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Relook at Aileron to Rudder Interconnect

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ABSTRACT

The implementation of interconnect gain from aileron to rudder surface on the majority of the aircraftis to decrease sideslip which is generated because of adverse yaw with the movement of control stick in lateral axis and also enhances the turning rate performance.The Aileron to Rudder Interconnect (ARI)involves significant part to decouple the Dutch roll oscillations from roll rate response to aileron command. ARI is feed-forward gain whichis susceptible to aircraft system uncertainty. Incorrect ARI gain can lead to side slip buildup which can cause aircraft to depart in case of fault scenarios. Four systematic ARI design methods are proposed. One of the proposed methods which use the norm of ARI transfer function at roll damping frequency is suitable for online reconfiguration of control law. The reconfiguration of ARI gain is illustratedwith the simulation responses of fault scenario case of aileron surface damage.

Keywords: Aircraft; Control law; Flight; Aileron to rudder interconnect gain; ARI; Reconfiguration

NOMENCLATURE

Ca_b	Variation of parameter Ca with parameter b
Ixx,Izz,Iyy	Rotational inertia in X, Z, and Y body axes in Kgm^2
Ixy, Ixz, Iyz	Product of inertia along XY, XZ, and YZ body axes plane in Kgm^2
K_{ARI}	Gain interconnecting between Aileron surface to rudder surface output
K_β, K_p	Feedback path gains of angle of sideslip and roll rate respectively
K_{B_w}	Feedback path gain of side slip rate
L_β	Roll acceleration due to sideslip in deg/sec^2/deg
$L_{\delta a}$, $L_{\delta r}$	Roll acceleration due to aileron and rudder control in deg/sec^2/deg
L_p, L_r	Roll acceleration due to roll and yaw rate and unit is 1/sec
loe, lie	Outboard, inboard elevon position of left side (port side) and unit is in degrees
N_β	Yaw acceleration sideslip in deg/sec^2/deg
$N_{\delta a}$, $N_{\delta r}$	Yaw acceleration due to aileron and rudder control in deg/sec^2/deg
N_p, N_r	Yaw acceleration due to roll and yaw rate in 1/sec
N_y	Lateral accelerometer Output in g's
$\dot{r}$ $\dot{p}$	Yaw and Roll acceleration respectively in deg/sec^2
p, q, r	Roll rate, Pitch rate, Yaw rate in body axis respectively in deg/sec
roe, rie	Outboard, inboard elevon position of right (starboard side) and unit is in degrees
X_e, Y_e, H	Vehicle position in X, Y, Z earth axes
V_T	True airspeed in meter/second
Y_p, Y_r	Lateral acceleration due to roll and yaw rate respectively in meter/second/degree
Y_β	Lateral acceleration due to sideslip in meter /sec^2/deg
$Y_{\delta a}$, $Y_{\delta r}$	Lateral acceleration due to aileron and rudder control in meter /sec^2/deg
α	AoA or Angle of Attack and unit is in degree
β	Beta or Angle of Sideslip and unit in degree
δe	Control deflection of elevator in degree
δa	Control deflection of aileron in degree
δr	Rudder control deflection in degree
ϕ,θ,ψ	Roll angle, Pitch angle, and Yaw angle respectively in degrees
$\dot{\beta}$	Sideslip rate in deg/sec
α_0	Initial AoA in degree
τ_w	Time constant of washout filter in second

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1. INTRODUCTION

The development of automatic control systems progressed primarily based on challenges encountered in controlling modern aircraft with power-driven aerodynamic control surfaces. A lot of interest has been focused on the study of interconnected multivariable control systems[1] over the past many years. Multi-Input Multi-Output system of directional and lateral axes of the aircraft[2,3] is coupled which it became very interesting to the scientific community. Classical multivariable interconnected control systems are studied and developed[4-7] for F-16, passenger jet aircraft, etc., Nevertheless, the design procedure is not easily available.

The major design objective of lateral and directional axis control law is to give expectable, fast, and sharp roll response while minimising the angle of sideslip. A feedback control system with p, r, and β to rudder and aileron may be designed for fulfilling requirements stability (static as well as dynamic). Due to the effect of cross-axis coupling, lateral control stick input causes the buildup of large sideslip and also shows degraded roll response because of the presence of the Dutch roll mode. The various methods of obtaining good turn coordination are described in Ref. 6. Most of the flights tend to deviate especially in the modes of wing rock, wing drop, etc. at high AoA because of multiple combinations of aerodynamic deficiencies.

The (ARI) Aileron to Rudder Interconnect gain is primarily designed to reduce sideslip by decoupling the Dutch roll (rolling and yawing oscillations) mode from the 'p' response to aileron command[8]. It is designed to generate the required yaw rate component to attain stability-axis roll and ensuring the stability of the aircraft in the lateral and directional axis. The oscillatory response (mainly at high AoA) and decrease of adverse yaw because of deflection of aileron and difficulties in control of sideslip are removed by an interconnect gain[9]. Specific flights testhas been conducted by NASA to assess the effects of interconnect gain between aileron and rudder during landing phases[10] of the F-14A airplane. The flight test results with ARI gain exhibited that the turn rate of the airplane was more responsive and improved handling qualities by lateral-control inputs. The pilot mentioned that control systems with ARI have shown better results in correcting heading deviations and offset adjustments during the final approach compared to standard control system without ARI[10]. Aircraft F16-A/B[11] manual brings out the fact that during wing damage or horizontal tail damage, the activity of roll stick is needed to get aircraft to the wings level. It is also stated that the task is manageable and challenging. In this type of control surface damages, aileron to rudder interconnect reconfiguration will improve aircraft handling qualities particularly, during landing phase tasks.

