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Design and Simulation of Attitude Control Systems for CHARLIE Aircraft

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Abstract- **The concept of stability and control, Aircraft dynamics and modern control theory culminate in the design of particular, important Automatic Flight Control Systems (AFCS) design. Also an understanding dynamic response of the aircraft to atmospheric turbulence and structural flexibility is also essential for the successful design of an AFCS. Among several fundamental modes of AFCS, the attitude control systems for CHARLIE Aircraft are implemented in this paper at various flight conditions. Automatic flight control systems for maintaining the attitude angles of an aircraft or for changing the aircraft attitude to a new command value are introduced. This paper emphasizes the principles of negative feedback control which are common to many varieties of attitude control systems used on aircraft in which the pitch attitude control system is dealt with extensively in this paper. The use of pre filter in conjunction with these types of AFCS to obtain the required handling qualities in the controlled aircraft and the use in such systems of phase advanced compensation networks or a roll rate damper as an inner loop to achieve the required dynamic response and gain scheduling as a means of maintaining the same closed loop performance over as much of the flight environment as possible is also treated in this paper. Finally the unwanted results of tight roll control, such as roll ratchet or pitching motion due to rolling are also treated briefly in this paper. SIMULINK is proposed to implement attitude control systems as they are important to form the inner loop of an integrated flight path control systems.**

Index Terms— **Stability and control, Aircraft dynamics, Attitude control systems, pitch and roll angle control systems.**

I. INTRODUCTION

Attitude control system find an extensive use on all modern air crofts. In order to maintain these modern aircrafts in any required orientation in space these attitude control system form the essential functions of an AFCS in response to a direct pilots command or in response to command signals obtain from an aircraft's guidance or weapons systems. An unattended operation of an aircrafts is also possible through these attitude control systems. The command function of an AFCS is attitude hold is often referred as control wheel steering (CWS) mode. Stability Augmentation systems of inform the inner loop of attitude control systems, the attitude systems then form the inner loop of the flight path control system. The attitude control systems use several of the aircraft control system or they may use feedback signals which depend up on the motion variables. Also attitude control systems are consequently more complex in their operation then stability augmentation system.

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 $H\infty$ 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. The controllers are designed using a coupled nonlinear dynamic model, derived for the YF-22 aircraft.

[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 in realizing 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 system integrates a remote controller, a remotely controlled airplane, a camera, a wireless transmitter/receiver, a ground control computer, 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

ISSN 2348 – 7968

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.

[7] One of an unmanned air vehicle (UAV) 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.

[8] This paper presents a technique for measuring, controlling, and stabilizing the attitude of a UAV by using a camera to monitor the visual horizon. The attitude of the aircraft is then measured using the position, shape, and orientation of the horizon profile in the camera image. Another approach is to use the direction of gravity, as sensed by accelerometers, to estimate and stabilize attitude. The use of sensors that provide direct information on absolute orientation is done in this paper.

[9] This paper proposes a new quaternion-based feedback control scheme for exponential attitude stabilization of a four-rotor vertical takeoff and landing aerial robot known as quad rotor aircraft. Despite the additional gyroscopic term the classical model-independent proportional derivative (PD) controller can asymptotically stabilize the attitude of the quad rotor aircraft. The proposed controllers as well as some other controllers have been tested experimentally on a small-scale quad rotor aircraft.

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. PITCH ATTITUDE CONTROL SYSTEMS

A block diagram representing pitch attitude control systems which involve the traditional use of elevator only is shown in Fig. 1. The representation of the dynamics of both the elevator actuator and the sensor of pitch attitude can also be seen in Fig. 1. Therefore the feedback control law being used in this section is given by

$$\delta_E = K_c K_\theta \theta \tag{1}$$

Also it was found that the aircrafts short period frequency ω_{sp}increases and although it's damping ratio ξ_{sp}decreases with increase in the feedback gain $K_c K_\theta$. However, the damping ration of the phugoid ξ_{ph} increases and the period of phugoid motion also increase until the mode became over damped and consequently non oscillatory. In general feedback of the pitch attitude causes the damping of the phugoid mode to increase at the expense of the damping of the short period mode.

