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Design and Simulation of Flight Path Control Systems for CHARLIE Aircraft

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Abstract: *Insight into the knowledge of Automatic Flight Control Systems gives an understanding of the basic problem of controlling the aircraft's flight, and enhances its ability to assess the solutions to the problems which are generally proposed. There are a number of flight missions which require that an aircraft be made to follow, with great precision, some specially defined path. Whenever a conventional aircraft is to be controlled, pilot can command rates of rotation in any or all of three axes: pitch, roll and yaw. Path variables, such as heading and pressure altitude, need to be measured in the aircraft; in considering their control in a treatment of flight control. These approximations are linear as well as sufficient; such systems can be regarded as members of the class of flight path control systems, and has been treated intensively in this paper. This paper being firmly based upon time-domain methods, presents modern methods of control theory, particularly the use of state equations which is a natural and effective technique and harmonizes with the mathematical description of the aircraft dynamics that are most completely and conveniently expressed in terms of a state and an output equation. Also this paper relates to particular modes of an AFCS, being concerned with Flight path control systems which has been implemented for the reference aircraft CHARLIE (a very large, four-engine passenger jet aircraft) at different flight conditions. SIMULINK is used to implement Flight path control systems as they are important to form the outermost loop of an integrated AFCS.*
Keywords: *Aircraft dynamics, Automatic Flight Control System (AFCS), Flight Path Control Systems (FPCS)*

I. Introduction

A particular AFCS may have some, or all, of these modes involved in its operation. Some modes being active at all times in the flight, and others being switched on by the pilot only when required for a particular phase of flight. The Flight path of any aircraft is the actual or planned course and there are a number of flight missions which require that an aircraft has to follow, with great precision, some specially defined path. In AFCS studies, the primary concern is to enhance the flyingqualities of the aircraft by the control action of the feedback control system whereas the command inputs are usually considered only secondary. Upon the control action by negative feedback control the disturbance upon the aircraft motion are suppressed. In spite of physical sizing of aerodynamic surfaces certain desirable values of the aircrafts non-dimensional stabilityderivatives, such as$C_{m_\alpha}, C_{m_q}, C_{n_\beta}, C_{n_r}, C_{n_p}, C_{l_\beta}$ can be achieved more effectively using Automatic Control.

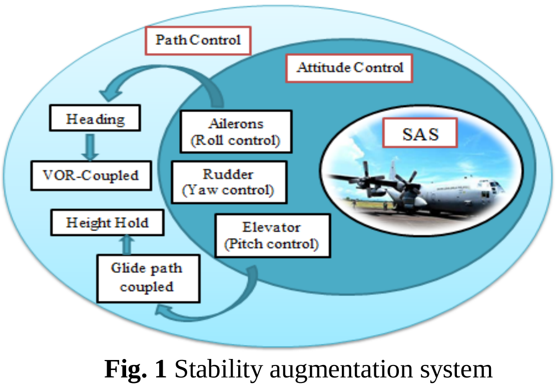

Fig. 1 Stability augmentation system

A. Actuator Dynamics

Generally electrohydraulic actuators are used in the combat and the transport aircraft. Sometimes electric actuators are used in general aviation aircraft. Such actuating systems have their own dynamic characteristics that can affect the performance of the closed loop SAS. The actuator dynamics can be considered to be a very simple transfer function given by

$$\frac{\delta(s)}{\delta_c(s)} = K \tag{1}$$

Where, 'K' is taken as the gain of actuator which is usually in deg/volt. For a large aircraft the actuator must provide large hinge moments for control, consequently the transfer function usually assumed in this situation is

$$\frac{\delta(s)}{\delta_c(s)} = \frac{K\lambda}{(s+\lambda)} \tag{2}$$

B. Sensor Dynamics

Every sensor used in an AFCS is a transducer. The main purpose is to measure motion variables and to produce output voltage or currents which correspond to these motion variables. For SAS's, the most commonly employed sensors are gyroscopes and accelerometers. A sensor is frequently represented by its sensitivity, i.e. K_s.

II. Background Work

[1] The problem considered in this article is the design and evaluation of the robust control law for a small helicopter, which allows for vertical, pitch, and travel rate dynamics tracking reference trajectories. The article is mainly dedicated to robust control aspects of helicopters, where the evaluation of the H∞ control is addressed to demonstrate the effectiveness of the performance and the robustness of the proposed control law.

[2] In this article, decentralized sliding mode controllers that enable a connected and leaderless swarm of unmanned aerial vehicles (UAVs) to reach a consensus in altitude and heading angle are presented. In addition, sliding mode control-based autopilot designs to control those states for which consensus is not required are also presented. By equipping each UAV with this combination of controllers, it can autonomously decide on being a member of the swarm or fly independently. The controllers are designed using a coupled nonlinear dynamic model, derived for the YF-22 aircraft, where the aerodynamic forces and moments are linear functions of the states and inputs.

