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Design of a Flight Control System in Compliance with Fling and Handling Quality Requirements

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Abstract- *This paper presents the design of a flight control system (FCS) that offers the Aerosonde Unmanned Vehicle (UAV) with stability and control characteristics and capabilities of a classical aerial piloted aircraft.*

The design problem was formulated with a narrow focus on the following aspects of the UAV.

- *The un-augmented open loop 6-DOF Aerosonde exhibits poor stability and control characteristics in compliance with MIL-F8785C flying quality standards.*
- *As in most classical fixed wing aircrafts, the control inputs to Aerosonde result in pitch, roll and yaw rate responses (rate command characteristics). This has made manual trimming of the UAV, at a given flight condition, with a human pilot-operator closing the loop, a daunting task.*

In the first phase of the design, a stability augmentation system was developed to improve the stability and control characteristics of the UAV compliance with MIL-F8785C flying qualities standard. In the second phase, a Rate-Command-Attitude-Hold system has been implemented to the FCS for longitudinal control of the UAV. In the third phase, an Auto-Stabilizer was deigned to control the Lateral-Directional command-response characteristics of the UAV.

Both the Longitudinal and Lateral-Directional FCSs have been tested and simulated using linear and non-linear flight dynamic models of the UAV.

Keywords: *UAV, Flight Control, Stability and Control, Flying Qualities*

Nomenclature

A – State matrix
p – Roll rate (deg/s)
r – Yaw rate (deg/s)
v – Lateral velocity (m/s)
h – Altitude (m)
x – State vector
K_w – Normal velocity gain constant
K_q – Axial velocity gain constant
K_r – Yaw rate gain constant
K_{ari} – Aileron-rudder interlink gain
q_d – Pitch rate demand (deg/s)
T_w – Washout filter time constant (s)
Greek Letters

B – Input matrix
q – Pitch rate (deg/s)
u – Axial velocity (m/s), Input vector
w – Normal velocity (m/s)
y – Output vector
K_u – Axial velocity gain constant
K_θ – Pitch angle gain constant
$K_{\varepsilon q}$ – Pitch rate integral gain const
K_p – Roll rate gain constant
K – Gain vector
J – Performance index

ϕ – Roll angle (deg)
θ – Pitch angle (deg)
ψ – Yaw angle (deg)
η – Elevator angle (deg)
ζ – Rudder angle (deg)
ξ – Aileron angle (deg)
τ – Engine throttle position
β – Sideslip angle (deg)
ζ_s – Short period pitch oscillation damping ratio
ζ_p – Phugoid damping ratio
ζ_{dr} – Dutch roll damping ratio
ω_p– Phugoid natural frequency (rad/s)
ω_{dr} – Dutch roll natural frequency (rad/s)
T_r – Roll subsidence time constant
ε_q – Integral of pitch rate error
ω_s – Short period natural frequency (rad/s)
ω_w– Washout filter break frequency (rad/s)

1 INTRODUCTION

UAVs have become a great source for future aerial operations. However, in Sri Lanka, UAV technology has been in a slow growth. The research work carried out by Tenakoon and Munasinghe (2008) to develop a UAV controller system provides a UAV good manoeuvrability with minimum overshoot settling time without oscillation. However, it would not guarantee flying and handling quality requirements of a UAV. Flying and handling quality requirements are of high importance, if the UAV is to be controlled by a human operator.

The proposed UAV for this research is Aerosonde which has been built by Aerosonde Ltd. It is a small UAV designed to collect weather data over oceans and remote areas. The aerodynamic design of the UAV is largely classical and therefore building a similar model for further research work only needs a reasonable and a manageable effort.

1.1 Aim

In general, UAVs are controlled autonomously or from a ground station human controller. In either event, a suitable flight control system should be implemented on the UAV to achieve objective tasks and missions.

Due to poor dynamic properties of the natural Aerosonde airframe (in compliance with MIL-F-8785C flying quality standards), it is purely impossible for a human operator to control the open loop UAV. Therefore, the aim of this ***flight control system*** (FCS) design is to make it possible for a human operator to control the UAV. A classical flight dynamics principle defines the properties of FCSs that a piloted aircraft should have. This FCS design incorporates those properties and therefore it enables the UAV to attain control characteristics of a classical piloted aircraft.

In addition, the design can later be extended to implement robust, optimal and trajectory planning control algorithms.

1.2 Aerosonde Dynamics

For this research, the full non-linear Aerosonde dynamic model available on Aerosim Matlab/Simulink block set has been used.