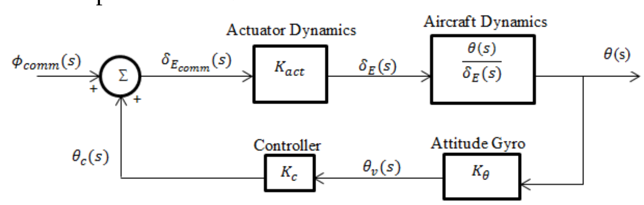

Fig. 1Block Diagram of Pitch Attitude Control System

The characteristic modes for nearly all aircrafts in most flight conditions are two oscillations: one of short period with relatively heavy damping, the other of long period with very flight damping. The period s and the damping of these oscillations vary from aircrafts to aircrafts and with the flight conditions. The short period oscillations are called the "short period mode" and primarily consist of variation of 'α' and 'θ' with very little change in the forward velocity. The long period oscillations is called the "phugoid mode" and primarily consists of variations of 'θ' and 'u' with 'α' about constant. The phugoid mode can be thought of an exchange between potential and kinetic energy. The aircraft tends to fly a sinusoidal flight path in the vertical plan. As the aircraft proceeds from the height point of the flight path to the lowest point, it picks of speed, thus increasing the lift of the wing and curving the flight path until the aircraft starts climbing again and the velocity decreases, the lift decreases and the flight path curves downward. This condition continues until the motion is damped out, which generally requires a considerable number of cycles. However the period is very long and the pilot can damp the phugoid successfully even it is slightly divergent or unstable.

If the phugoid mode is increased, for example, it can only be at the expense of the short period damping. By choosing in such a way that the phugoid mode is heavily damped then

ISSN 2348 – 7968

the phugoid motion will be almost completely absenting the pitch attitude response of the controlled aircraft. The response of the pitch attitude control system used for CHARLIE at flight condition 3 is shown in Fig. 2.

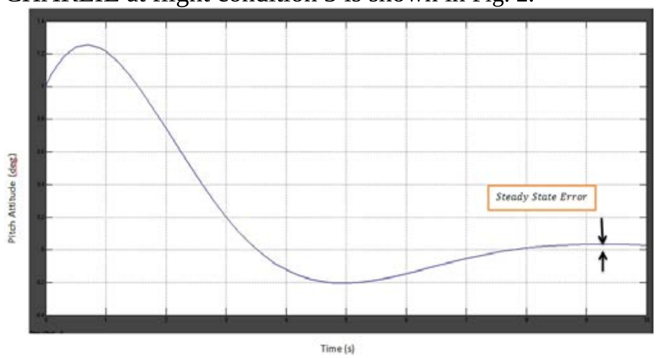

Fig. 2 Response of pitch attitude control system

When the phugoid mode is so heavily damped, any changes which occur in other motion variables (such as speed and height) as a result of pitch command signals are small and the response associated with such variables are well damped with long period. It is for such reasons that the use of pitch attitude feedback to the elevator is used as one of the most successful feedback control techniques in AFCS. However when such a system is said to be type-0 a steady state error would result in response to any step command or disturbance. The steady state error can be removed by introducing an integral term in the control law; the inclusion of this additional term however may reduce the damping of the short period motion. By adding this third term to the control law, the feedback control law is given by:

$$\delta_E = K_c K_\phi \theta + K_b K_I \int \theta \, dt + K_d K_q q \qquad (2)$$

$$= K_2 \theta + K_0 \int \theta \, dt + K_1 q \qquad (3)$$

A typical block diagram representing a pitch attitude control system using control law of Eq. 3 is shown in Fig. 3.

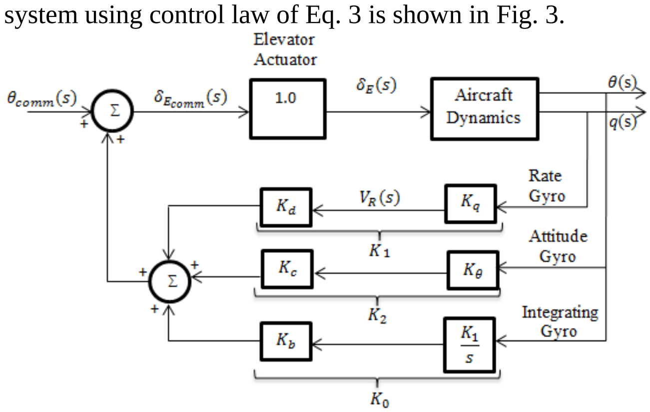

Fig. 3Block Diagram of Pitch Attitude Control System Using Integral term

However this type of control law is not universal and the degree of steady state error that exists in many systems is acceptable with the chosen values of controller gain and. Also, many pitch attitude control systems have a control law which consists only two terms:

$$\delta_E = K_2 \theta + K_1 q \qquad (4)$$

The feedback control is very effective in general use.