Even though feed forward control does not have an impact on aircraft stability, it helps in adjusting the manipulated variable to reduce the deviations from desired or target value by measuring disturbance. It is well evident that the feedback components in a control system aid in reducing the uncertainty of the control system while the feed forward components make the control system more susceptible to uncertainty of the plant. The lateral and directional control law with feedback paths of p, r, and β take care of uncertainty to a large extent. The stringent MIL requirement on gain margin[6] decibels and phase margin of 35° amounts to 50% plant variation. Note that loop gain can vary by up to 50% due to the 6dB gain margin. Thus, flight control system is designed robust enough to tackle some fault scenarios without the need of reconfiguration of flight control law. The feed forward ARI gain is used to decouple the lateral and directional axis. ARI gain design is an iterative procedure[5] as per present industry standard practice. Generally, the design of ARI gain is done by using the trim algorithm[5] and modifications using nonlinear simulation and these methods cannot be used to handle fault scenarios in real-time. Hence, non-iterative design of ARI is essential.

An organised non-iterative ARI gain design procedure is described in this paper which is useful for online reconfiguration of ARI gain and it is illustrated on the fault scenario of aileron control surface damage. This technique requires estimation of transfer function coefficients in real-time, but it is not the focus of the current work. A typical single-engine tailless fighter aircraft is considered in this paper which has control surfaces of four elevons (aileron and elevator) and one rudder.

2. CONVENTIONAL DESIGN METHODOLOGY OF ARI GAIN

The right-handed orthogonal coordinate reference system of the body axis with its origin (O) at the aircraft center of gravity is considered[5]. The positive X, Y, and Z axes are directed towards the nose, right-wing (starboard), and bottom of the aircraft. The XOZ plane coincides with the aircraft's plane of symmetry. The equations of motion[5,6,7] for a rigid body is the basis for controller design and simulation. These are ordinary nonlinear differential equations with twelve states as [α, β, V_T, p, q, r, ϕ, θ, ψ, X_e, Y_e, H]. The first six states are defined the body axis coordinate system and the last six states are about the earth axis NED coordinate system. Out of twelve states, the first eight describe complete aircraft dynamics for stability analysis. The other four states [ψ, X_e, Y_e, H] do not affect the stability and fast response. These later parameters vary slowly and are used primarily for guidance and navigation. Therefore, these parameters are not required for stability analysis and transient response shaping. Therefore, for primary control law design for the inner loop is carried out without considering these four states. The linear model with eight aircraft states is further divided into a four-state model. One 4th order set characterises longitudinal dynamics and the other set characterises lateral and directional dynamics. This is a standard practice used by practicing flight control designers for control law design using simplified linear models. This helps control law design to be done based on the understanding of the flight dynamics of the plant. The designed controller is finally tested using nonlinear six degrees of freedom simulation software. The standard sign convention for control surface deflection is used[5] and the same table is reproduced in Table 1.

The lateral and directional dynamics aircraft model is linearised by technique of small perturbation. The obtained linear model is appropriate for the design of controller with β, p, r, ϕ as state vectors and δa, δr as input vectors and ϕ, β, p, r, Ny as observation vectors respectively. By neglecting the ϕ dependence and ϕ equation, the aircraft lateral and directional model in a state-space domain is provided in Eqn (1). However, spiral mode is not significant in the view of the handling point

Table 1. Control surface sign convention of aircraft

Control surface	Sign	Deflection and its effects
Elevator	Positive	Trailing edge goes down which creates a negative moment in pitch
Rudder	Positive	Trailing edge goes left which creates a negative moment in yaw
Ailerons	Positive	Right-wing trailing edge goes down which creates a negative moment in roll

because of its large time constant. Hence, it can be excepted from aircraft model of state-space. So, a simplified third-order model of aircraft is studied with state vector defined as $x = [p, r, \beta]$ and $u = [\delta a, \delta r]$. The state space model is $\dot{x} = Ax + Bu$, and A, B can be considered from Eqn (1)

$$\begin{bmatrix}\dot{p}\\ \dot{r}\\ \dot{\beta}\end{bmatrix} = \begin{bmatrix} L_p & L_r & L_\beta \\ N_p & N_r & N_\beta \\ (Y_p / V_T) + sin\alpha_0 & (Y_r / V_T) - cos\alpha_0 & Y_\beta \end{bmatrix} \begin{bmatrix} p \\ r \\ \beta \end{bmatrix} + \begin{bmatrix} L_{\delta a} & L_{\delta r} \\ N_{\delta a} & N_{\delta r} \\ Y_{\delta a} & Y_{\delta r} \end{bmatrix} \begin{bmatrix} \delta a \\ \delta r \end{bmatrix} \quad (1)$$

The block schematic of lateral and directional law[12,13] for fighter aircraft is shown in the Fig. 1. This control law augments directional stability, improves Dutch roll damping, and helps in better flying qualities like the rejection of disturbance, fast response of roll rate and improves turn coordination. The aircraft needs to roll about the velocity vector to remove the kinematic coupling among α and β at high alpha regions to lateral stick input. The angular rates in the stability axis $p_s = p$ and $r_s = r - ptan\alpha$ are used instead of angular rates of body axis for the feedback. The term of inertial coupling compensation $\left(\Delta\dot{r} = \left(\frac{I_y - I_x}{I_z}\right) pr\right)$ is added in the feedback path of directional axis. The design of lateral and directional controller is done to meet the MIL requirement specifications[14-15].