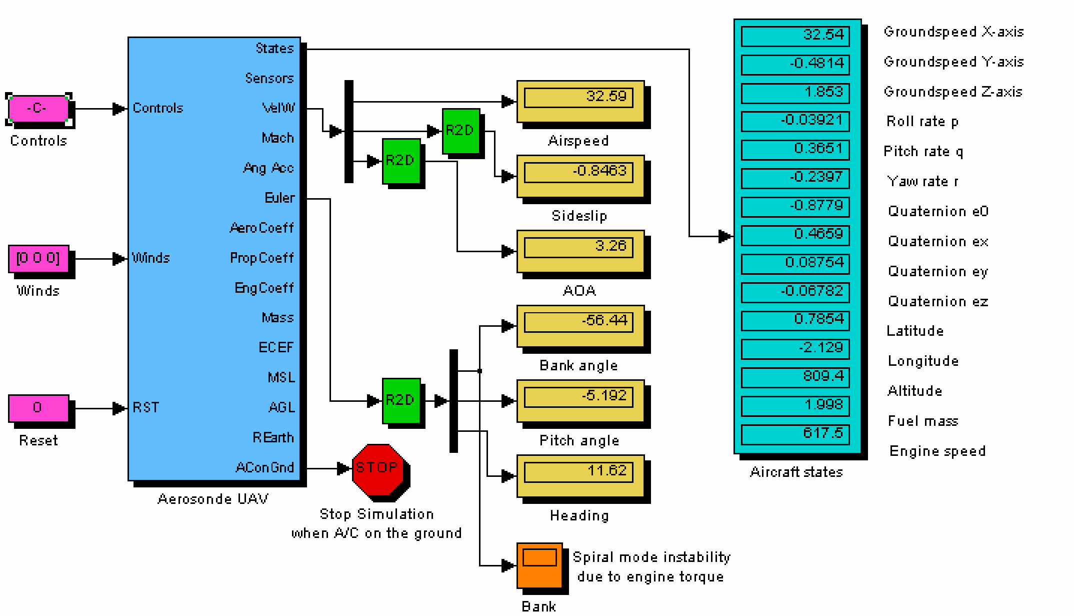

Figure 1 - Nonlinear Aerosonde open loop model

The control inputs to the UAV are deflections of Flap, Elevator, Aileron & Rudder control surfaces and change in Engine throttle position. In addition, the above model allows controlling of fuel mixture and ignition inputs and however, they are set at 13 and 1 for all operations of this research.

1.2.1 Trimming and Linearization

Before applying any flight control algorithm the nonlinear model (Aerosim Aerosonde model) should be trimmed (steady-state equilibrium) and linearized.

<u>Trimming</u>

By definition, steady-state wings level flight and steady turning flight conditions are permitted to be trimmed flight conditions. If the change in atmospheric density with altitude is negligible, a wings-level climb and a climbing turn can also be taken as trimmed flight conditions (Cook, 1997).

To design the objective FCS, the UAV is trimmed at the below mentioned wings-level flight condition with the following constrains applied on the states.

$$\phi, \dot{\phi}, \dot{\theta}, \dot{\psi} \equiv 0 \quad (\therefore p,\ q,\ r \equiv 0)$$

Flight condition:

Trim airspeed – 30 m / s
Trim altitude – 1000 m
Trim bank angle – 0 rad
Flap setting – 0

The result obtained is:

$$x_{trim} = \begin{pmatrix} u_e = 29.99m/s \\ v_e = 0.01m/s \\ w_e = 0.74m/s \\ p_e = 0\deg/s \\ q_e = 0\deg/s \\ r_e = 0\deg/s \\ \phi_e = -0.02\deg \\ \theta_e = 1.41\deg \\ \psi_e = -0.44\deg \\ h_e = 1000m \end{pmatrix}, \; u_{trim} = \begin{pmatrix} \eta_e = 2.02\deg \\ \xi_e = 0.46\deg \\ \zeta_e = 0.05\deg \\ \tau_e = 1.1054 \end{pmatrix}, \; y_{trim} = \begin{pmatrix} V_e = 30m/s \\ \beta_e = 0.02\deg \\ \alpha_e = 1.41\deg \\ \phi_e = -0.02\deg \\ \theta_e = 1.41\deg \\ \psi_e = 359.56\deg \\ h_e = 1000m \end{pmatrix} \quad (1.1)$$

The trim state can be validated by simulating the model for few seconds with $u = u_{trim}$ and $x_0 = x_{trim}$.

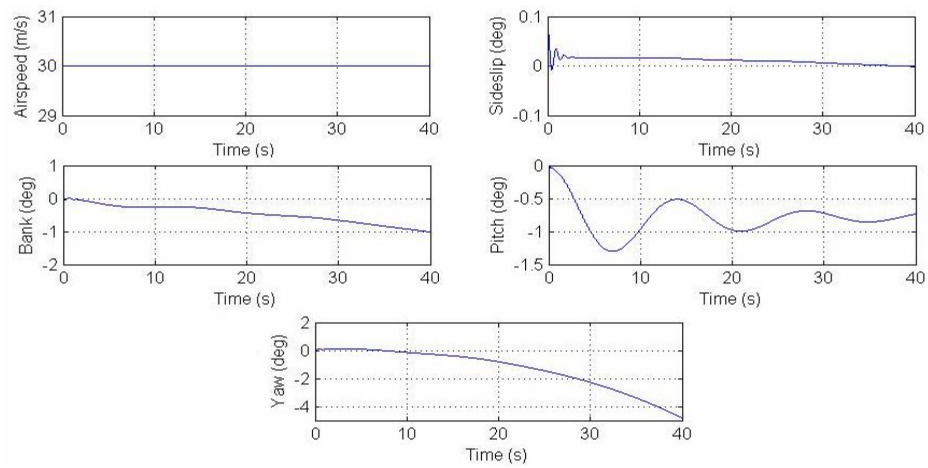

Figure 2 - Nonlinear Aerosonde in trim

<u>Linearizing</u>

The next objective is to obtain a linear approximation of the non-linear trimmed UAV model. Linear approximations are much more desirable to work with as it is easier to predict the behaviour.

The Matlab *linmod* function is used to linearize the model numerically and this brings us to a state-space representation of the form:

$$\dot{x} = Ax + Bu \quad (1.2)$$

Where: $x = [u\ w\ q\ \theta\ v\ p\ r\ \phi]^T$

Thereafter, the state-space model is decoupled into longitudinal and lateral / directional senses by assuming that the coupling factors are insignificant. After decoupling the state equation, two state-space linear representations are obtained for Longitudinal and lateral-directional dynamics.

Where: Longitudinal states $x = [u\ w\ q\ \theta]^T$
Lateral-Directional states $x = [v\ p\ r\ \phi]^T$

1.2.2 Stability

Two and three dynamic stability modes are described respectively in the longitudinal and lateral-directional senses.
Longitudinal modes: *Short period pitch oscillation and Phugoid mode*
Lateral-Directional modes: *Roll subsidence mode, Spiral mode and Dutch roll mode*

Before deciding the flight control law parameters,
Stability properties of the above modes of the UAV must be checked. Since command-response characteristics of the UAV are expected to be similar to a piloted aircraft, any necessary stability augmentation is done compliance with MIL-F-8785C flying quality standards (Anon, 1980). The American Military Specification MIL-F-8785C specify acceptable stability and control standards, more commonly known as *flying qualities requirements*, that a piloted aircraft should achieve.

