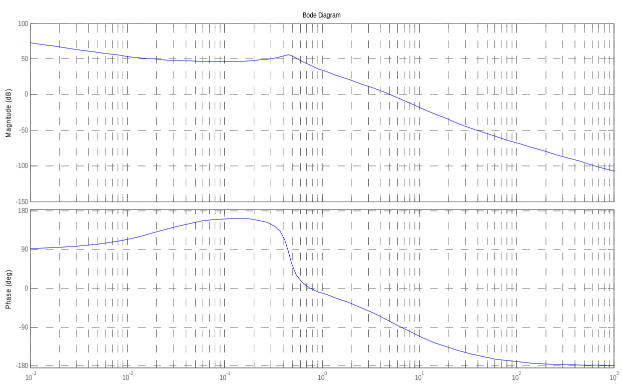

Fig. 8 Frequency response of the transfer function; $\Delta\dot{h}/\Delta\delta_e$

From this design, the gain margin is 17.44dB and the phase margin is 61.22 degrees. The loop gain is -18.7dB at 10 rad/sec. Thus, the design requirements are satisfied.

VI. CONCLUSION

An altitude hold control structure presented in the paper basically contains two control loops. The inner loop is for altitude rate regulation. The outer loop is for altitude control. In the inner loop design, a fast controller is designed to satisfy the high frequency requirements. It is needed to design, test, and optimize a control system that can be able to control all actuators simultaneously. Similarly, the controllers for other control loops are also needed to design and choose suitable one from P, PI and PID controllers using MATLAB.

The performance of the full closed loop autopilot system is considered for further research.

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