Defence Science Journal, Vol. 74, No. 6, November 2024, pp. 795-803, DOI : 10.14429/dsj.74.19488

A Baseline Nonlinear Flight Controller Design Approach For A Fighter Aircraft With A Shorter Design Cycle Time

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ABSTRACT

In this paper, a framework for designing a baseline full envelope flight controller for the stability augmentation of an unstable fighter aircraft is presented. The flight dynamics assessments for these aircraft are normally carried out using both off-line and real-time simulations. The framework, referred to here as Simplified Nonlinear Dynamic Inversion (SNDI), results in significant reduction in the overall design cycle time. The SNDI based controller produces simultaneous control allocation and decoupling in the time domain. The Aileron to Rudder Interconnect (ARI) gain which is commonly used for the lateral-directional control decoupling is generalised here for an aircraft with multiple redundant controls. The off-line simulations and simulator based pilot evaluations carried out for the assessment of aircraft dynamics enable an early comparison between different candidate configurations to narrow down to the final choice of the design configuration in a short time. The proposed approach is illustrated using the model of Aero Data Model In Research Environment (ADMIRE) delta canard fighter aircraft. Off-line simulation results show that the proposed controller design achieves a similar performance when compared to that of other existing controllers available in the open literature which are designed based on a conventional approach. The proposed design is carried out in a shorter span of design cycle time and meets all the design specifications considered.

Keywords: Nonlinear flight control; Control allocation; Simplified nonlinear dynamic inversion; Fighter aircraft

NOMENCLATURE

ADMIRE	: Aero data model in research environment
AoA, AOA, alpha, α	: Angle of attack (deg)
AoSS, AOSS, beta, β	: Angle of sideslip (deg)
A	: Augmented plant matrix
B	: Control matrix
DLIE	: Left inboard elevon deflection (deg)
DLOE	: Left outboard elevon deflection (deg)
DLC	: Left canard deflection (deg)
DRC	: Right canard deflection (deg)
DROE	: Right outboard elevon deflection (deg)
DRIE	: Right inboard elevon deflection (deg)
DRUD	: Rudder deflection (deg)
EAFC	: Existing ADMIRE flight controller
g	: Acceleration due to gravity
h	: Altitude (m)
I_X	: Roll moment of inertia (Kg-m^2)
I_Y	: Pitch moment of inertia(Kg-m^2)
I_Z	: Yaw moment of inertia(Kg-m^2)
NDI	: Nonlinear dynamic inversion
N_z	: Load factor (normal acceleration) (g units)
N_y	: Lateral acceleration
p	: Body axis roll rate (deg/s)
p_s	: Stability axis roll rate (deg/s)
q	: Body axis pitch rate (deg/s)
r	: Body axis yaw rate (deg/s)
r_s	: Stability axis yaw rate (deg/s)
S	: Control allocation matrix
SNDI	: Simplified nonlinear dynamic inversion
TSS	: Time scale separation
T_s	: Body axis to stability axis transformation matrix
u	: Control vector [DRC DLC DROE DRIE DLIE DLOE DRUD]^T
$\overline{u}$	: Pseudo control matrix $[\delta_{pitch}\ \delta_{roll}\ \delta_{yaw}]^T$
V_T	: True air speed (m/s)
V	: Velocity of aircraft (m/s)
x	: State vector of rotational states $[q\ p\ r]^T$
$\overline{x}$	: Transformed x vector $[q\ p_s\ r_s]^T$
y	: Vector of aerodynamic states $[\alpha\ q\ \beta\ p\ r]^T$

Received : 19 September 2023, Revised : 31 May 2024
Accepted : 10 June 2024, Online published : 25 November 2024

$\overline{y}$: Transformed vector of aerodynamic states $[\alpha\ q\ \beta\ p_s\ r_s]^T$
ϕ : Roll aAngle (deg)
μ : Velocity vector roll angle (deg)
γ : Flight path angle (deg)

1. INTRODUCTION

Designing flight controllers for a high performance fighter aircraft operating at higher angles of attack is a highly challenging task since they are operating in highly nonlinear regimes of flight within the flight envelope. Before describing the proposed approach, a brief review of existing flight controller design methods is presented next.