[3] The work presented here is concerned with the robust flight control problem for the longitudinal dynamics of generic air breathing hypersonic vehicles (AHVs) under mismatched disturbances via a nonlinear-disturbance-observer-based control (NDOBC) method. Compared with other robust flight control method for AHV, the proposed method obtains not only promising robustness and disturbance rejection performance but also the property of nominal performance recovery.

[4] This article considers the practical problems of the control law implementation and system integration using existing control technology. It discusses the altitude control of a flapping-wing micro aerial vehicle (MAV) that describes a control law for stabilizing the vertical motion of a flapping-wing MAV and developed a system architecture that is potentially beneficial inrealizing the autonomous flight of flapping-wing MAVs fewer than 10 g.

[5] This paper is on a vision-based flight control system that uses a skyline-detection algorithm is developed for application to small unmanned aerial vehicles. The skyline-detection algorithm can detect straight or uneven skylines. The system integrates a remote controller, a remotely controlled airplane, a camera, a wireless transmitter/receiver, a ground controlcomputer, and the proposed skyline-detection algorithm to achieve automatic control of flight stability. In the dynamic tests, straight and circular flights are used to verify lateral and longitudinal stability for the proposed flight control system. The experimental results demonstrate the performance and robustness of the algorithm and the feasibility and potential of a low-cost vision-only flight control system.

[6] A case study of an open-source low-cost reconfigurable autopilot design for small unmanned aerial vehicles (Remotely operated Aerial Model Autopilot (RAMA) control system) is presented in this paper. A novel distributed hierarchical architecture, implementing graceful degradation and run-time system reconfiguration techniques, is introduced. RAMA is capable of reconfiguration in case of emergency, meaning that the most critical functions, needed for vehicle controllability, can be taken over by another node of the system if the primary node fails, sacrificing some noncritical functionality. RAMA also utilizes a novel control scheme and controller implementation.

[7] Unmanned air vehicles (UAVs) are useful for observing and planning rescue activities in dangerous areas such as those affected by volcanoes, earthquakes and fires. One such an airplane is a Kite plane that has a large delta shaped main wing that is easily disturbed by the wind, which was minimized by utilizing trim flight with drift. The proposed AFCS for autonomous trajectory following with a wind disturbance include fuzzy logic controllers, speed controllers, a wind disturbance attenuation block and low level feedback controller. And this proposed AFCS succeeded in following the desired trajectory, under the wind disturbances.

III. Disturbances Affecting Aircraft Motion

There are several factors which affect the motion of an automatically controlled aircraft. They are Maneuver commands, Atmospheric effects, Noise from the system and noise from the sensors. Maneuver commands are applied by human pilot or a guidance commands, navigation or a weapon systems which are deliberate inputs to the AFCS and intended to change the aircraft path. Also the motion of an aircraft is erratic when the air through which an aircraft flies is never still. Microburst, a severe downburst of air is another violent atmospheric phenomenon which can be encountered in flight. Atmospheric turbulence is a stationary random process and its statistical properties are independent of time. Wind shear is another rapid change of airflow which could be hazardous particularly to aircraft lying at low altitudes and at low speeds. In the midst of all these disturbances the primary concern of an AFCS is to suppress as much as possible the unwanted effects of such disturbances.

IV. State And Output Equations

A state equation is a first order vector differential equation and it is a natural form to represent aircraft equations of motion. Its most general expression is

$$\dot{x} = Ax + Bu \tag{3}$$

Where $x \in R^n$ is the state vector, $u \in R^m$ is the control vector. If the concern is with the motion variables other than those chosen as state variables, then an output equation is wanted. The output equation is merely an algebraic equation which depends solely upon state vector, and, occasionally, upon the control vector also. Its customary form of expression is:

$$y = Cx + Du \tag{4}$$

C. Aircraft Equations of Longitudinal Motion

If the state vector is defined as, say:

$$x = \begin{bmatrix} u \\ w \\ q \\ \theta \end{bmatrix} \tag{5}$$

And if the aircraft is being controlled only by means of elevator deflection, δ_E such that its control vector is defined as

$$u \triangleq \delta_E \tag{6}$$

Then from the Eq. (1) it can be written as

$$A \triangleq \begin{bmatrix} X_u & X_w & 0 & -gcos\gamma_0 \\ Z_u & Z_w & U_0 & -gsin\gamma_0 \\ \widetilde{M}_u & \widetilde{M}_w & \widetilde{M}_q & \widetilde{M_\theta} \\ 0 & 0 & 1 & 0 \end{bmatrix} \tag{7}$$

$$B \triangleq \begin{bmatrix} X_{\delta_E} \\ Z_{\delta_E} \\ \widetilde{M}_{\delta_E} \\ 0 \end{bmatrix} \tag{8}$$