1.2.3 Flying Qualities

MIL-F-8785C is defined for a range of aircrafts depending on the *class of the aircraft* and *flight phase* (Cook, 1997; Anon, 1980). Aerosonde is a class 1 aircraft with manoeuvre and mission categories lie in flight phase category C.

The ***Short period pitch oscillation*** damping for the Aerosonde should lie in the following regions. Level 1 flying qualities means a fully functional aircraft with 100% capability of achieving the missions that are described in its flight phase category.

Table 1 - Level 1, 2 and 3 short period pitch oscillation damping

Flight Phase	**Level 1**		**Level 2**		**Level 3**
	$\zeta_{s\,min}$	$\zeta_{s\,max}$	$\zeta_{s\,min}$	$\zeta_{s\,max}$	$\zeta_{s\,min}$
CAT C	0.50	1.30	0.35	2.00	0.25

The *Short period pitch oscillation* natural frequency should lie within the following range.

ω_s: $4.0 <= \omega_s <= 25$ (rad / s)

For the ***phugoid*** mode the flying qualities requirements can be summarized as follows.

$\omega_s / \omega_p > 0.1$

ζ_p:- $0.04 <= \zeta_p$

Dutch roll, ***Roll subsidence*** and ***Spiral*** mode level 1 stability requirements are shown below;

Table 2 - Level 1 lateral-directional flying qualities

Mode	**Level 1 Requirements**
Dutch roll mode	$\zeta_{dr} \geq 0.19$, $\omega_{dr} \geq 1.0$ rad/s, $\omega_{dr}\zeta_{dr} \geq 0.35$ rad/s
Roll subsidence mode	$0 \geq T_r \geq 1.0$ s
Spiral mode	When unstable: $t_{2\phi} \geq 12$ s

It is significant that more emphasis has been placed on the short period stability modes (short period pitch oscillation and Dutch roll) when defining flying qualities requirements.

1.3 Flight Control System Design Techniques

To design aircraft control systems, both classical and modern control techniques are used. The general norm of classical design is loop closure to provide inner-rate feedback around the plant for the purpose of artificial damping improvement. In conjunction with this, standard integral compensator structures are used to eliminate steady state errors. Modern control techniques on aircraft FCSs have been employed since mid 80's (Stevens and Lewis, 1992). However, modern techniques are not highly encouraged on aircraft control due to their dependency on selecting large number of parameters such as *performance index weighting matrices*.

There are many classical Command and Stability Augmentation Systems developed for aircraft control such as *Longitudinal Rate Command Attitude Hold (RCAH)* controller, C* controller (Example: A320), C*-U controller and *Lateral-Directional Auto-stabilizer* (LDA) controller etc.

1.3.1 Longitudinal Flight Control System Design

The RCAH controller has been used for longitudinal control of the Aerosonde as the dynamic structure of the aircraft and its stability and control characteristics should remain classical. One reason for this is that classical control and stability characteristics can easily be tuned in compliance with the Flying Quality standards. In addition, when controlling the UAV on the Simulation or Real-Time the human intuition appreciates classical command-response characteristics. This RCAH design has been successfully used on F-16 fighter jet.

1.3.2 Lateral-Directional Flight Control System Design

The LDA controller is proposed to use for lateral and directional control of the UAV with the same reasoning as mentioned above. Since the trim state is assumed in both lateral and directional senses, the LDA design is mainly concerned with manoeuvring the UAV about the trim state.

2 LONGITUDINAL FLIGHT CONTROL SYSTEM DESIGN

As mentioned in the introduction, the linearized longitudinal state-space representation of the trim state can be shown as below (For elevator inputs only).

$$\begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} -0.2690 & 0.4017 & -0.7248 & -9.7973 \\ -0.5318 & -4.9550 & 29.3296 & -0.2404 \\ 0.3421 & -5.2290 & -5.7174 & 0 \\ 0 & 0 & 1.0000 & 0 \end{bmatrix} \begin{bmatrix} u \\ w \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} -0.2976 \\ -3.4212 \\ -46.308 \\ 0 \end{bmatrix} \eta \quad (2.1)$$

2.1 Flying qualities requirements

With reference to MIL-F-8785C, the longitudinal flying qualities are analysed.

Table 3 - Open loop level 1 longitudinal flying qualities

Mode	Level 1 Requirements	Aerosonde
Short Period Pitch Oscillation	$0.5 \geq \zeta_s \geq 1.30$ $4.0 <= \omega_s <= 25$ (rad / s)	$\zeta_s = 0.396$ $\omega_s = 13.5$ rad / s
Phugoid mode	$\omega_s / \omega_p > 0.1$ $0.04 <= \zeta_p$	$\omega_s / \omega_p = 27.6$ $\zeta_p = 0.274$

It is clear that the short period damping is too low while the mode frequency meets the requirements. The phugoid mode flying qualities requirements are already well within the range. Therefore the feedback gains should ensure the improvement in short period mode damping.

2.2 Pitch Rate Command-Attitude Hold Command and Stability Augmentation System (CSAS)

In order to design the objective CSAS design, a pitch rate feedback proportional gain K_q plus integral (P+I) controller is incorporated on pitch rate feedback to elevator. Since the state-space representation is a multivariable control problem, all the other proportional state feedback gains [K_u, K_w, K_θ] should also be included into the design. Proportional gains are used to provide the system with desired closed loop stability and rate command characteristics while the integral gain, $K_{\varepsilon q}$ drives the rate command error signal to zero.