Different multi-variable nonlinear flight control design techniques, like Nonlinear Dynamic Inversion (NDI) via Time Scale Separation (TSS), Block backstepping and Sliding-Mode Control, that offer shorter design cycles are presented in Thunberg & Robinson[1]. The ability to arrive at a baseline flight controller with lesser amount of design efforts enables one to make a better choice for a control configured vehicle.

For an aircraft, the total control demand generated from a flight controller can be distributed among the available control effectors using control allocation schemes that include optimization, pseudo-inverse and control ganging approaches[2-3]. Performing control allocation separately benefits in accounting for both actuator position and rate limits for dissimilar actuators. Handling of multiple control surfaces in the dynamic inversion framework[3]. The control allocation methods, such as Ganged Pseudo-Inverse, Weighted Pseudo-Inverse, cascaded Generalized-Inverse and Daisy Chain etc., with an application to ADMIRE aircraft are discussed in[4].

The highly nonlinear flight regimes are dominated by inherent nonlinearities such as gravity correction, kinematic and inertia coupling terms which cannot be ignored during the flight controller design. To minimize the effects of kinematic coupling, the stability axis angular rates (p_s and r_s) given in Eqn. (1) are recommended for feedback in place of the body axis angular rates (p and r).

$$\begin{bmatrix} p_s \\ r_s \end{bmatrix} = \begin{bmatrix} \cos\alpha & \sin\alpha \\ -\sin\alpha & \cos\alpha \end{bmatrix} \begin{bmatrix} p \\ r \end{bmatrix} = T_s \begin{bmatrix} p \\ r \end{bmatrix} \quad (1)$$

where, T_s is the body to stability axis transformation matrix. The inclusion of gravity compensation, kinematic and inertia coupling terms to flight control laws of Eurofighter 2000[5].

A typical flight envelope of a fighter aircraft covers speeds ranging from Mach 0.2 to 2.0 resulting in significant nonlinear variations of its aerodynamic characteristics. The control law synthesis cycle for a typical fighter aircraft is shown in Fig. 1. The three important steps of this synthesis cycle in the order in which they are carried out are: scrutiny of aerodynamic data, controller design and analysis and off-line and real-time nonlinear simulations for the verification and validation of the designed controller gains.

The above steps of the control law synthesis shown in Fig. 1 will be iterative and time consuming involving the controller design and then its evaluation in a six degrees of freedom nonlinear simulation environment with both off-line and real-time simulations. This controller design process normally takes around three to four months. The above discussed synthesis cycle is currently in vogue with respect to the controller design schemes for fighter aircraft observed in the literature.