The sideslip and roll rate transfer functions are derived from Eqn (1), and given as $G_{p\delta a}(s) = \frac{p(s)}{\delta a}, G_{p\delta r}(s) = \frac{p(s)}{\delta r}, G_{\beta\delta a}(s) = \frac{\beta(s)}{\delta a}, G_{\beta\delta r}(s) = \frac{\beta(s)}{\delta r}$ as given in Eqns (2), (2a) and (2b).

$$\begin{bmatrix} p(s) \\ \beta(s) \end{bmatrix} = \begin{bmatrix} G_{p\delta a}(s) \\ G_{\beta\delta a}(s) \end{bmatrix} \delta a(s) + \begin{bmatrix} G_{p\delta r}(s) \\ G_{\beta\delta r}(s) \end{bmatrix} \delta r(s) \quad (2)$$

$$p(s) = G_{p\delta a}(s)\delta a(s) + G_{p\delta r}(s)\delta r(s) \quad (2a)$$

$$\beta(s) = G_{\beta\delta a}(s)\delta a(s) + G_{\beta\delta r}(s)\delta r(s) \quad (2b)$$

After the design of primary feedback loops, interconnect gain from aileron to rudder (K_{ARI}) has been synthesised to decouple the Dutch roll from the p output to δa input. The implementation of (K_{ARI}) gain is considered as shown in Eqn (3)

$$\delta r = K_{ARI}\delta a \quad (3)$$

With the ARI gain as given in Eqn (3), (2a) is reshaped and given in Eqn (4)

$$p(s) = (G_{p\delta a}(s) + K_{ARI} G_{p\delta r}(s))\delta a(s) \quad (4)$$

Therefore, K_{ARI} increase has resulted in movement $G_{p\delta a} + K_{ARI} G_{p\delta r}$ of zeros towards the $(G_{p\delta r})$ zeros. The gain needs to be carefully chosen for introduction of zero near poles of Dutch roll after augmentation for cancelling poles and zeros approximately.

Example 1:

The state-space model in linear domain of the typical aircraft at FC of 1.0 Km altitude and 0.4 Mach No. (Hence forward 0.4 M and 1.0 Km) has been considered for detailed study.

Figure 2 shows the location of $G_{p\delta a}$ zeroes with various K_{ARI} and the poles of Dutch roll after augmentation. The angle of sideslip (beta) response is plotted in Fig. 2 for various K_{ARI} gains with lateral stick doublet input given at 1 second. Itis observed from Fig.2 that distance between pole and zero is minimum with an approximate K_{ARI} values ranging from 2.2 to 2.4. The optimum K_{ARI} =1.7gain is selected for the particular FC as it generates the minimum sideslip. The example highlights the problems of conventional design process. It is noticed that small variation in K_{ARI} leads to variation in initial sideslip response which generates either more proverse or more adverse yaw. This degrades handling quality (HQ) about adverse yaw. The degradation of adverse yaw results in overshoots of sideslip, heading for a given lateral control input as demonstrated in[16]. The optimum ARI gain =1.7 is obtained for a particular FC where it generates minimum adverse yaw which improves HQ. Note that, even though K_{ARI} =2.4 does better cancellation Dutch roll poles (Fig. 2) and leading to decoupled from roll rate, the sideslip response shows adverse yaw