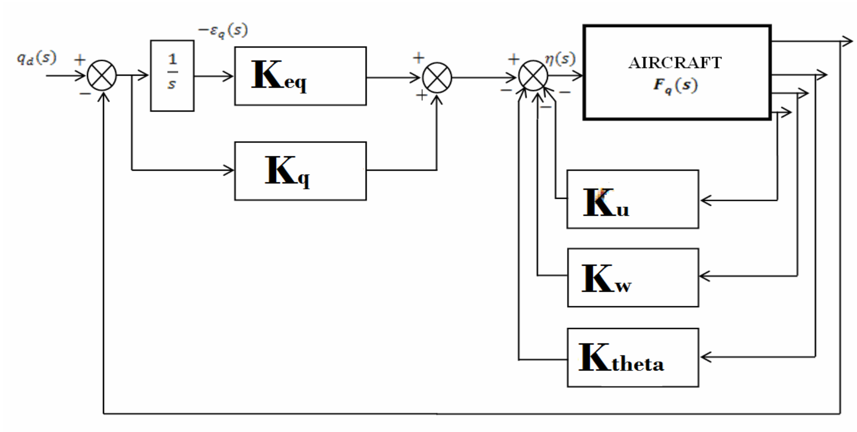

Figure 3 - RCAH system structure

Therefore the design objective is to find appropriate values for the proportional and integral gains with the following constrains applied.

The augmented UAV should meet MIL-F-8785C and the short period dynamics should be *second order like*. Therefore any additional dynamics introduced by the integrator should not be visible to the human controller.

2.2.1 Augmenting the Open Loop State Equation

In order to find appropriate values for the proportional and integral gains it is necessary to have the integral state variable available as an additional state.

Extra state variable: $\varepsilon_q(t) = \int (q(t) - q_d(t))dt$ (2.2)

Where, the state equation may be written: $\dot{\varepsilon}_q(t) = (q(t) - q_d(t))$ (2.3)

Now the integral state equation can be augmented into the original longitudinal state equation as follows;

$$\begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \\ \dot{\varepsilon}_q \end{bmatrix} = \begin{bmatrix} -0.2690 & 0.4017 & -0.7248 & -9.7973 & 0 \\ -0.5318 & -4.9550 & 29.3296 & -0.2404 & 0 \\ 0.3421 & -5.2290 & -5.7174 & 0 & 0 \\ 0 & 0 & 1.0000 & 0 & 0 \\ 0 & 0 & 1.0000 & 0 & 0 \end{bmatrix} \begin{bmatrix} u \\ w \\ q \\ \theta \\ \varepsilon_q \end{bmatrix} + \begin{bmatrix} -0.2976 \\ -3.4212 \\ -46.308 \\ 0 \\ 0 \end{bmatrix} \eta + \begin{bmatrix} 0 \\ 0 \\ 0 \\ 0 \\ -1 \end{bmatrix} q_d \qquad (2.4)$$

Then the open loop longitudinal state equation may be written as;

$$\dot{x} = Ax + Bu + Nv \qquad (2.5)$$

2.2.2 Designing the closed loop system stability

The feedback control law can be defined as follows;

$$u = -Kx + Mv \qquad (2.6)$$

where K is the feedback gain vector and M is the feed forward gain scalar which is assumed to be 1 at this instance (therefore no feed forward gain is illustrated in Figure 1). By substituting the control law into the previous state representation the closed loop state-space equation is obtained in the general form.

$$\dot{x} = [A - BK]x + [BM + N]v \qquad (2.7)$$

where $K = [K_u \; K_w \; K_q \; K_\theta \; K_{\varepsilon q}]$

Now the task is to select an appropriate feedback gain vector K such that it provides level 1 flying qualities in compliance with MIL-F-8785C and objective command-response characteristics. This has been achieved by solving a Linear Quadratic Regulator (LQR) control problem.

<u>LQR control problem</u>

If the open loop state-space representation is $\dot{x} = Ax + Bu$, the problem is to find an optimal constant feedback controller K_0 such that it minimizes the following quadratic cost function.

$$J = \int_0^\infty (x^T(t)Qx(t) + u^T(t)Ru(t))dt \qquad (2.7)$$

where $u(t) = -Kx + Mv$, Q and R are weighting matrices.

LQR Solution:

The Optimal controller is

$$K_o = R^{-1}B^T P \qquad (2.8)$$

where $P = P^T \geq 0$ is the solution of

$$A^T P + PA - PBR^{-1}B^T P + Q = 0 \qquad (2.9)$$

Matrix *P* is a positive definite (or semi-definite) matrix where $x^T Px \geq 0$ for all $x \neq 0$.

The choice of Q: As observed previously, it is desired to obtain level 1 flying qualities of the short period mode and pitch rate command-attitude hold characteristics. The short period mode is significant (dominant) in w and q responses while the integral action is critical in command-response characteristics and therefore Q can be selected such that it gives a considerable emphasis on K_w, K_q and $K_{\varepsilon q}$ gains and less emphasis on K_u and K_θ gains. After a series of trials the following Q matrix was selected.

$$Q = diag\left(10\times10^{-10}, 10\times10^{-4}, 10\times10^{-4}, 10\times10^{-10}, 100\right)$$

R is selected to be 1 to correctly scale the control inputs.

After solving the above problem on Matlab the optimal feedback gain vector was found to be;

K= [-0.0092 0.0897 -0.4535 -0.0120 -9.9799]

<u>Closed loop system analysis</u>

The closed loop state equation,

$$\begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \\ \dot{\varepsilon}_q \end{bmatrix} = \begin{bmatrix} -0.2718 & 0.4284 & -0.8597 & -9.8009 & -2.9700 \\ -0.5634 & -4.6482 & 27.7783 & -0.2815 & -34.1431 \\ -0.0860 & -1.0763 & -26.7158 & -0.5560 & -462.1459 \\ 0 & 0 & 1.0000 & 0 & 0 \\ 0 & 0 & 1.0000 & 0 & 0 \end{bmatrix} \begin{bmatrix} u \\ w \\ q \\ \theta \\ \varepsilon_q \end{bmatrix} + \begin{bmatrix} 0.7142 \\ 8.2109 \\ 111.1390 \\ 0 \\ -1.0000 \end{bmatrix} q_d \qquad (2.10)$$

w and q transfer functions:

$$\frac{q(s)}{q_d(s)} = \frac{-46.3079s\,(s-9.98)\,(s+4.511)\,(s+0.3284)}{s\,(s+4.15)\,(s+0.3297)\,(s^2+27.16s+502.5)}$$

$$\frac{w(s)}{q_d(s)} = \frac{-3.4212\,(s+402.7)\,(s-9.98)\,(s^2+0.2756s+0.1813)}{s\,(s+4.15)\,(s+0.3297)\,(s^2+27.16s+502.5)}$$

The closed loop integral lag pole is at (s + 4.15), which equates to a lag time of 0.241 seconds. Thus even if the lag is visible to the pilot it is less intrusive as it is much faster. However, the effect of the integral lag time constant can be diminished by a proper selection of M feed forward gain.