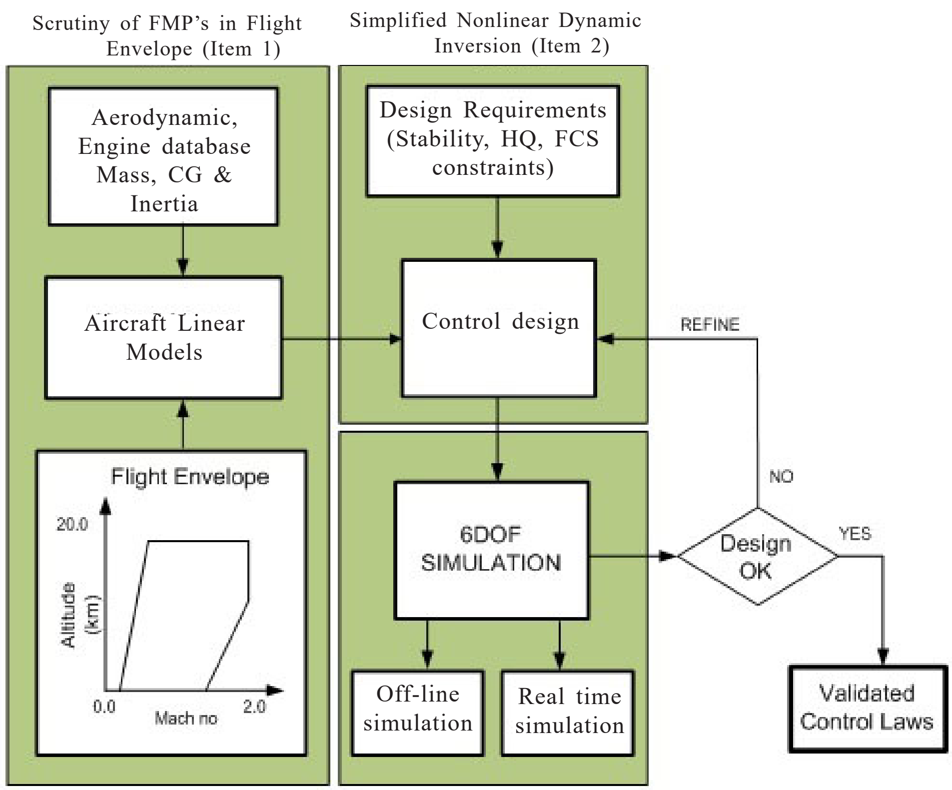


Figure 1. Schematic of a full envelope control law synthesis cycle.

The flight controller design approach proposed in this paper successfully reduces the above mentioned time and efforts involved in the iterative process described in Fig. 1. The proposed controller design can be carried out in a month and is achieved in two steps with respect to Fig. 1. First, the scrutiny of the flight dynamics parameters is carried out for the entire flight envelope (Item 1 of Fig. 1). Next, the SNDI based controller design is undertaken to include the important nonlinearities (Item 2 of Fig. 1) into the design phase itself. This design approach enables an early comparison between different configurations in a flight simulation environment to narrow down the final choice in a short time.

The proposed SNDI based flight controller is based on a cascaded structure and comprises of two outer loops and three inner loops[6]. The two outer loops are: AoA and AoSS and the three inner loops are: pitch rate, stability axis roll and yaw rates. It is to be noted that the proposed controller design is non-iterative and can achieve simultaneous control allocation and decoupling for a fixed wing fighter aircraft in a shorter design cycle time. This is the main contribution of this paper.

This paper aims at exploring the SNDI framework towards simultaneous control allocation and decoupling in a shorter design cycle time with ADMIRE[7-8] as the candidate aircraft.

Performance and robustness of the proposed controller are demonstrated by considering its performance for command following, robustness to parameter variations and actuator failure cases. Comparison of ADMIRE nonlinear simulation responses, generated using SNDI based controller and Existing ADMIRE Flight Controller (EAFC) taken from the open literature[7-8], has clearly shown that the proposed controller achieves similar performance and in some cases better control decoupling. The EAFC based simulation responses show excursions of N_y, AoSS, AoA and N_z higher compared to those generated using SNDI based controller in some cases. It is also observed that the SNDI based controller is more robust to parameter variation and failure cases.

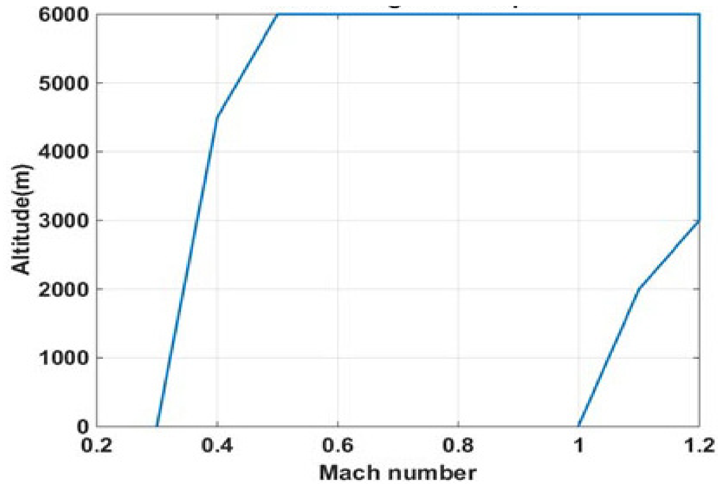
The paper is organised as follows. Section 2 presents the ADMIRE aircraft details and describes the proposed controller design method using SNDI framework. In Section 3, performance of SNDI based controller for ADMIRE is presented first, followed by the details of EAFC. Finally, a comparison of nonlinear simulation responses, obtained using SNDI based controller and EAFC, is presented. Section 4 summarises the main conclusions from this study.