The step responses of a pitch attitude control system of CHARLIE-3 are shown in Fig. 4 for the control laws

$$\delta_E = 65.0\theta + 0.78q \qquad (5)$$

$$\delta_E = 0.5\theta + 0.6q + 0.7 \int \theta dt \qquad (6)$$

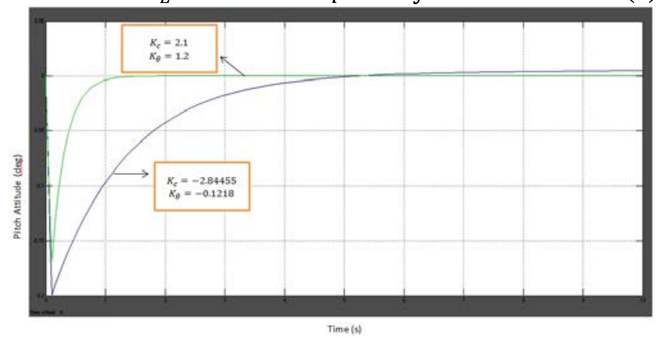

Fig. 4 Damping of phugoid mode comparison

Fig. 4 shows the transient response of same system for the same aircraft and the flight condition using control law where the steady state error has been removed. Therefore it can be found that the improvements which the use of an integral term and pitch rate feedback has made in the dynamic response are evident by comparing Fig. 2 & 4.

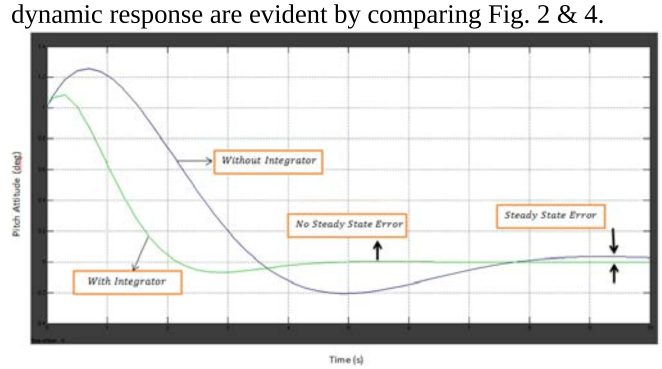

Fig. 5 Response of pitch attitude control system with integral term and pitch rate feedback

Another way to produce a better command response is by first generating the command signal from a pre filter which follows the stick input in a pitch attitude control system. One form of pitch attitude is the C* criterion filter which has a transfer function:

$$\frac{\theta_{comm}(s)}{P_c(s)} = \frac{(1+0.5s)}{(1+0.1s)(1+0.2s)} \qquad (7)$$

The reason for using this C* pre-filter is to improve transient response to disturbances by changing the values of 'K_θ' and 'K_q' very often.

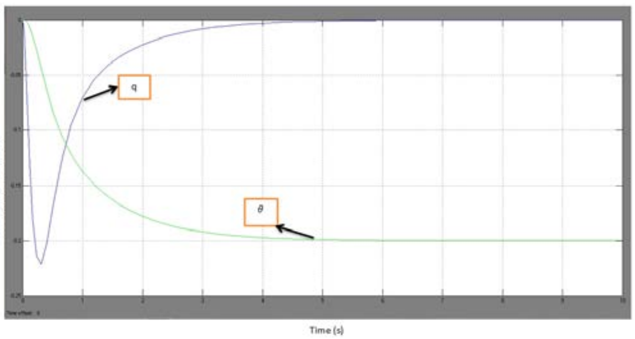

Fig. 6 C* criterion filter with above transfer function

Fig. 6 shows the step response of the system whose response without pre filter and with free filter corresponding to Fig. 1. However a common flight situation occurs in which making pitch attitude to light could be disadvantageous. When the aircraft is flying in the presence of atmospheric turbulence, the pitch attitude tends to hold the pitch attitude at a constant value. The fixity of the attitude opposes the natural tendency of an aircraft to nose into the wind, thereby reducing the acceleration being experienced by the aircraft. It also allows the angle of attack to coincide with the gust. The next result of these two effects is that the accelerations experienced in gusty conditions are higher than they would be otherwise, with a consequent increase of the load being imposed upon the structure of the aircraft.