2.2.3 Designing the feed forward gain

Up to now M was taken to be 1, since no information about the integrator pole was available. The feed forward gain determines the *integral zero*. Since the feedback gain vector K is known, the closed loop state equation may now be written as;

$$\begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \\ \dot{\varepsilon}_q \end{bmatrix} = \begin{bmatrix} -0.2718 & 0.4284 & -0.8597 & -9.8009 & -2.9700 \\ -0.5634 & -4.6482 & 27.7783 & -0.2815 & -34.1431 \\ -0.0860 & -1.0763 & -26.7158 & -0.5560 & -462.1459 \\ 0 & 0 & 1.0000 & 0 & 0 \\ 0 & 0 & 1.0000 & 0 & 0 \end{bmatrix} \begin{bmatrix} u \\ w \\ q \\ \theta \\ \varepsilon_q \end{bmatrix} + \begin{bmatrix} 0.7142M \\ 8.2109M \\ 111.139M \\ 0M \\ -1.0000 \end{bmatrix} q_d \qquad (2.13)$$

It can easily be shown that integral zero is given by;

$z = \frac{K_{\varepsilon q}}{M}$, since the integral gain is -9.9799 and integral pole is at 4.15, M = -2.4

w and q transfer functions with the feed forward gain M:

$$\frac{w(s)}{q_d(s)} = \frac{8.2246\,(s+402.7)\,(s+4.151)\,(s^2+0.2756s+0.1813)}{s\,(s+4.15)\,(s+0.3297)\,(s^2+27.16s+502.5)}$$

$$\frac{q(s)}{q_d(s)} = \frac{111.3242s\,(s+4.511)\,(s+4.151)\,(s+0.3284)}{s\,(s+4.15)\,(s+0.3297)\,(s^2+27.16s+502.5)}$$

Inspection of equations indicates that cancellation of the integral pole is exact and therefore the short period mode is *second order like*. The dynamics introduced by the integrator is not visible to the human controller.

The stability characteristics of the closed loop UAV can be shown as follows;

Table 4 - Closed loop level 1 Longitudinal flying qualities

Mode	Closed Loop Aerosonde
Short Period Pitch Oscillation	ζ_s = 0.606 ω_s = 22.4 rad / s
Phugoid mode	T_1= 1 / 0 = ∞ s T_2= 1 / 0.3297 = 3.03 s
Integrator lag	T_{lag} = 1 / 4.15 = 0.241 s

It is clear that the closed loop UAV holds level 1 flying qualities with the implemented controller. However, a significant change in the phugoid mode can be seen due to the integral feedback. The significant variable of the phugoid mode is θ and the integral of the pitch rat is also pitch attitude θ. Therefore, due to the change in q feedback integral gain the closed loop phugoid mode has considerably modified. The mode is no longer oscillatory. However, a non-oscillatory phugoid is acceptable with reference to MIL-F-8785C technical literature (Anon, 1980).