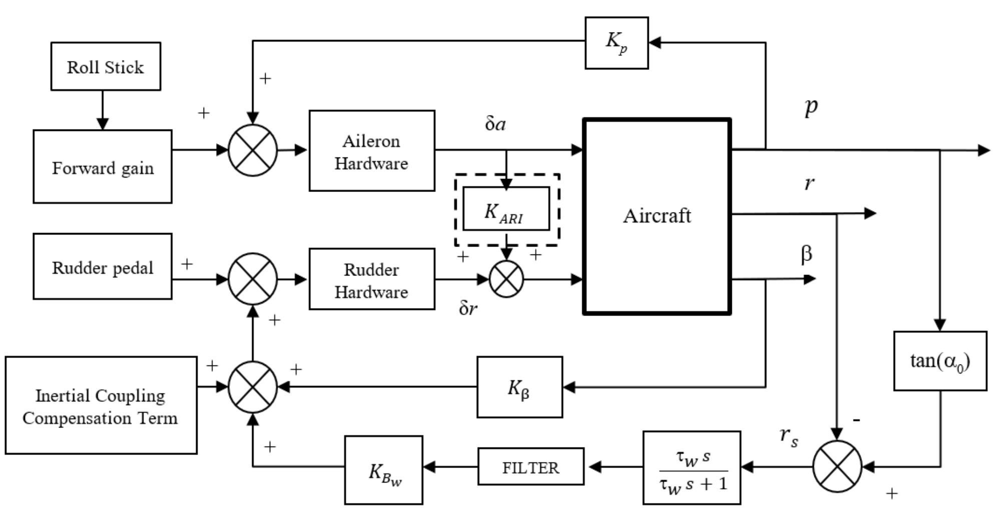

Figure 1. Block schematic of control law in Lateral and Directional axis

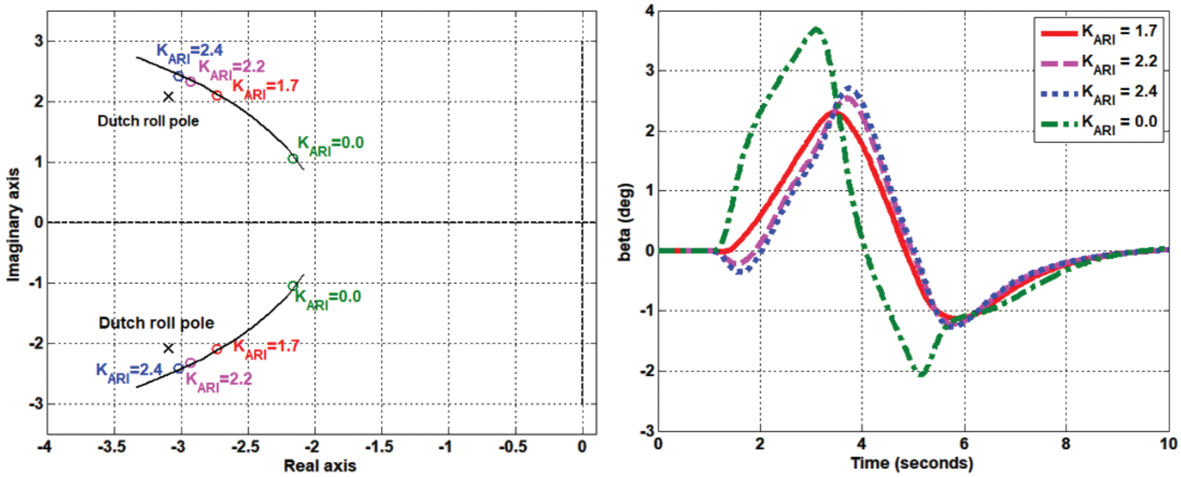

Figure 2. Plot to demonstrate K_{ARI} sensitivity near location of Dutch roll pole and zero of G_{pa} and β response plot with lateral stick input at 0.4 M and 1 km for various ARI gains.

(Fig. 2) and therefore not acceptable from the HQ perspective. The sensitivity of ARI gain to side slip response reveals that a reliable and optimum method is required for the design.

The plant parameters do vary with dynamic pressure (a function of the square of speed i.e, V_T^2 and air density). Therefore, it is necessary to schedule gains with dynamic pressure. As the aircraft speed increases, the compressibility effects also start becoming dominant on the aircraft model. Hence, ARI gain is designed at various operating flight conditions and then scheduled with Mach No. and altitude. As FC changes in the nonlinear simulation, these gains are interpolated smoothly between the various designed operating points. Moreover, the design of ARI gain is primarily to improve the transient response at a particular FC. In this paper, these ARI gains are designed using a linear aircraft model and then scheduled in nonlinear simulation for checking the smooth transition as aircraft speed and altitude varies.

Selection of ARI gain can't be automated as it is bottomed on the root placement location near to the Dutch roll pole and then checking time response simulation in closed-loop or handling qualities assessment in the environment of real-time simulation. Currently, K_{ARI} design methods are observing the root locus or trim algorithm and then nonlinear simulation time response which is an iterative process. The Hankel norm[17] is used for this purpose. For reconfiguration of control law in real-time, a non-iterative method of K_{ARI} gain design is necessary.

3. DESIGN METHODOLOGY OF ARI GAIN

3.1 Method 1 for ARI Gain Design

K_{ARI} is found to decrease both contaminations of Dutch roll mode in p response as well as side slip response to aileron input which is given in Eqn (5).

$$K_{ARI} = \min_{\substack{\omega\in[\omega_{min},\omega_{max}] \\ K_A=-8\ to\ 8}} \left(avg_{\omega} \left| \frac{G_{\beta\delta a}(j\omega)}{G_{p\delta a}(j\omega)} \right| \right) \qquad (5)$$

where $\frac{G_{\beta a}(s)}{G_{pa}(s)} = \frac{\frac{\beta(s)}{\delta a(s)}}{\frac{p(s)}{\delta a(s)}} = \frac{\beta(s)}{p(s)}$

However, both the transfer functions $G_{\beta\delta a}(s)$, $G_{p\delta a}(s)$ simultaneously can be used either before or after closing the feedback loops. Aileron to rudder interconnect gain design is independent of stability augmentation feedback loops[13]. The detailed steps to calculate ARI gain are described below.