2. SNDI BASED CONTROLLER DESIGN FOR ADMIRE AIRCRAFT

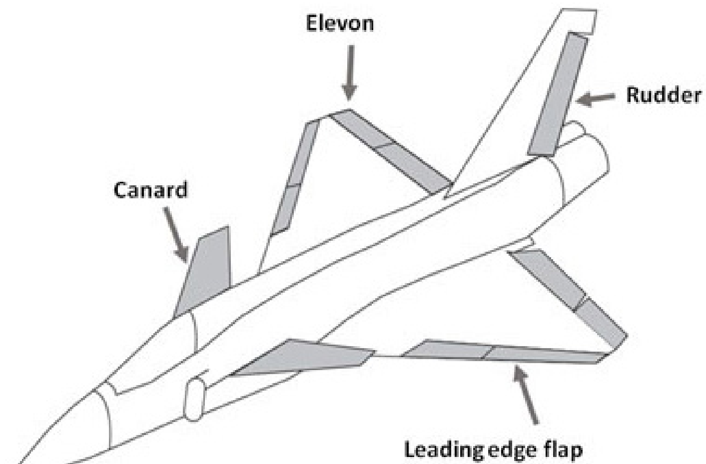
In this section, ADMIRE aircraft details are presented first followed by a description of SNDI based flight controller design.

2.1 ADMIRE Aircraft

ADMIRE is a small single seater fighter aircraft with delta canard configuration. The flight envelope considered is shown in Fig. 2(a). The control surfaces include two canards, four elevons and a rudder[7-8]. The arrangement of control surfaces is shown in Fig. 2(b). Details of mass, CG and moments of inertia are given in Table 1.



(a)



(b)

Figure 2. (a) Approximate ADMIRE flight envelope[7-8]; and (b) Arrangement of ADMIRE control surfaces[7-8].

TABLE 1. ADMIRE configuration data[7-8]

Description	Value & Unit
Wing area	45.0 m^2
Wing span (b)	5.2 m
Mean aerodynamic chord (cbar)	10.0 m
Mass	9100 kg
I_X	21000.0 kg-m^2
I_Y	81000.0 kg-m^2
I_Z	101000.0 kg-m^2
I_{XZ}	2100.0 kg-m^2
XCG	0.0 m
YCG	0.0 m
ZCG	-0.15 m

The actuator and sensor models are taken[7-8]. First order actuator model ($\frac{1}{0.05s+1}$) is used for the canards, elevons and rudder. The rate limiter for each elevon actuator is set to 150° per sec.; for each canard actuator it is 50° per sec. and for rudder actuator it is 100° per sec..

Sensor Models:

Air data sensors (V_T, α, β, h): $\frac{1}{1+0.02s}$

Inertial sensors (p,q,r,N_z):
$\frac{1+0.005346s+0.0001903s^2}{1+0.03082s+0.0004942s^2}$

Attitude Sensors θ ϕ : $\frac{1}{1+0.0323s+0.00104s^2}$

Due to space limitations, the full description of ADMIRE model along with the Matlab/ Simulink implementation is given in the Supplementary Material.

The SNDI framework uses NDI with Time Scale Separation (TSS), backstepping and control allocation. The NDI with TSS exploits basic TSS between aircraft velocity components and angular rates in body axes. Control allocation ensures appropriate useof multiple control effectors for the control ofpitch, roll and yaw channels. The backstepping procedure creates a feedback signal into the control structure. The kinematic and inertia coupling terms are included. The gravity correction termsincluded in the pitch and yaw channels minimize the drop in AoA and building up of AoSS due to multiple rolls.

Handling qualities for the piloted aircraft dictate the bandwidth of inner loop. Typically, it is in the range of 2-10 rad/s throughout the flight envelope[9]. Inner loop gains are obtained assuming a rigid aircraft. These gains arechosen on the higher side to meet HQ requirementsand achieve robustness against actuator failures and for rejecting external disturbances.Outer loop gains are chosen so as to keep a good separation from the inner loop. This approach is adequate for trade-off studies between competing aircraft configurations in the initial design phase.

2.2 SNDI Based Controller Design

A systematic basis needs to be laid out considering the operational constraints and maximum performance achievable within these operational constraints for flight controller design.

3. METHODOLOGY

- Scrutiny of aerodynamic data to obtain maximum achievable performance within the operational constraints.
- Generate the linearised aircraft mathematical models in state space form by trimming the aircraft nonlinear model at the straight and level flight condition. Obtain the control allocation matrix 'S' based on the study of 'B' matrix such that $(BS)^{-1}T_1$ is diagonal
- Compute the inner loop and outer loop feedback gains
- Carryout the simulation studies with aircraft nonlinear model.