D. Aircraft Equations of Lateral Motion

For lateral motion the control motion may be defined as

$$u \triangleq \begin{bmatrix} \delta_A \\ \delta_R \end{bmatrix} \tag{9}$$

If the state vector, x is defined as

$$x = \begin{bmatrix} v \\ p \\ r \\ \phi \\ \varphi \end{bmatrix} \tag{10}$$

Then the state equation is given by Eq. 1 where

$$A \triangleq \begin{bmatrix} Y_v & 0 & U_0 & -gcos\gamma_0 & 0 \\ L_v{}' & L_p{}' & L_R{}' & 0 & 0 \\ N_v{}' & N_p{}' & N_r{}' & 0 & 0 \\ 0 & 1 & tan\gamma_0 & 0 & 0 \\ 0 & 0 & sec\gamma_0 & 1 & 0 \end{bmatrix} \tag{11}$$

$$B \triangleq \begin{bmatrix} 0 & Y_{\delta_R} \\ L_{\delta_A}{}' & L_{\delta_R}{}' \\ N_{\delta_A}{}' & N_{\delta_R}{}' \\ 0 & 0 \\ 0 & 0 \end{bmatrix} \tag{12}$$

The reference aircraft is a CHARLIE aircraft which is a very large four engine passenger aircraft. And the stability derivatives for CHARLIE aircraft have been taken from [8] for all flight conditions to implement all the Stability Augmentation System in both Longitudinal and Lateral motion.

E. Height Control Systems:

When a system is used to control the height at which an aircraft is flying, it acts as a feedback regulator to maintain the aircraft's height at a reference value, even in the presence of disturbances. The pilot can either fly the aircraft by manual control or use the pitch attitude control system to control the climb of the aircraft until it has reached the required height. When that height has been reached, the height control system is selected to maintain that height thereafter. The height control system is required to control the aircraft in a manner which will cause the aircraft's path to follow closely, and with good dynamic response, a particular height profile. In general a height control system is often referred to as a 'height hold' system.

Supersonic Transport (SST) aircraft are known to have phugoid modes of very long periodto recover the aircraft's attitude and height after an upset. For such SST aircraft, a height hold system is a necessity.

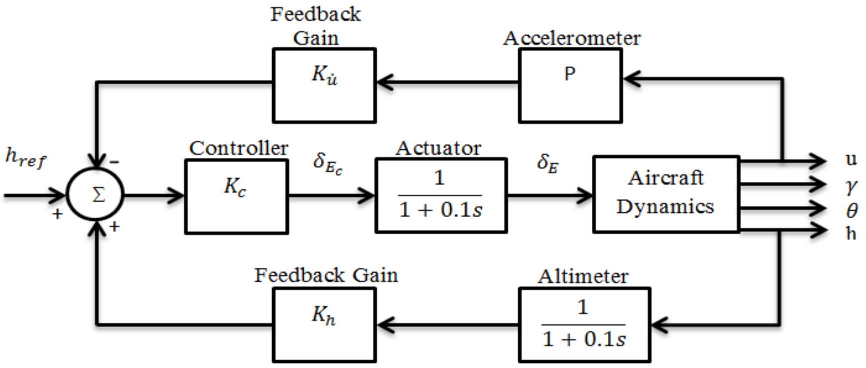

Fig. 2 Height Hold Control Systems 1

A block diagram representing a typical height hold system is shown in Fig. 2. The height of the aircraft can be seen to be controlled by means of elevator deflection; that deflection is produced by an actuator, the dynamics of which have been represented as a first approximation by a first order transfer function, with a value of time constant of 0.1s. The dynamics of the altimeter have also been assumed to be linear and first order, with the same value of time constant. For its successful operation, the system requires a longitudinal accelerometer to provide a feedback signal proportional to u.

The sensitivity of the altimeter, denoted byk_h , can be taken to be unity without loss of generality. Being a closed loop feedback control system, the height hold system may be stable, unstable, oscillatory or over-damped depending upon the aircraft dynamics and the values of the controller gains, $k_{\dot{u}}$ and k_c . With $\dot{K}_u$ being selected at -200.0, it is found that a somewhat oscillatory, response results with an evident error in the steady state value of the height when the controller gain is chosen to be 0.08 mVm^{-1}. It is obvious from an inspection of the response shown in Fig. 3, how large are the variations in flight path angle and for how long they persist.