V. ROLL ANGLE CONTROL SYSTEMS

Roll angle is generally controlled simply and effectively by the ailerons at low to medium speeds on all types of aircrafts. For military aircrafts spoilers are used at high speed. As these spoilers on the wing of the aircraft are very effective means of producing roll moments, sometimes these moments are nonlinear and produce considerable drag when accompanied by a perverse yaw moment. For a swing-wing aircraft, roll control is generally produced by means of differentially moving control surfaces located at tail. Swing-wing contains the spoilers to augment the roll control power of the tail surfaces. Except at high speeds, a differential tail is not very effective at producing rolling moments, since the differential deflection which can be applied is necessarily restricted to allow the same surfaces to be used for longitudinal control. Associated with the rolling (motion) moments produced by this method is a large, adverse yawing moments. Unless particles are exercised in the design of the basic aircraft, it is possible for the spoilers and the differential tail to produce rolling moments which opposes, any yawing which aid. Here the symbol 'δ_A' is used to denote any means of producing rolling moments. The complete transfer function relating bank angle to aileron deflection is given by

$$\frac{\phi(s)}{\delta_{A(s)}} = \frac{K_\phi(s^2+2\xi_\phi\omega_\phi s+\omega_\phi^2)}{(s+\frac{1}{T_s})(s+\frac{1}{T_R})(s^2+2\xi_D\omega_D S+\omega_D^2)} \quad (8)$$

From the above equation 'T_s' can be very large and is given in table below

Aircraft	T_s			
	FC1	FC 2	FC 3	FC 4
CHARLIE	23.42	111.11	97.09	-128.20

Table 1.Spiral Mode Time Constants

However, the spiral mode can correspond to either a slow convergent or a divergent motion.one of the most Important function of any AFCS operating on lateral motion must be therefore to attain a high degree of spiral stability, but it must also improve the others latest flying qualities to that a pilot is not 'worn out' whenever he is flying in atmospheric turbulence. By providing the aircraft with good static stability good spiral stability can be achieved. Therefore in order to achieve degree of dynamic stability desired in roll requires the use of the roll attitude control system such a feedback control system maintains the roll attitude in the preserve of disturbances and the responds rapidly and accurately to roll commands from the pilot or a guidance system. For most aircrafts the following assumptions hold: (i)$T_R \ll T_s$, and (ii) the quadratic term in numerator of Eq. 8 cancels the quadratic term in the denominator.

With the above assumptions, the aircrafts roll dynamics may be represented by a single degree of freedom approximation:

$$\frac{p(s)}{\delta_A(s)} = \frac{K_\phi}{(s+\left(\frac{1}{T_R}\right))} \quad (9)$$

Where:

$$K_\phi = L'_{\delta_A} \quad T_R = -(L'_p)^{-1} \quad p = \frac{d\phi}{dt} \quad (10)$$

A. Bank angle control system

A block diagram representing bank angle control system where the actual response is assumed to instantaneous is shown in Fig. 7.

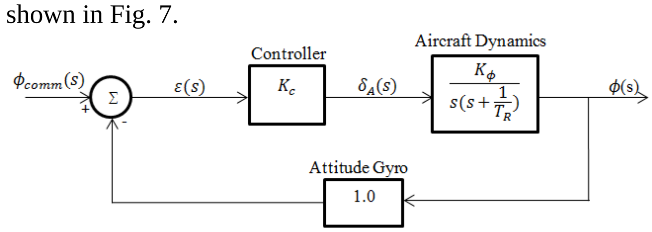

Fig. 7 Bank Angle Control System

From the figure it can easily be shown that

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{K_\phi K_c}{s^2+\frac{1}{T_R}s+K_cK_\phi} \quad (11)$$