- Find the ratio of the magnitude of beta to roll rate transfer function for the interested frequency range (i.e., 0.1 Hz to 0.5 Hz) for different ARI gain values (-8 to 8). The frequency range between 0.08 Hz to 1.5 Hz and 0.08 Hz to 5 Hz is considered to know the sensitivity of the frequency range on the optimum selection of ARI gain. Based on a sensitivity study at different parts of the flight envelope and it is concluded that 0.1 to 0.5 Hz frequency range is adequate to calculate the optimum ARI gain.
- Find the average value for the above frequency range for various ARI gains
- Select the optimum K_{ARI} , where the average of magnitude ratio is minimum

A typical plot of beta to roll rate transfer function magnitude with different ARI gains is shown in Fig. 3 for the range of frequencies (0.1 Hz to 1.5 Hz) along with average magnitude (marked as a square symbol) for 0.26 M and 1.3 Km (at the same FC described in Fig. 2). The K_{ARI} value is selected as the ARI gain of minimum average magnitude as shown in Fig. 3. Hence, this method can be used effectively for the calculation of ARI gain non-iteratively off-line.

3.2 Method 2 for ARI Gain Design

An alternate method for K_{ARI} is proposed in the current section. In this method, the basic requirement is to cancel/nullify sideslip which is caused due to aileron input by operating suitable rudder input through ARI. Thus, taking sideslip to be zero in Eqn (2b) which is system transfer function in open-loop domain,

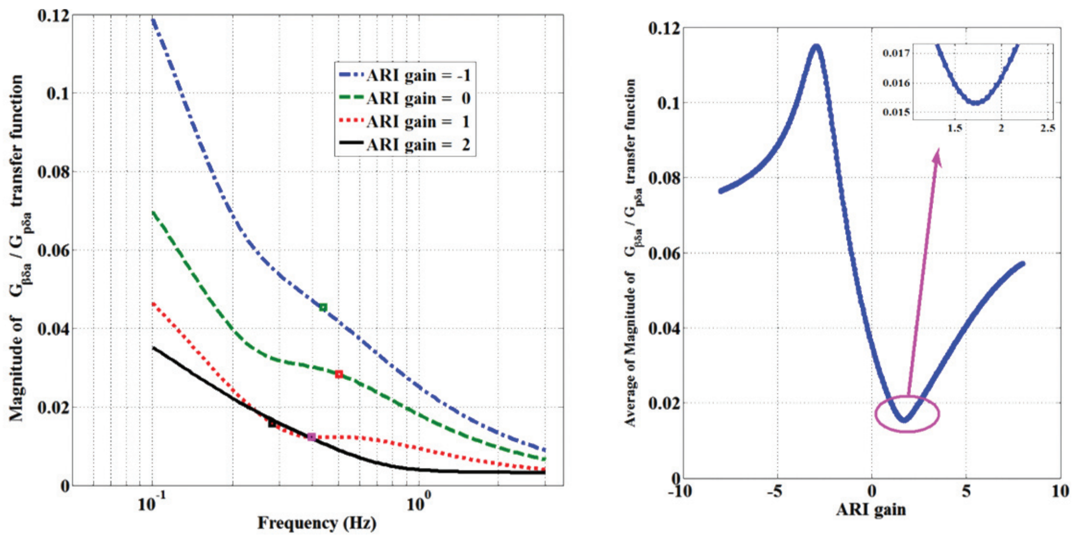

Figure 3. Bode Magnitude plot $\frac{G_{\beta a}(s)}{G_{pa}(s)}$ and its average magnitude marked symbol as 'square' for one typical FC of 0.26 M and 1.3 Km.

$$\beta(s) = G_{\beta\delta a}(s)\delta a(s) + G_{\beta\delta r}(s)\delta r(s) = 0$$

The rudder δr can be derived as in Eqn (6)

$$\delta r = -\frac{G_{\beta\delta a}(s)}{G_{\beta\delta r}(s)}\delta a = -G_{ARI}(s)\delta a \qquad (6)$$

ARI transfer function in Eqn (6) $G_{ARI}(s)$. The above given derivation of $G_{ARI}(s)$ holds good in the closed-loop domain also.

$G_{ARI}(s)$ is derived analytically from the lateral and directional model of third order with states as *p*, β, B_w as given in Eqn (7). In this method, instead of the static K_{ARI} gain, ARI can be directly considered as the transfer function shown in Eqn (7).

$$G_{ARI}(s) = -\frac{\frac{\beta(s)}{\delta a(s)}}{\frac{\beta(s)}{\delta r(s)}} = -\frac{G_{\beta\delta a}(s)}{G_{\beta\delta r}(s)} = \frac{A1s^2 + A2s + A3}{B1s^2 + B2s + B3} \qquad (7)$$

where $A1 = Y_{\delta a}$, $B1 = Y_{\delta r}$,

$$A2 = -\left(Y_{\delta a}L_p + Y_{\delta a}N_r - L_{\delta a}Y_p - N_{\delta a}Y_r\right),$$

$$B2 = -\left(Y_{\delta r}L_p + Y_{\delta r}N_r - L_{\delta r}Y_p - N_{\delta r}Y_r\right),$$

$$A3 = Y_{\delta a}\left(L_pN_r - L_rN_p\right) - L_{\delta a}\left(Y_pN_r - Y_rN_p\right) - N_{\delta a}\left(L_pY_r - L_rY_p\right),$$

$$B3 = Y_{\delta r}\left(L_pN_r - L_rN_p\right) - L_{\delta r}\left(Y_pN_r - Y_rN_p\right) - N_{\delta r}\left(L_pY_r - L_rY_p\right)$$

For the real-time implementation of the K_{ARI} design when a fault occurs, it requires a online estimation of the lateral and directional parameters as listed in Eqn (7). All elements or parameters required for $G_{ARI}(s)$ calculation can be calculated from the parameters estimated in real-time. The aero database parameters may be carried online for reliable and accurate parameter estimation.