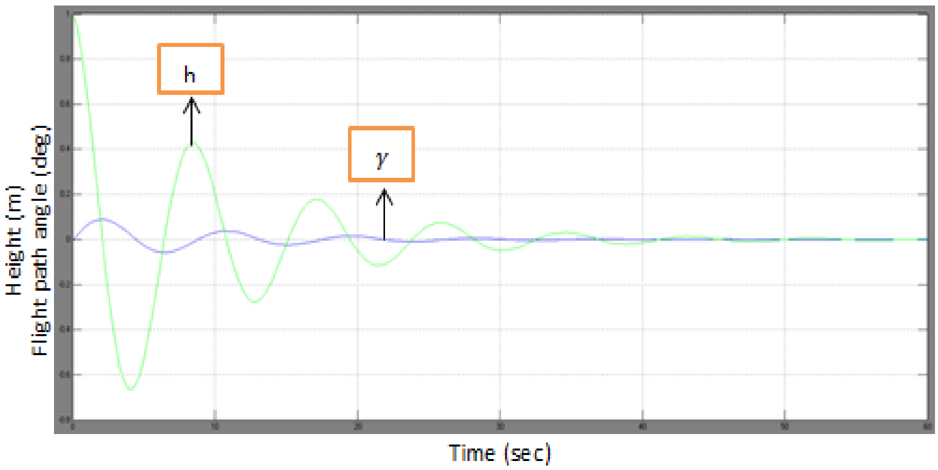

Fig. 3 Response of Height Hold Control System-1

The block diagram of an alternative height hold system is shown in Fig 4. Notice that it represents a pitch attitude control system; with a pitch rate SAS as its inner loop. An outer loop, involving the use of an altimeter to provide a feedback signal proportional to height, is used to achieve the height hold function.

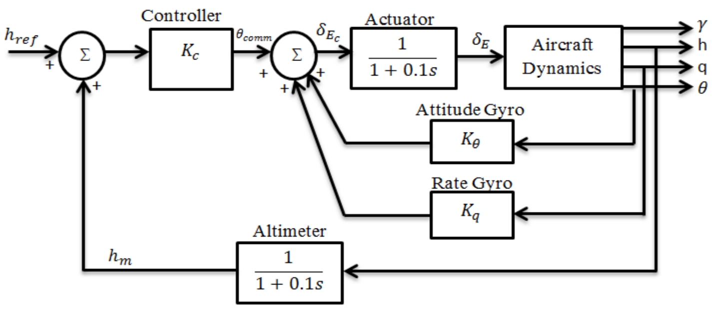

Fig.4 Height Hold Control Systems 2

An outer loop, involving the use of an altimeter to provide a feedback signal proportional to height, is used to achieve the height hold function. It can be seen from Fig. 5, how much improved are the dynamic response and how the error has been very nearly eradicated.

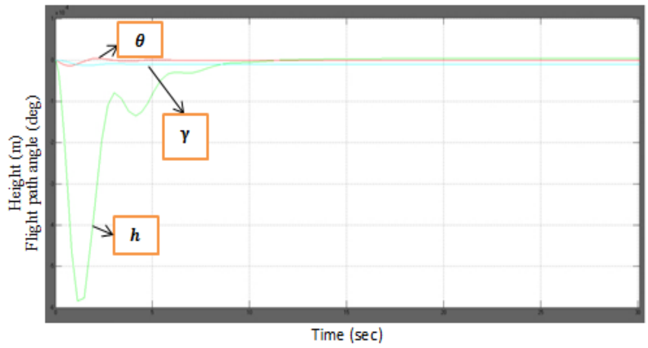

Fig. 5 Response of Height Hold Control System-2

F. Speed Control System:

Although speed is not truly a path variable, its exact control is essential for many tasks related to the control of an aircraft's flight path. If speed can be controlled, the position of an aircraft, in relation to some reference point, can also be controlled. A block diagram representing a typical airspeed control system is shown in Fig. 6.

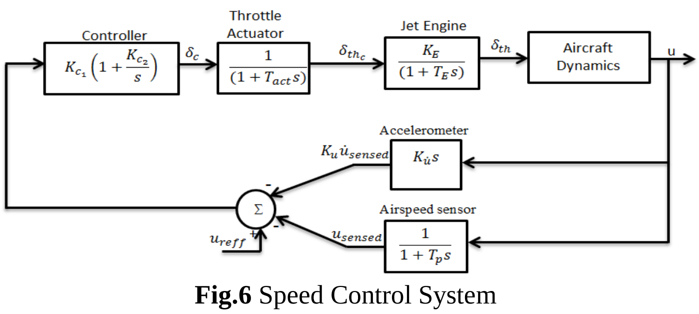

Fig.6 Speed Control System

Speed is controlled by changing the thrust δ_{th} ,of the engines; such a change in thrust is obtained by altering the quantity of the fuel flowing to the engines by means of the throttle actuator. Typical values for the time constant,T_E of a jet engine lie in the range 0.3-1.5 s depending on the thrust setting and the flight condition. Since the airspeed sensor is usually a barometric device, it has been represented by a first order transfer function, with a time constant of T_p. The controller is a proportional plus integral type; the integral term has been added to remove, if required, any steady state error in the response of the airspeed system to constant airspeed command. If it is assumed, in the first place, that the aircraft is to be maintained at its equilibrium airspeed,U_0then no significant changes in airspeed, u, should persist, hence u_{reff} if taken to be zero.