2.2.4 Implementing the controller design

The corresponding system structure can be realized as follows,

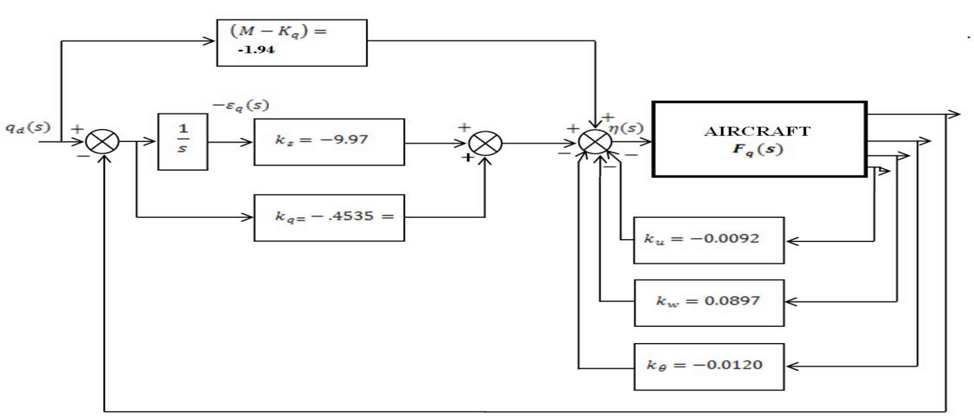

Figure 4 - Equivalent classical P+I controller

Response time histories

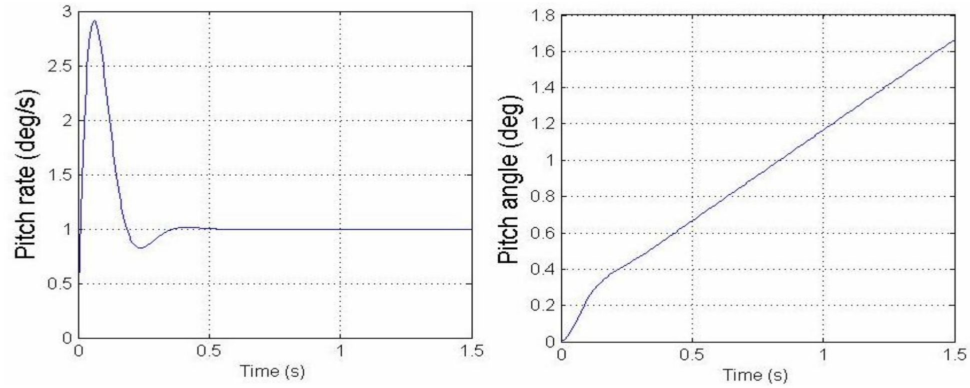

Figure 5 - Closed loop response to a 1⁰ pitch rate demand

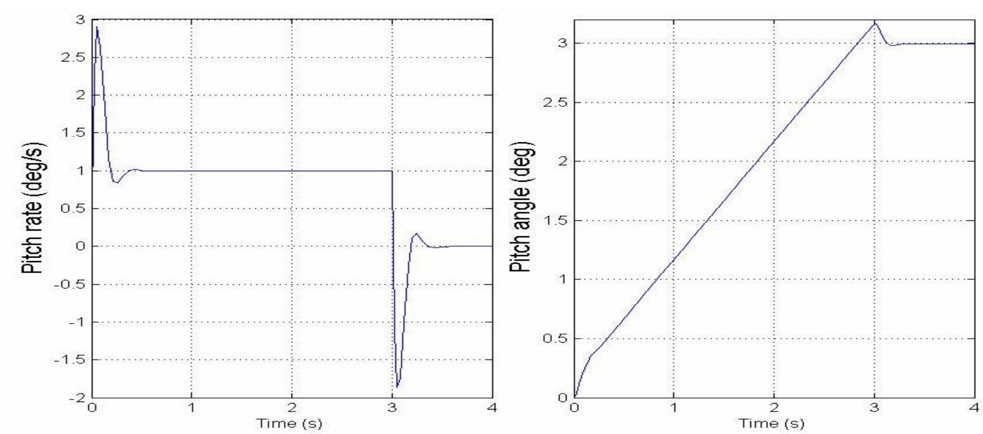

Figure 6 - Closed loop response to a 1⁰ pitch rate demand of 3 second

It is clear that the design objectives have been met successfully. The pitch rate responses like a well damped classical aircraft. When a demand for a pitch rate of $1^0/s$ is held for 3 seconds, the pitch attitude ramps up at $1^0/s$ and when the input is removed the attitude settles to 3^0 (Figure 6). This precisely describes the rate command-attitude hold characteristics of the design.

3 LATERAL-DIRECTIONAL FLIGHT CONTROL SYSTEM DESIGN

The linearized lateral-directional state-space representation of the trim state for aileron and rudder inputs is shown below;

$$\begin{bmatrix} \dot{v} \\ \dot{p} \\ \dot{r} \\ \dot{\phi} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} -0.7655 & 0.7358 & -29.9906 & 9.7973 & 0 \\ -5.0538 & -24.8886 & 11.9786 & 0 & 0 \\ 0.8202 & -3.2283 & -1.2520 & 0 & 0 \\ 0 & 1.0000 & 0.0245 & 0 & 0 \\ 0 & 0 & 1.0003 & 0 & 0 \end{bmatrix} \begin{bmatrix} \beta \\ p \\ r \\ \phi \\ \psi \end{bmatrix} + \begin{bmatrix} -1.9687 & 5.0251 \\ -172.8548 & 3.1102 \\ -6.8154 & -31.7508 \\ 0 & 0 \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \xi \\ \varsigma \end{bmatrix} \quad (3.1)$$

Flying qualities requirements

The MIL-F-8785C flying qualities requirements and Aerosonde lateral-directional mode stabilities are summarized as below;

Table 5 - Open loop level 1 lateral-directional flying qualities

Mode	Level 1 Requirements	Aerosonde
Dutch roll mode	$\zeta_{dr} \geq 0.19$ $\omega_{dr} \geq 1.0$ rad/s $\omega_{dr}\zeta_{dr} \geq 0.35$ rad/s	$\zeta_{dr} = 0.146$ $\omega_{dr} = 7.97$ rad/s $\omega_{dr}\zeta_{dr} = 1.16$ rad/s
Roll subsidence mode	$0 \geq T_r \geq 1.0$ sec	$T_r = 0.24$ s
Spiral mode	When unstable: $t_{2\phi} \geq 12$ s	Stable

The Aerosonde meets level 1 requirement with the exception of the Dutch roll mode where the damping is too low.

3.1 Lateral-Directional Auto-Stabilizer (LDA) Design

To manoeuvre the UAV about the lateral-directional trim state with improved turn coordination a typical lateral-directional auto stabilizer is designed (Cook, 1997).

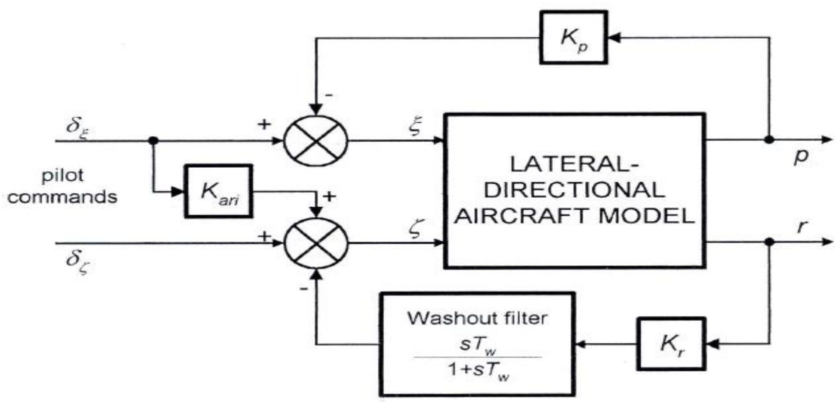

Figure 7 - Lateral-directional auto stabilizer

The LDA design consists of a *roll damper* loop and a *yaw damper* loop. The roll rate feedback gain K_p is used to augment the roll subsidence mode time constant to bring it to an acceptable level. The yaw rate feedback gain K_r is chosen to modify the Dutch roll mode damping ratio. Aileron-rudder interlinks gain K_{ari} is designed to improve roll-yaw coordination in a turn.