Hence:

$$\xi = \frac{1}{2T_R(K_cK_\phi)^{1/2}} \quad (12)$$

$$\omega_n = (K_cK_\phi)^{1/2} \quad (13)$$

For a specific damping ratio of this roll attitude control system, the value of a specific damping ratio of this roll attitude control system, the value of controller gain needed is given by equation:

$$K_c = \frac{1}{K_\phi(2\xi T_R)^2} = \frac{L_p'^2}{4\xi^2 L'_{\delta_A}} \quad (14)$$

Now here for CHARLIE -2 it can be shown that:

$$\frac{P(s)}{\delta_A(s)} = \frac{0.02(s-0.002)(s^2+0.32s+1.2)}{(s+0.01)(s+0.9)(s^2+0.16s+1.2)} \cong \frac{0.21}{(s+0.09)} \quad (15)$$

ISSN 2348 – 7968

$$\frac{\phi(s)}{\phi_c(s)} = \frac{0.21K_c}{(s^2+0.9s+0.21K_c)} \quad (16)$$

Suppose $\xi = 0.6$ is required. Therefore:

$$\frac{\phi(s)}{\phi_c(s)} = \frac{\omega_n^2}{s^2+0.2\omega_n s+\omega_n^2} \quad (17)$$

Where:

$$\omega_n^2 \cong 0.21K_c \quad (18)$$

But$\omega_n = 0.74\, rad\ s^{-1}$, hence:

$$K_c = 2.6V/V \quad (19)$$

Fig 8 shows the response of this system

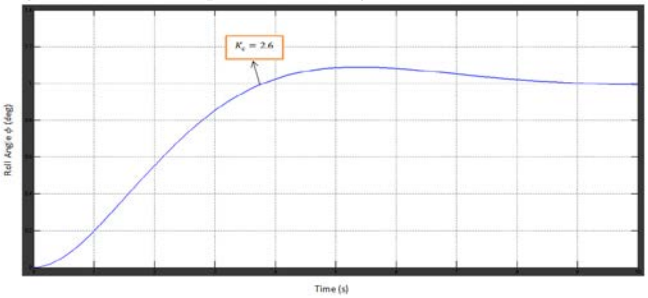

Fig. 8 Step response of bank Angle Control System

B. *Phase advance compensation*

As it is seen in the previous sections, sometimes the cancellation of the numerators and denominator quadratics is in exact. In such cases a significant difficulties for a pilot roll oscillation in rolling motion can lead to serious difficulties for a pilot flying that aircraft. The control law in these types of situations is changed from $\delta_A = K_c\varepsilon$ to:

$$\delta_A = K_c\varepsilon + K_cT_c\varepsilon \quad (20)$$

Where:

$$\varepsilon \triangleq (\phi_{comm} - \phi) \quad (21)$$

The equation introduces damping and corresponds to a phase advance term. The corresponding block diagram is shown in Fig. 9.

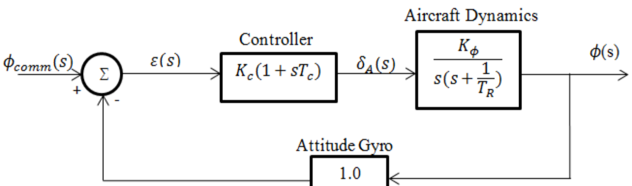

Fig. 9 Bank Angle Control System with Phase Advance

The transfer function of the closed loop system with control law mentioned in Eq. 15 is given by

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{K_cK_\phi(1+sT_c)}{s^2+\left(\frac{1}{T_R}+T_cK_cK_\phi\right)s+K_cK_\phi} \quad (22)$$

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{K_cK_\phi(1+sT_c)}{T_cT_\phi(1+sT_c)(1+sT_\phi)} \quad (23)$$

$$K = \frac{K_cK_\phi}{T_cT_\phi} \quad (24)$$

Let us consider a CHARLIE -2 with following Transfer Function:

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{0.21\,K_c(1+sT_c)}{s^2+(0.9+0.21T_c)s+0.21K_c} \quad (25)$$

Let:

$$s^2 + (0.9+0.21T_c)s + 0.21K_c \cong (s+\tfrac{1}{T_c})(s+\tfrac{1}{T_\phi}) \quad (26)$$

$$\therefore s^2 + (0.9+0.12T_c)s + 0.21K_c$$
$$= s^2 + \left(\frac{1}{T_c}+\frac{1}{T_\phi}\right)s + \frac{1}{T_cT_\phi} \quad (27)$$