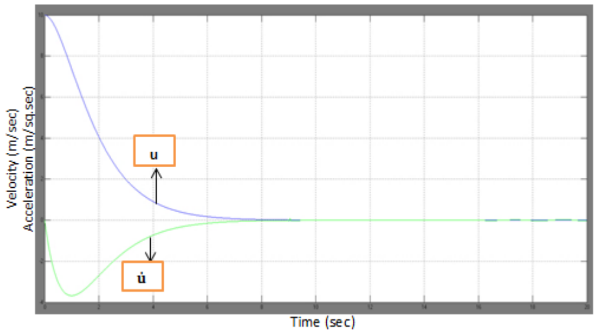

Fig. 7 Response of Speed Control System

The dynamic response of the system of Airspeed control system to an initial airspeed error of 10 ms^{-1} in the equilibrium airspeed of 75ms^{-1} , for CHARLIE-1, is shown in Fig. 7. Note the small error at values of time greater than 12s. In the response below the longitudinal acceleration, u is also shown. The key factor in the response of this speed control system is the authority allowed over the engines thrust.

G. Mach Hold System

In the Mach hold mode the aircraft is made to fly at a constant Mach number by automatically controlling the flight path angle through the elevators. For this mode of operation the aircraft is first trimmed to fly straight and level and the power adjusted to yield the desired Mach number. The Mach hold mode of the flight control system is then engaged. As the aircraft cruises, fuel is used, the weight of the aircraft decreases, and the speed tends to increase. The increase in speed is sensed by the control system and corrected for by an up-elevator signal causing the aircraft to climb. The net result of operation in the Mach hold mode is that the aircraft is made to climb slowly as fuel is consumed in order to maintain a constant Mach number.

Since the fuel consumption of a jet aircraft decreases as the altitude increases, this mode of operation is desirable for long-range operation; however, there are many times when it is necessary to fly at a constant altitude. Under these conditions the altitude hold mode is selected, the aircraft's flight path angle is controlled by the elevators and the airspeed or Mach number is controlled, either manually or automatically, by use of the throttle.

Modern jet aircraft are often fitted with such a control system; its purpose is to hold the set Mach number in the presence of disturbances, provided that the change in height is not very great. Variations in Mach number can be represented by variations in velocity since:

$$\mathrm{M} = \frac{\mathrm{V}}{\mathrm{a}} = \frac{\mathrm{U_0 + u}}{\mathrm{a}} \qquad (13)$$

A block diagram of a typical system is shown in Fig. 8.

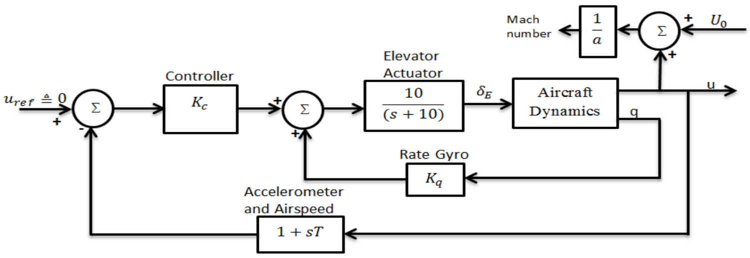

Fig.8 Mach Hold System

A pitch rate SAS has been used as an inner loop in the Mach hold system.

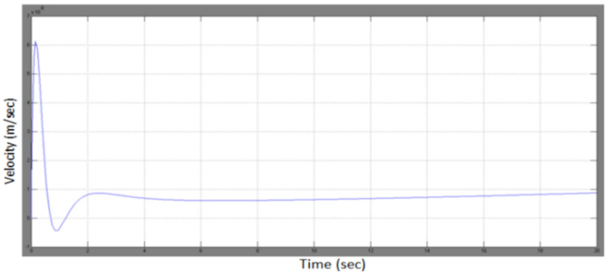

Fig. 9 Response of Mach Hold Control System

It is evident from the Fig. 9, how effectively the speed and Mach number have been held nearly constant.

H. Direction Control System

The purpose of such a system is to allow an aircraft to be steered automatically along some set direction. A block diagram representation of a typical system is shown in Fig. 10.