3.3 Method 3 for ARI Gain Design

ARI gain (K_{ARI}) is found from the $G_{ARI}(s)$ (Method-2) frequency response value as s→∞. To approximate one static gain K_{ARI} instead of $G_{ARI}(s)$, K_{ARI} is computed from Eqn (8) based on norm evaluated by varying ARI gain values from -8 to 8. Considered $\omega_{min}, \omega_{max}$ are 0.1 Hz and 5 Hz (rigid body frequencies) respectively. This method is also effective and non-iterative used in the calculation of ARI gain.

$$K_{ARI} = \min_{\substack{\omega \in [\omega_{min}, \omega_{max}] \\ k = -8 \text{ to } 8}} \left(\| G_{ARI}(j\omega) - k \|_2 \right) \qquad (8)$$

3.4 Method 4 for ARI Gain Design

Traditionally the ARI has been considered as scheduled gain, which is a function of flight-condition, instead of a transfer function. It is observed from Fig. 2 that the prime consequence of the K_{ARI} gain is noticeable in initial transient response when lateral stick input is given. Thus, initial momentary response is dominated by roll mode and which is governed by roll damping (L_p) frequency. Aero database table L_p can be carried on board or can be estimated through an extended Kalman filter. The sign of K_{ARI} is selected from the real part of $G_{ARI}(s)$ as given in Eqn (9).

$$K_{ARI} = -sign\left(real\left(G_{ARI}(jL_p)\right)\right) * \left|G_{ARI}(jL_p)\right| \qquad (9)$$

In the real-time implementation, this reconfiguration method of K_{ARI} is computationally efficient and non-iterative. This Method 4 is more coherent to Method -2 instead of carrying the transfer function. The approximation of one static gain value of $\frac{G_{\beta\delta a}(s)}{G_{\beta\delta r}(s)}$ from Method-4 can be applied for control law reconfiguration when the fault has occurred.

3.5 Nonlinear Evaluation with Various ARI Gain Design Methods

Nonlinear 6 degrees of freedom simulation is carried out with all twelve states[5]. The simulation software is validated using flight data. The simulation is carried out using the standard numerical integration scheme ODE23 which is the Bogacki-Shampine method of explicit Runge-Kutta (2,3) pair. The step size used is 80 Hz (0.0125 seconds) and flight simulations are done for mass of ~9000 Kg, the moment of inertia (Iyz=32.5,Iyy=66558.9,Ixx=14087.9,Izz=77875.2,Izx=2490.5, Ixy = 162.5) and CG of (34.0% MAC). For aircraft model considered in this paper, the position limit of rudder and aileron control surface deflection is ± 30°, and ± 18° respectively and corresponding their rate limits are ±100 °/sec and ± 45 °/sec. Maximum control deflection occurs at low speed and it decreases as the speed increases.

In this section, the ARI gain obtained from above mentioned four methods are tabulated in Table 2 with three wings level (1g trim) flight conditions. It can be observed that ARI gain values attained by the conventional method, Method 1, Method 3, Method 4 are in good accordance with each other.

The response from an off-line simulation in the nonlinear domain with an input of roll stick doublet for the three flight conditions mentioned in Table 2 with ARI gain values obtained from proposed methods is plotted in Fig. 4. It can be realised that the time responses of p, r, and β are similar to G_{ARI} from Method 2 and with K_{ARI} from Method 1, Method 3, Method 4. The comparison of time response variations of other parameters like aileron (δa), aileron rate, rudder (δr), rudder surface rate, Mach No., Altitude with the roll stick doublet input at 1 sec. for all four proposed methods for one FC of 0.7 M, 6 km is given in Fig. 5.

From simulation responses obtained for considered aircraft at different FC, it can be observed that K_{ARI} gain attained from Method 4 gives the least side slip. Hence, it is recommended from the analysis that ARI gain can be computed using Method 4 for online control law reconfiguration whenever fault occurs..

4. K_{ARI} RECONFIGURATION FOR ONE AIRCRAFT CONTROL EFFECTOR EFFECTIVENESS DECREASE

Control surface impairment can happen in flight due to several reasons like bird strikes or combat scenarios like battle damage. The damage can be a complete failure or partial failure (physical). These types of damages are sometimes difficult to detect by software and hardware redundancy management. The fatalities can lead to even loss of aircraft at times, especially during full stick (pitch/roll) maneuvers. In this regard, damage to the control surface is demonstrated as a decrease of the control surface. The flight validated simulation software of combat aircraft is used for the modelling of the considered damage scenario of the 'loe' control effector.