Hence:

$$0.21K_c = \frac{1}{T_cT_\phi} \quad (28)$$

$$(0.9+0.21T_c) = \frac{T_\phi+T_c}{T_\phi T_c} \quad (29)$$

Since there are three unknowns,T_c, T_ϕ and K_c and only two equations it is necessary to choose one and evaluate the others. Suppose K_c is chosen to be 10.0, then, by elementary algebra, it can be found that there are two possible values ofT_c which can be used, namely $T_c = 3.107$s or$T_c = 0.2742$s. If the former value is used, the resulting value of T_ϕ is 0.1633, whereas, when T_c is chosen to be 0.2742, the corresponding value for T_ϕ becomes 1.737, i.e. when $T_c = 3.107$s then:

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{4.41}{1+1.737s} \quad (30)$$

When $T_c = 0.2742$ the result is:

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{4.41}{(1+1.737s)} \quad (31)$$

The response of the system corresponding to eq. (A) is ten times faster than the response obtained from a system corresponding to eq. (B). Hence, system (A) would be the preferred system because the quality of rolling motion of the aircraft is better.

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{0.21\,K_c(1+sT_c)}{s^2+(0.9+0.21T_c)s+0.21K_c} \quad (32)$$

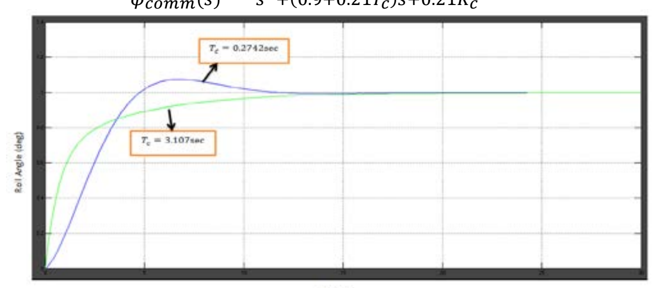

Fig. 10 Step response of Bank angle control system

C. *Roll Rate Inner Loop Damper*

In situations where the feedback or command signals are prone to noise interference, phase advance compensation is often unsuccessful. In order to achieve a considerable freedom in arriving at the required dynamic performance of the roll angle system is to employ a roll rate inner loop damper. The roll damping of the aircraft can be considerably augmented by such on inner loop to values even greater than that needed by the roll angle system thereby achieving good steady state performance and the required transient response. A Block diagram of such roll angle system is shown in Fig. 11 where the actuator dynamics is represented by a simple first order lag.

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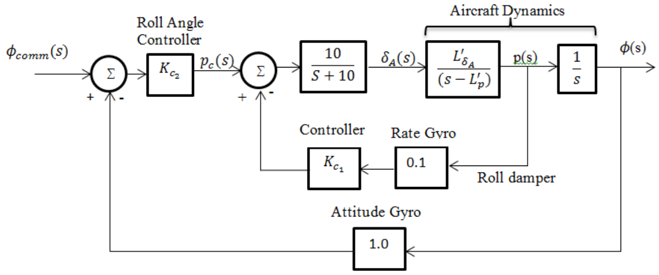

Fig. 11Bank Angle control system with Roll rate inner loop damper

This technique when implemented required another motion sensor, a rate gyroscope for use in the roll damper. Referring to Fig. 11 a closed loop transfer function of the roll angle control system is established as

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{L'_{\delta_A} K_{c_2}}{\{s^2 + s(L'_p + 0.1K_{c_1}L'_{\delta_A}) + K_{c_2}L'_{\delta_A}\}} \quad (33)$$

By using the roll state inner loop damper, the frequency of the roll angle system can be controlled by K_{c_2} and the damping by K_{c_1}. In order to understand the bank angle control system with roll rate inner loop damper let us consider the same CHARLIE-2 in which the single degree of freedom approximation for rolling motion as a result of aileron deflection can be approximated by the transfer function:

$$\frac{p(s)}{\delta_A(s)} = \frac{0.21}{(s+0.9)} \quad (34)$$

If the system used as a roll angle control system is that represented by Figure 11, then the corresponding closed loop transfer function is:

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{0.21k_{c_2}}{s^2 + (0.1*0.21*K_{c_1} + 0.9)s + 0.21K_{c_2}} \quad (35)$$

System A- If k_{c_2} is chosen to be, say, 10.0 and K_{c_1} is selected to be 31.55, and then the characteristic polynomial of the roll angle system becomes:

$$(s^2 + 1.5625s + 2.1 = (1 + 3.107s)(1 + 1.1533s) \quad (36)$$

However, since phase-advance is not being used, there is no numerator term and the factor $(1 + 3.107s)$ is not cancelled. As a result, the response of this system, although heavily damped, is sluggish.