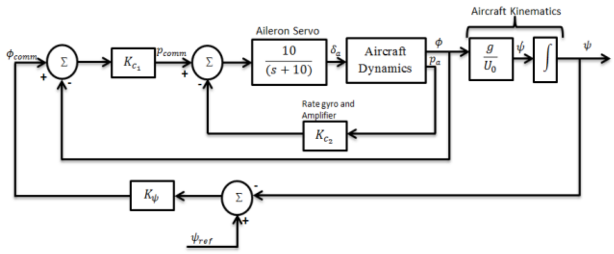

Fig.10 Direction Control System

The heading of the aircraft is taken as its yaw angle, since it is assumed that any turn the aircraft makes under automatic control will be coordinated. Hence, any sideslip angle β **is** zero. The aircraft heading is assumed to be sensed by a gyrocompass of sensitivity 1 V/ deg, hence providing a unity feedback path. The control law for this direction control system is

$$\phi_{comm} = K_\psi(\psi_{reff} - \psi) \tag{14}$$

The unit step response of the system is shown in Chapter 8, in which the corresponding bank angle ϕ is also shown. The long settling time required to achieve the new heading should be noted. Although it has been assumed that the turn was coordinated, there is some residual sideslip angle, β with a peak deviation of 0.38deg. The response of the system can be made more rapid by using an improved value of K_ω. The system was subjected to a sideslip crosswind. The effectiveness of this direction control system can also be seen by considering how well it performs to suppress the effects of a sideslip shear. This direction control system forms the basis of the automatic azimuth tracking systems which use guidance commands from the VOR (VHF Omni range) and ILS (instrument landing system) localizer systems.

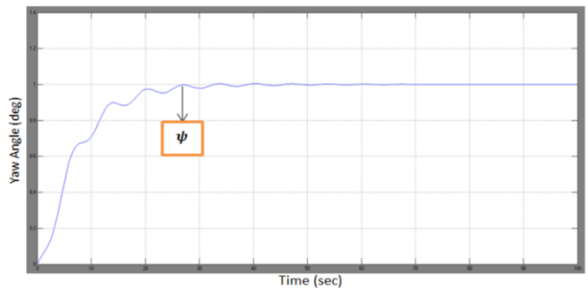

Fig. 11 Step Response of Direction Control System

The unit step response of Direction Control System is shown in the response Fig. 11.

I. Heading Control System The heading angle λ of an aircraft is defined by:

$$\lambda = \beta + \psi \tag{15}$$

The direction control system operated by means of coordinated turns, thereby ensuring that the sideslip angle β was effectively zero. The required turning manoeuvre to be effected by means of the ailerons. If rudder use is involved, then it would seem that the yaw angle, ψ could be controlled by means of a yaw damper system, and with sufficient sideslip suppression could provide the basis of a heading control system. A block diagram of heading control system is shown in Fig. 12.

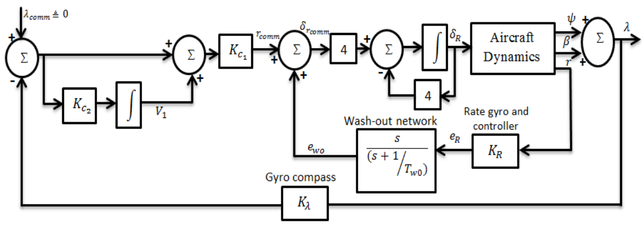

Fig.12 Heading Control System

A heading signal is usually obtained from a gyrocompass and its sensitivity K_λ is 1 V /deg. Using a value of K_{c_1} of 0.875, with K_{c_2} = 0.01 results in a stable, but lightly damped and oscillatory, closed loop system.

The response to an initial error of 1^0 in heading is shown in the response Fig.13 for value of yaw damper gain, $K_r = 0.3$.