During a steady turning flight (with Aileron inputs only), the yaw rate feedback loop would oppose the turn. Therefore, the washout filter is used to prevent this happening by choosing a proper filter time constant T_w.

3.1.1 Design of the Roll Damper

The lateral roll stability of the UAV appears to be adequate and therefore no roll rate feedback to aileron is required ($K_p = 0$).

3.1.2 Design of the Yaw Damper

The loop feedback gain K_r is selected using the yaw rate response to rudder input root locus.

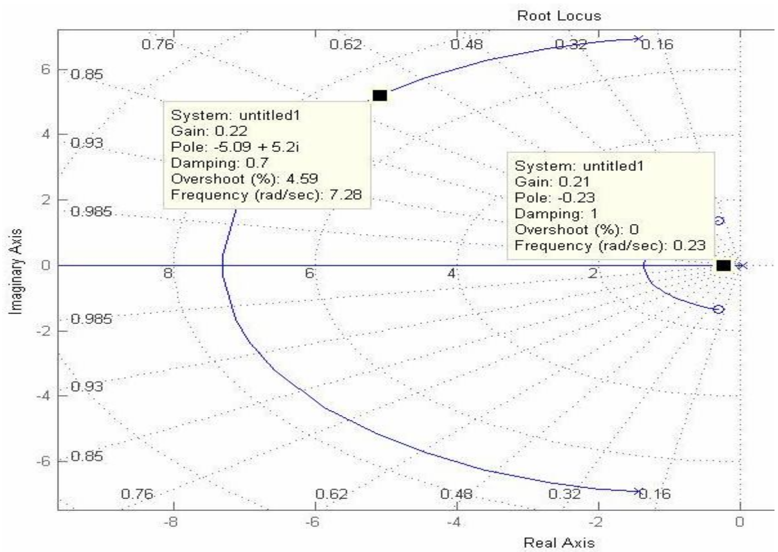

Figure 8 - Root locus plot for yaw rate feedback to rudder

By observing the root locus, it is decided to select a gain of 0.22 for K_r which provides an adequate Dutch roll damping of 0.7. It is also observed that modifications made to the roll and spiral modes due to K_r are minor.

3.1.3 Design of the Washout Filter

The filter is designed in such a way that it attenuates the steady state feedback signal r at low frequencies. However, the feedback signal r should be passed with minimum phase shift and therefore the break frequency ($\omega_w = 1/T_w$) is chosen to be little lower than the Dutch roll frequency. For many piloted aircraft, a filter time constant T_w in between 0.5 and 1 seconds would be satisfactory (Cook, 1997). For the Aerosonde the time constant was chosen to be 0.7s which gives a break frequency of 1.42 rad / s.

3.1.4 Design of the Aileron-Rudder Interlink Gain

The design of the gain K_{ari} is concerned with selecting a value that minimizes sideslip during an aileron command turn. When K_{ari} is selected correctly it will minimize the adverse yaw transient during turn and reduce sideslip to zero. Design of K_{ari} is well described in Stevens and Lewis (1992) where it suggests scheduling K_{ari} as a function of dynamic pressure. However, for the chosen trim flight condition a value of 0.2 is suggested after a series of careful trials.

Response time histories

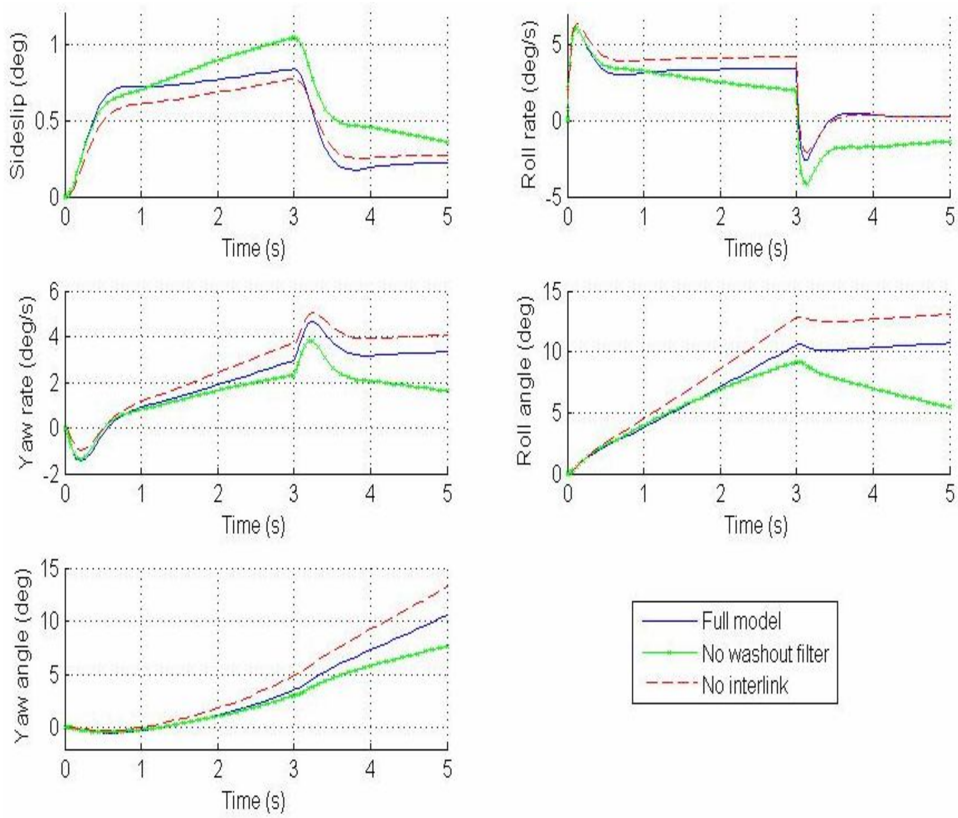

Figure 9 - Response to 1 deg-3s aileron pulse

The turning performance of the aircraft with and without the washout filter and interlink has been compared in the above figure. With the filter, a much steadier yaw rate can be viewed with lesser tendency to roll out. The implementation of the aileron-rudder interlink gain K_{ari} has minimised the sideslip as expected.