System B- A better choice of K_{c_1} is 95.156 (k_{c_2} remains fixed at 10.0), for this results in the system being critically damped, i.e.:

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{1}{(1+0.69s)^2} \quad (37)$$

The step responses of systems A and B are shown in Fig 12; the superiority of B is evident from inspection.

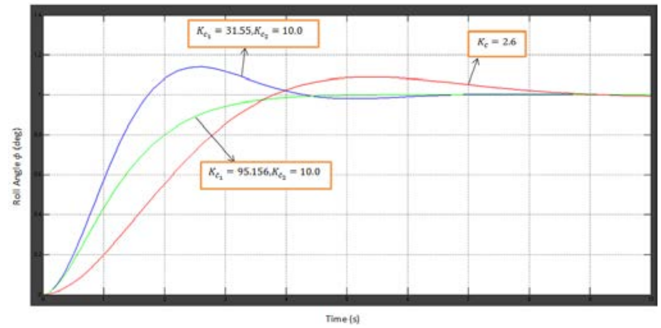

Fig. 12 Single Degree of freedom Approximation when bank angle control system uses roll rate inner loop damper

D. Use of Yaw term in Roll Control Law

If the control law being used in a roll angle control system is modified to become

$$\delta_{A_{comm}} = K_{c_1}\phi + K_{c_2}\psi \quad (38)$$

Then the mode associated with the yawing motion of the aircraft can then become a subsidence mode and when K_{c_2} increased its damping would increase substantially dutch roll damping is also decreased.

The best practical arrangements can be found from experiment and flight test when $K_{c_1}/K_{c_2} = 1.0$. The step response of roll angle control system used with CHARLIE -2 and using the control law for different values of K_{c_1}/K_{c_2} is shown in Fig. 13. It is evident that the best choice of K_{c_1}/K_{c_2} is unity when these resorts are compared with these shown in Fig. 13.

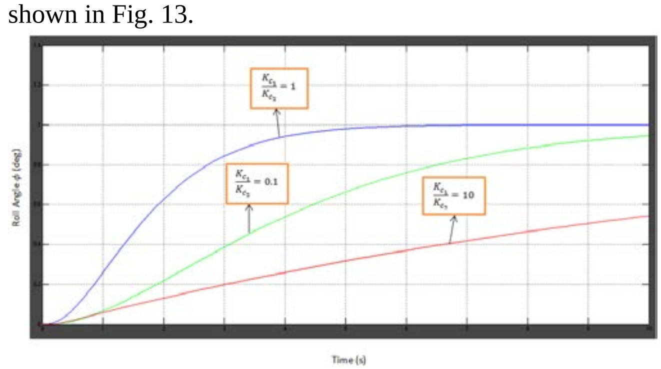

Fig. 13 Step response of Bank angle control system with Yaw term added

E. Roll ratchet caused by excessive roll damping

Most of the modern fighter pilots experienced excessive values of rolling accelerations in various flight experiments when trying to reach some desired value of roll rate. To avoid such accelerations the pilot must apply more slowly the input to the roll control system through the primary flight control. But since the pilot's reaction is sudden and instinctive, the closed loop system formed by the pilot and the aircraft dynamics oscillate in roll. This particles oscillatory motion occurring at a high frequency (1.8-3GHz) is referred to as 'roll ratchet. Pilots encounter with this phenomenon in excessive roll damping situations.