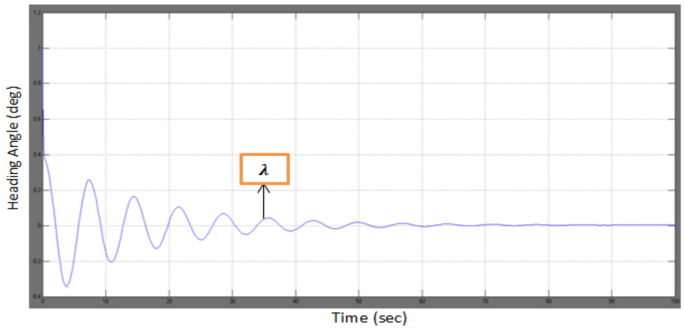

Fig. 13 Response of Heading Control System

J. Glide Path Coupled Control System

This system uses the output signal from the airborne glide path receiver as a guidance command to the attitude control system of the aircraft. The loop is closed via the aircraft kinematics which transforms the pitch attitude of the aircraft into a displacement from the preferred descent path, the glide path, into the airport. The glide path angle is denoted by γ_G , and its nominal value is -2.5deg. If an aircraft is flying into an airport, but it is displaced below the glide path by a distance, d, that distance is negative. If the value of the aircraft's own flight path angle is - 2.5deg, the displacement is 0. Any angular deviation from the Centre-line of the glide path transmission is measured by the airborne glide path receiver that deviation depends upon the displacement, d, and the slant range from the transmitter. Since the value of γ_G is so small, it is customary to regard the slant and horizontal ranges, R and x**,** respectively, as identical; the correct relationship is:

$$x = R\cos 2.5^0 \qquad (16)$$

In this section, xand R, are taken identical. Therefore, the angular deviationΓ is defined as:

$$\Gamma = \frac{d}{R} \qquad (17)$$

Where Γ is in radians.

The component of the airspeed which is perpendicular to the glide path is $U_0 sin\Gamma$; this quantity represents the rate of change of the displacement, i.e.:

$$\dot{d} = U_0 \sin\Gamma \cong {(U_0}/{57.3)}\,\Gamma \qquad (18)$$

However, the aircraft's flight path angle is less than 2.5deg, therefore Γ is positive

$$\Gamma = (\gamma + 2.5^0) \qquad (19)$$

And d is positive.

The aircraft flight path angle γ is defined by:

$$\gamma = (\theta - \alpha) \qquad (20)$$

The block diagram of a typical glide path control system is shown in Fig. 14.

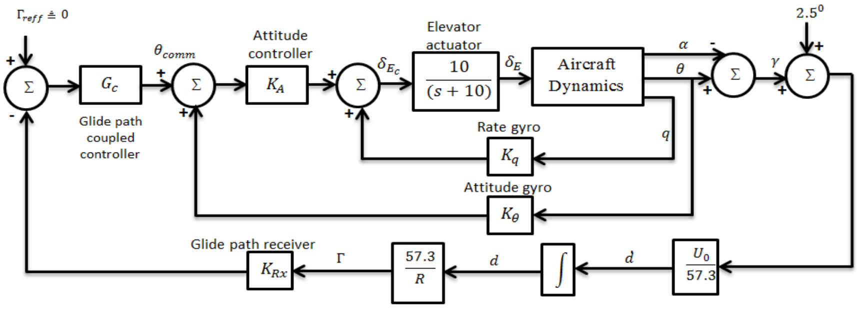

Fig.14 Glide path coupled control system

Consequently, the flight path angle is most effectively controlled by using a pitch attitude control system, with a pitch rate SAS as an inner loop, to effectively control any changes in the angle of attack which may arise as a result of the elevator's being used to drive the aircraft back onto the glide path.

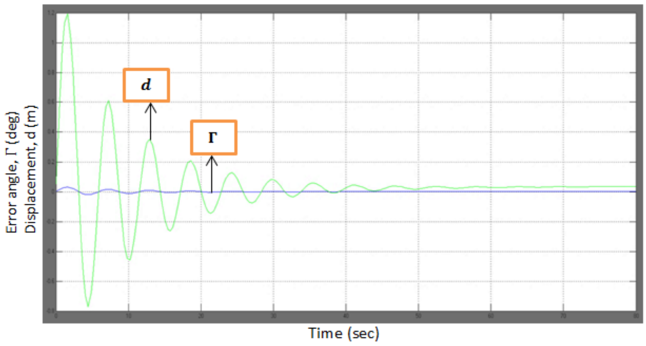

Fig. 15 Response of Glide Path Coupled Control System

K. Automatic Landing System

Although the contribution to the development of airborne automatic landing systems has been international, the basis of most of the operational systems in service is the systemdeveloped in the UK by the Blind Landing Experimental Unit of the Royal Aerospace Establishment. It makes use of the ILS, and the entire automatic landing segment is made up of a number of phases which are shown in Fig. 16.