The operational constraints and performance are obtained based on the scrutiny of aerodynamic data. Fig. 3 shows the operational constraints of the ADMIRE with respect to AoA and N_z. More details of SNDI based controller design[6]. The preliminary work carried out with delta canard aircraft[6]. This paper includes the additional work carried as well.

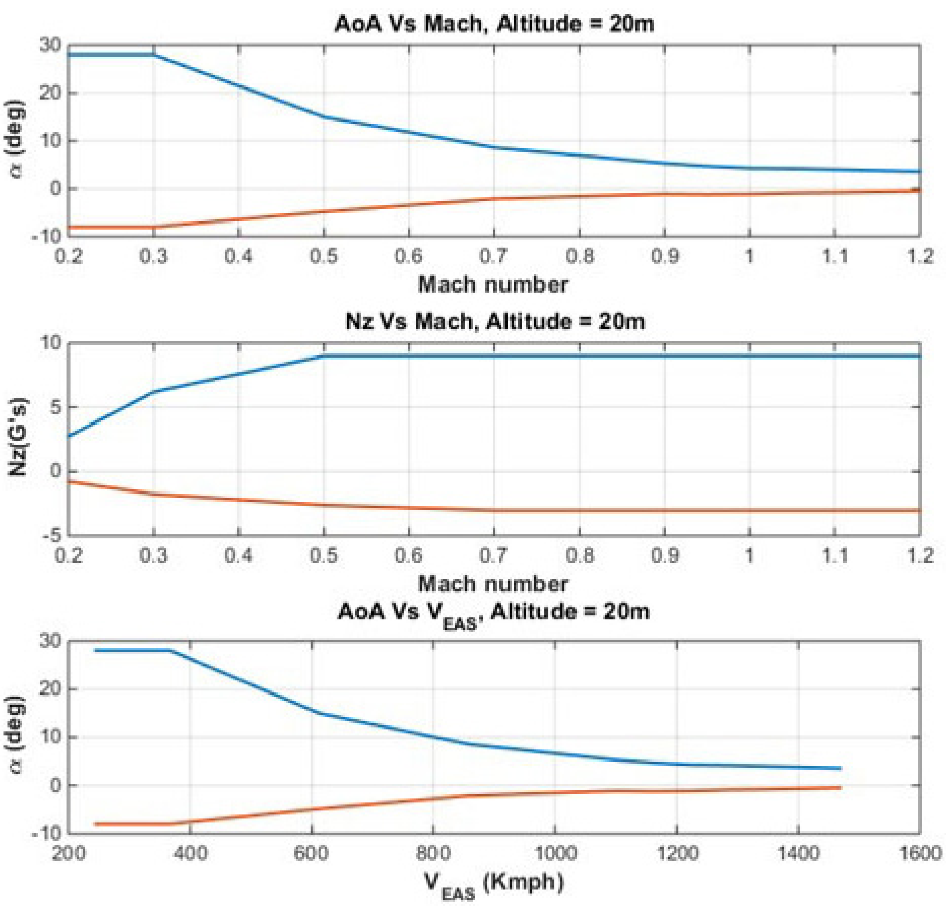


Figure 3. ADMIRE operational constraints.

Development of flight controller in SNDI framework with simultaneous control allocation and decoupling is explained next.

The simplified aircraft rotational dynamics equations can be written as:

$$T_1\dot{\bar{x}} = \begin{bmatrix} \left(\dfrac{I_z - I_x}{I_y}\right)pr \\ \left(\dfrac{I_y - I_z}{I_x}\right)qr \\ \left(\dfrac{I_x - I_y}{I_z}\right)pq \end{bmatrix} + AT_2\bar{y} + BS\bar{u} \tag{2}$$

where, the dimensions of T_1 and T_2 matrices are 3×3 and 5×5, respectively. The 'S' matrix is for control allocation and depends on the control matrix 'B' of the dimension 3×7.

Eqn. (2) can be re-arranged as:

$$(BS)^{-1}T_1\dot{\bar{x}} = (BS)^{-1}\begin{bmatrix} \left(\dfrac{I_z - I_x}{I_y}\right)pr \\ \left(\dfrac{I_y - I_z}{I_x}\right)qr \\ \left(\dfrac{I_x - I_y}{I_z}\right)pq \end{bmatrix} + (BS)^{-1}AT_2\bar{y} + \bar{u} \tag{3}$$

During the flight controller design, following aspects are considered to meet the HQ criteria. These criteria are arrived at based on the general industry practice and MIL standards[9] for fighter aircraft design and development. The design of stick path is performed accordingly.