To demonstrate the damage case of '90% decrease of the

Table 2. Comparison with various design methods of K_{ARI}

FC (Mach (M) and altitude (Km))	Method-1 (K_{ARI} from Eqn (5))	Method-2 ($G_{ARI}(s)$ *instead of* K_{ARI})	Method-3 K_{ARI} from Eqn (8)	Method-4 K_{ARI} from Eqn (9)
0.26 M & 6 km	3.47	$\frac{3.0658*(-0.047422s^2+6.0933\,s+1)}{0.089507s^2+5.3797\,s+1}$	3.412	3.483
0.7 M & 6 km	-0.6	$\frac{-0.7164*(0.0095s^2+0.4727\,s+1)}{0.00482s^2+0.59304\,s+1}$	-0.61	-0.604
1.3 M & 6 km	-0.319	$\frac{0.1492*(0.00453s^2+0.81534s+1)}{0.00142s^2+0.30861s+1}$	-0.313	-0.303

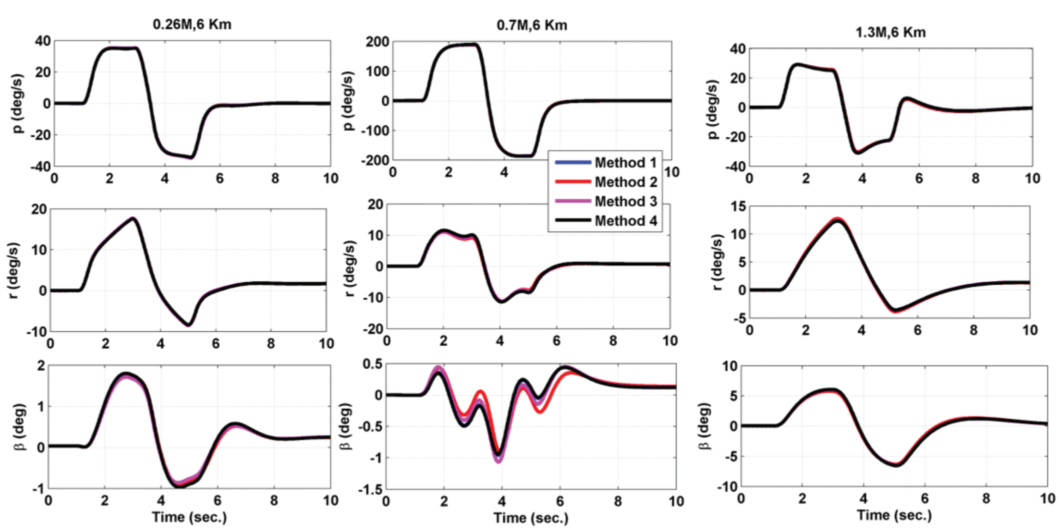

Figure 4. Nonlinear evaluation with various ARI gain design methods for flight conditions specified in Table 2 for input of lateral stick doublet at 1 sec.

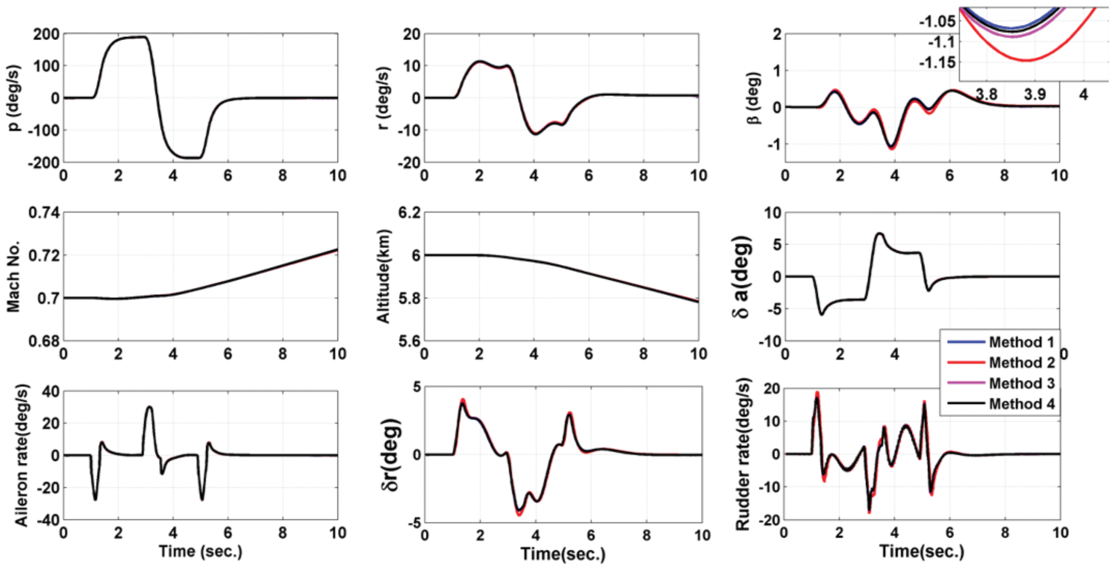

Figure 5. Simulation results plot at 0.7 M and 6 km with input of lateral stick doublet at 1 sec for all four proposed methods in non-linear domain.

port side 'loe' surface effectiveness', the value of 'loe' control surface in aero database tables (force and moment coefficients which are a function of elevon control surfaces) is decreased to 10%. To illustrate as an example, the Rolling moment coefficient7 which is modelled for 'loe' surface damage is shown as in Eqn (10).

$$C_l = C_{l\beta}(\beta) + C_{l\delta r}(\alpha, Mach, \delta r) + C_{lr}(Mach)\frac{rb}{2v} + C_{lp}(Mach)\frac{pb}{2v} + \\ C_{l\delta a}(\alpha, Mach, loe*0.1) + C_{l\delta a}(\alpha, Mach, roe) + \\ C_{l\delta a}(\alpha, Mach, rie) + C_{l\delta a}(\alpha, Mach, lie) \tag{10}$$

where C_{lp}, C_{lr}, $C_{l\beta}$, $C_{l\delta r}$, $C_{l\delta a}$ are rolling moment coefficient aero database tables for *p, r*, and β, rudder, aileron which are a function of α, β, lie, loe, rie, roe, δr, Mach number and C_l is total rolling moment coefficient.