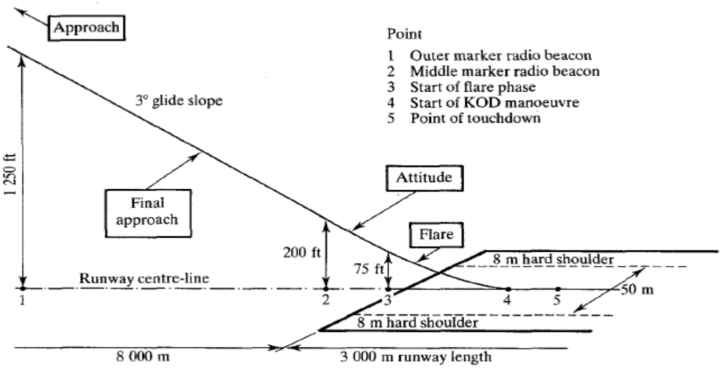

Fig.16 Aircraft Automatic Landing

Usually the point of touchdown, which is aimed for, is 300 m from the runway threshold which is the nominal location of the glide path transmitter. Assuming that the airspeed does not change significantly throughout the flare trajectory, then:

$$h_0 = U_0 \sin\gamma = U_0 \sin(2.5^0) = -2.5\text{m/s} \qquad (21)$$

Assuming landing speed for CHARLIE-1 as 57.3 m/s

$$\dot{h} = \frac{-h_0}{\tau} e^{-t/\tau} = \frac{-h}{\tau} \qquad (22)$$

If the time to complete the exponential flare is taken as 5τ then:

$$(x + 300) = U_0 5\tau = 286.5\tau \qquad (23)$$

$$\dot{h_0} = {}^{-h_0}/_{\tau} \qquad (24)$$

Hence:

$$-2.5 = {}^{-h_0}/_{\tau} \qquad (25)$$

$$\therefore h_0 = 2.5\tau \qquad (26)$$

From Figure Flare Trajectory

$$h_0 = x \tan 2.5 = 0.0435x \qquad (27)$$

$$\tau = \left(\frac{0.0435}{2.5}\right) x \qquad (28)$$

$$x + 300 = \left(286.5 \times \frac{0.0435}{2.5}\right) x \qquad (29)$$

$$\therefore x = 75.3\text{m} \qquad (30)$$

$$\therefore h_0 = 3.25\text{m} = 10.65\text{ft} \qquad (31)$$

$$\tau = 1.3\text{sec} \qquad (32)$$

Hence, the ideal flare manoeuvre is assumed to take 6.5 s to completion. The law which governs the flare trajectory is given by:

$$\dot{h} = -0.77h \qquad (33)$$

A block diagram of an automatic flare control system is shown in Fig. 17.

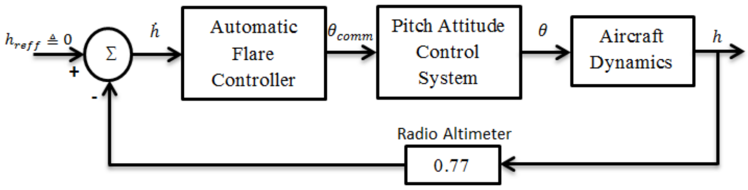

Fig.17 Block Diagram of Automatic Flare Control System

Note that the pitch attitude control system is used: change in θ results in a change in flight path angle, and consequently, a change in height. Because the heights involved are very low, an accurate measurement of height is necessary for this control system and so a low range altimeter is used. Obviously the methods of modern control theory can as easily provide a feedback control law to achieve automatic flare control.

The pitch attitude control system is used as change in pitch angle results in a change in flight path angle and a change in height. Because the heights involved are very low, an accurate measurement of height is necessary for this control system, a low range altimeter is used. The results of a simulation of such an automatic flare control system for a particular flare entry condition is shown in the Fig. 18 below.

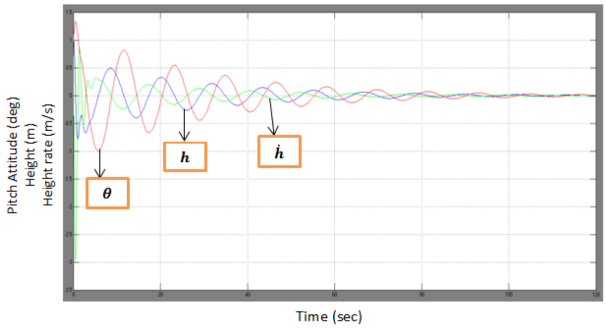

Fig. 18 Response of Automatic Flare Control System

The pitch attitude control system is used as change in pitch angle results in a change in flight path angle and a change in height. Because the heights involved are very low, an accurate measurement of height is necessary for this control system, a low range altimeter is used.

V. Conclusion

This paper has been dealt with detailed studies of automatic control systems which control aircraft flight variablessuch as airspeed, Mach number and height, rather than path variables such asheading. Direction and heading control are treatednext, so that they can be used as elements in the automatic tracking systems whichdepend upon the radio transmissions and appropriate airborne receivers forVOR, ILS localizer and glide path to obtain the appropriate guidance commandsfor these tracking systems. After a detailed consideration of ILS localizer and ILSglide-path-coupled systems the automatic landing,including a flare phase which allows anaircraft to land automatically is considered. The results obtained are found to be satisfactory.

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