It is also observed that manoeuvre capabilities of the UAV about the trim flight condition are adequate. According to above figure, when the aileron input is released, the UAV attains a steady yaw rate with a constant roll angle and a minimum sideslip.

4 IMPLEMENTATION ON THE NONLINEAR MODEL

Finally both the longitudinal and lateral-directional FCSs have been implemented on the non-linear Aerosonde Simulink model.

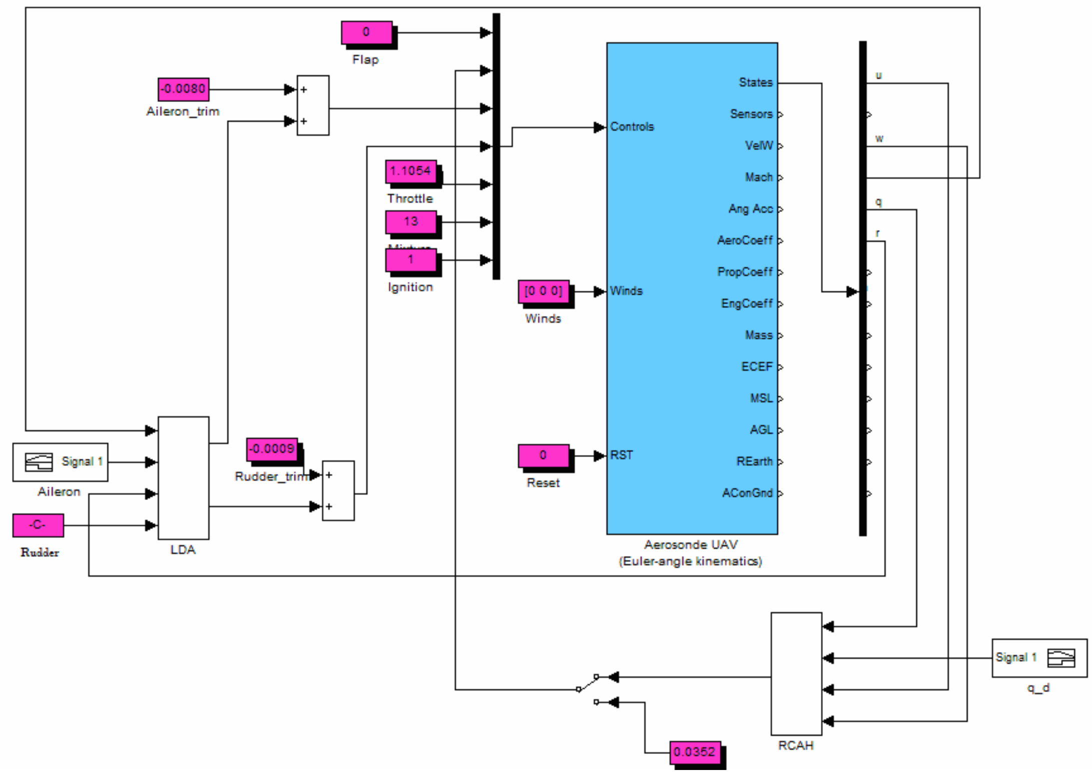

Figure 10 - Nonlinear Aerosonde with RCAH and LDA control loops

4.1 Non-linear Response Time histories

Following are response time histories for a 1.41 deg/s pitch rate demand for 1.8 seconds, .45 deg aileron pulse input for 3 seconds and a continues zero rudder input.

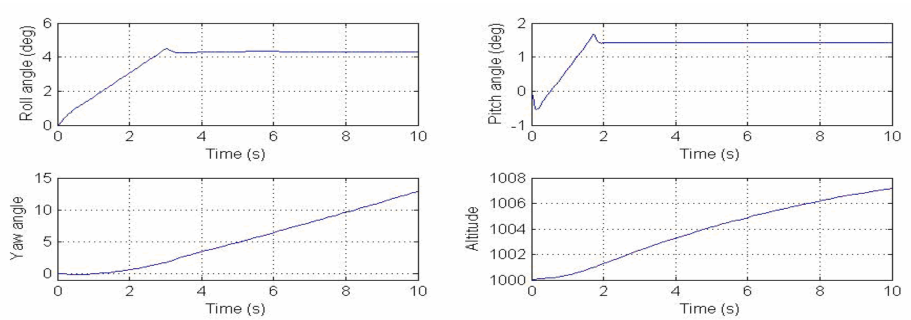

Figure 11 - Modified nonlinear Aerosonde responses

The time histories represent closely related responses to the linear response time histories. However, nonlinear effects are not completely negligible. As illustrated in the linear model, the pitch angle has settled at 1.4 after 1.8 seconds of pitch rate demand and after holding the aileron command for 3 seconds, the roll angle has settled at 4.1 deg as well. However, the altitude keeps increasing with the absence of any altitude holding characteristics of the model.

5 CONCLUSIONS

The full nonlinear Aerosonde model has been trimmed and linearized in order to apply the objective control schemes in compliance with MIL-F-8785C flying qualities requirements to enable the UAV to be controlled by a human operator. Two control schemes were developed in the longitudinal and lateral-directional senses on the linearized model and later the designs were tested on the nonlinear model. The control techniques used for the design are classical theories with the exception of LQR techniques. The FCSs has been proved, to work properly on nonlinear simulation, and to provide command-response characteristics that of a classical piloted aircraft.

However, the design doesn't contain speed and altitude auto-control loops and therefore a constant controlling of the engine throttle is required to maintain the airspeed and altitude. In addition, the FCS parameters have been decided for the chosen trim flight condition. Hence for the design to work over a larger flight envelop, the FCS parameters should be scheduled for changing flight conditions (gain scheduling).

Further this design can be extended to have robust, optimal and path planning control algorithms.

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