ISSN 2348 – 7968

If the closed loop transfer function of a roll damper given by

$$\frac{\phi(s)}{p_{comm}(s)} = \frac{K}{s(1+sT)} \tag{39}$$

If the damping is large T→ 0, the above equation can be approximated to equation

$$\frac{\phi(s)}{p_{comm}(s)} = \frac{K}{s} \tag{40}$$

When a pilot closes the command loop around a roll damper SAS the system is represented as shown in figure. The mathematical model (used to represent) the pilot as shown in fig represents a proportional gain K_p followed by a pure time delay 'τ' representing pilots reaction time of about 0.13sec. Therefore:

$$\frac{\phi(s)}{\phi_{comm}(s)} = \frac{\hat{K}e^{-st}}{s+\hat{K}e^{-st}} \tag{41}$$

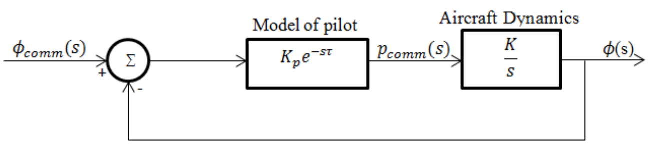

Fig. 14 Pilot-in-the loop roll ratchet

In this, pilot in the loop shown in figure, when the loop gain has a value of say 12 and the time delay function is approximated by$e^{-st} = (2-\tau s)/(2+\tau s)$, then:

$$\frac{\phi(s)}{P_{comm}(s)} = \frac{92.512(2-0.13s)}{s^2+3s+185} \tag{42}$$

Therefore, the system will oscillate with very little damping ($\xi = 0.01$) at a frequency of 13.6rad/sec in response to a unit step function. The result of applying a unit step function for equation (3) is shown in figure. It is clearly evident that the roll ratchet oscillation is at a frequency of 13rad/sec.

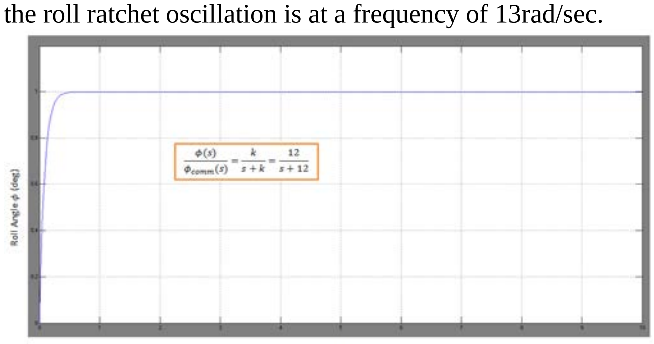

Fig.15 Bank angle control system: Pilots reaction instantaneous

The other figure shows the step response for the same simulation where T is equation is not entirely negligible. The step response for the first case 'T' being 0.01 and for the second case 'T' being 0.2 is shown in Fig 16.

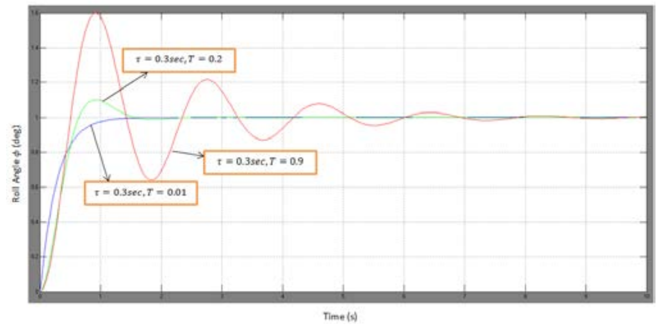

Fig.16 Bank angle response with pilot reaction time of 0.3 sec

From the simulation it can be seen the roll ratchet is only evident (vanished) in first case and in second case where 'T' has increased the roll damping reduced there by roll ratchet vanishes.

CONCLUSION

In this paper, automatic control systems for maintaining the attitude angles of an aircraft, or for changing an aircraft's attitude to a new command value, are introduced. In a pitch attitude control system when attitude gyro is used in feedback a steady state error is observed. This steady state error is removed by introducing an integral term in the control law. In the phase advanced compensator when T_c is higher, then the system is faster than the response obtained from a system corresponding to lower T_c value. Thus the quality of rolling motion of the aircraft is better when T_c is higher.

By using roll damper as an inner loop in bank angle control system, the frequency of roll angle system can be controlled by K_{c_2} and the damping can be controlled by K_{c_1}. By using a yaw term in the roll control law and when these results are compared with step response of bank angle control system, the best choice is rectified as $K_{c_1}/K_{c_2} = 1$.

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