Six degrees of freedom non-linear simulation results at FC (9.6 Km altitude, 0.5 Mach No.) fora given input of roll stick doublet at 2 seconds is presented. Figure 6 shows a comparison of responses with the considered three cases given below to validate the reconfiguration with ARI gain for fault scenario of 'loe' control surface damage.

(a) No-Fault condition

Responses of *p, r,* and β, Roll acceleration, δ*a*, aileron rate, position of 'rie' and 'loe', α*,* δ*r*, rudder rate are adequate as observed from Fig. 6 with roll stick input.

(b) Initiation of fault at 1 second and no control reconfiguration

The fault is introduced at 1 second in this case. It is observed from the responses in Fig. 6 that AoA increases to about 30° and builds up of about 7° sideslip. Control surfaces are also observed that maximum saturation limits are reached. Responses like *p*, *r*, and roll acceleration shown abrupt changes. From the simulation results, it was observed that if the fault damage is not a, it could cause saturations of control surfaces which may lead to aircraft departure and stability loss.

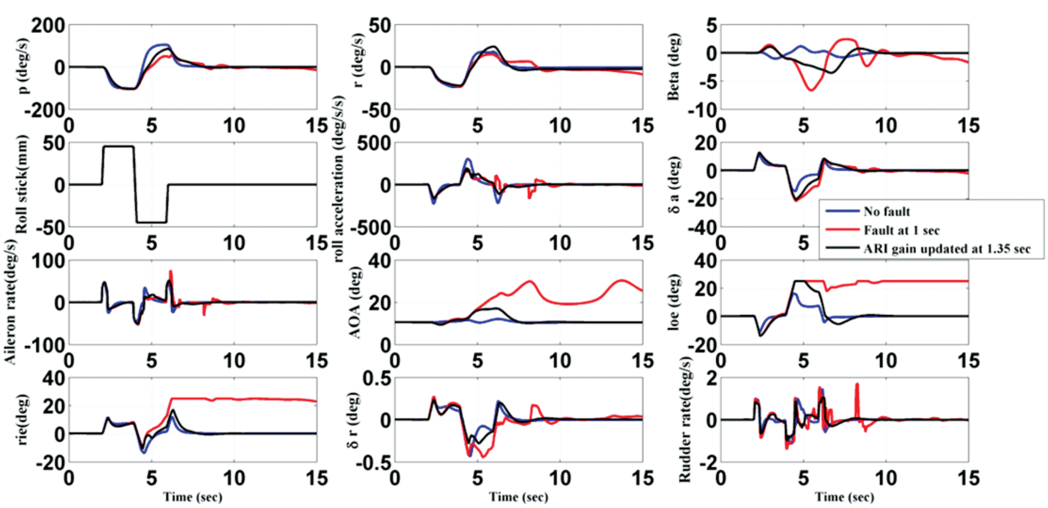

Figure 6. Reconfiguration of K_{ARI} using Method 4 for fault scenario (90% reduction of 'loe' surface effectiveness) at 0.5 M and 9.6 km altitude.

(c) Fault at one second, Reconfiguration of K_{ARI} at 1.35 sec.

For flight control law reconfiguration, fault identification is required. The fault identification process for the reduction of one control surface effectiveness is presented in[18]. With control surface damage, control derivatives $L_{\delta a}$, $N_{\delta a}$, $Y_{\delta a}$ change and have been estimated using Kalman filter (EKF) algorithm[18]. The calculation of ARI gain uses these parameters from onboard aero database tables and estimated parameters after the fault occurrence. 0.35 sec delay is accounted for online estimation of lateral and directional axis parameters and computation of K_{ARI} for the considered fault case as reported in[18]. Hence, in this example, ARI gain is reconfigured using Method 4 and is updated at 1.35 seconds. The results with updated ARI gain are presented in Fig. 6. From the plots, it can be seen that after the gain reconfiguration of K_{ARI} , sideslip is reduced to less than 4° from 7°, AoA is within limits, and control surface saturation is avoided resulting in the recovery of stable flight.

5. CONCLUSIONS

The interconnect gain from aileron to rudder plays a significant function of Dutch roll mode decoupling from the response of roll rate while guaranteeing velocity vector roll for demanded lateral stick input. However, it is the correction in the feedforward system that is prone to uncertainty in the plant and hence it requires fast modification/adaptation when the fault has occurred. Four innovative methods are proposed for the designs of Aileron-rudder interconnect gain in this paper and a detailed discussion is carried out. It is demonstrated that computations of design of interconnect gain from aileron to rudder by G_{ARI} magnitude at L_P (i.e., Method 4) is more efficient especially for online control law reconfiguration in event of damage of control surface or actuator fault. Corresponding simulation results are presented. The sideslip response is minimal for the roll stick inputs with optimum aileron to rudder interconnect gain designed with the proposed methods.

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