- The value of $\dot{\alpha}$ should be limited to 25 deg/sec through out the envelope
- At each of the flight condition, maximum achievable roll rate and roll acceleration should not exceed 200 deg/sec and 600 deg/sec^2, respectively for full roll stick input
- During roll maneuvers, the AoSS build-up should be kept at minimum possible
- The SNDI framework results in a first order time response with the damping ratio typically close to unity. Hence, the HQ requirements, which typically demand damping ratio ≥0.5, are met.

3.1 Simultaneous Control Allocation and Decoupling Scheme

At high angles of attack and rotational rates, dynamic coupling cannot be ignored. The lateral-directional decoupling is ensured by properly determining the ARI gain[10].

In the present paper, we discuss employing a control allocation scheme with control decoupling in general and lateral-directional control decoupling especially using ARI gain. As mentioned earlier, the ADMIRE aircraft has seven control surfaces using which the pitch, roll and yaw channels are controlled. The control matrix, 'B' is studied to determine the structure of the control allocation matrix, 'S'.

A pre-determined scheme for ganging multiple control effectors to produce a single effective control law for each channel is considered[3] by reducing the control dimension from seven to three $[\delta_{pitch}\ \delta_{roll}\ \delta_{yaw}]^T$. Hence, 'S' matrix has the dimension of 7×3. The 'S' matrix is scheduled as a joint function of airspeed and AoA. The ARI gain, an element of 'S' matrix, is computed through optimisation.

The following approach is considered to determine 'S' matrix including ARI gain:

- The symmetric elevons (KP_3) and the symmetric canards (KP_1) are used for pitchcontrol. The differential elevons (KR_4) and thedifferential canards (KR_2) are used for roll control. The differential elevons (KY_4), the differential canards (KY_2) and the rudder (KY_5) are used for yaw control.

The S matrix is given by:

$$S = \begin{bmatrix} KP_1 & -KR_2 & -KY_2 \\ KP_1 & KR_2 & KY_2 \\ KP_3 & -KR_4 & -KY_4 \\ KP_3 & -KR_4 & -KY_4 \\ KP_3 & KR_4 & KY_4 \\ KP_3 & KR_4 & KY_4 \\ KP_5 & KR_5 & KY_5 \end{bmatrix} \tag{4}$$

The gain KR_5 (ARI gain) and maximum of KY_2, KY_4 or KY_5 is chosen for optimization resulting in a diagonal $(BS)^{-1}T_1$ matrix and thereby achieving the control decoupling of longitudinal, lateral and directional channels.

If the resulting optimized matrix $(BS)^{-1}T_1$ is not diagonal, then there would be coupling between the terms. This may not result in control decoupling of pitch, roll and yaw channels. A study was carried out to examine the $(BS)^{-1}T_1$ matrix, which is not diagonal, corresponding to altitude 15000 ft. and Mach number 1.0 at level flight trim condition and presented as follows.

$$(BS)^{-1}T_1 = \begin{bmatrix} -0.0157 & 0.0 & 0,0 \\ 0.0 & 0.0149 & -0.0535 \\ 0.0 & -0.0054 & 0.0638 \end{bmatrix}$$

The nonlinear simulation response match for this case is presented in Fig. S8 of Supplemental Information and the match is found to be satisfactory with good control decoupling.

- The maximum stability axis roll rate (velocity vector roll rate) computed from 'S' matrix is used for forward path command scaling throughout the aircraft speed range.

The above procedure to determine 'S' matrix including ARI gain is a key contribution of this paper. It allows handling ofany number of redundant control effectors in a practical manner while achieving lateral-directional decoupling. This method is also amenable to automation, which helps to expedite the design process.

The inner loop feedback gains corresponding to pitch (K_q), roll (K_{pS}) and yaw (K_{rS}) channels are selected with an aim that the time-scale separation between bandwidths of actuator and inner loop ensures no rate limiting during high gain maneuvers. The inner loop feedback gains are inversely proportional to respective dimensional control derivatives and scheduled as function of dynamic pressure. The stability axis roll rate and pitch rate loops are considered to be command

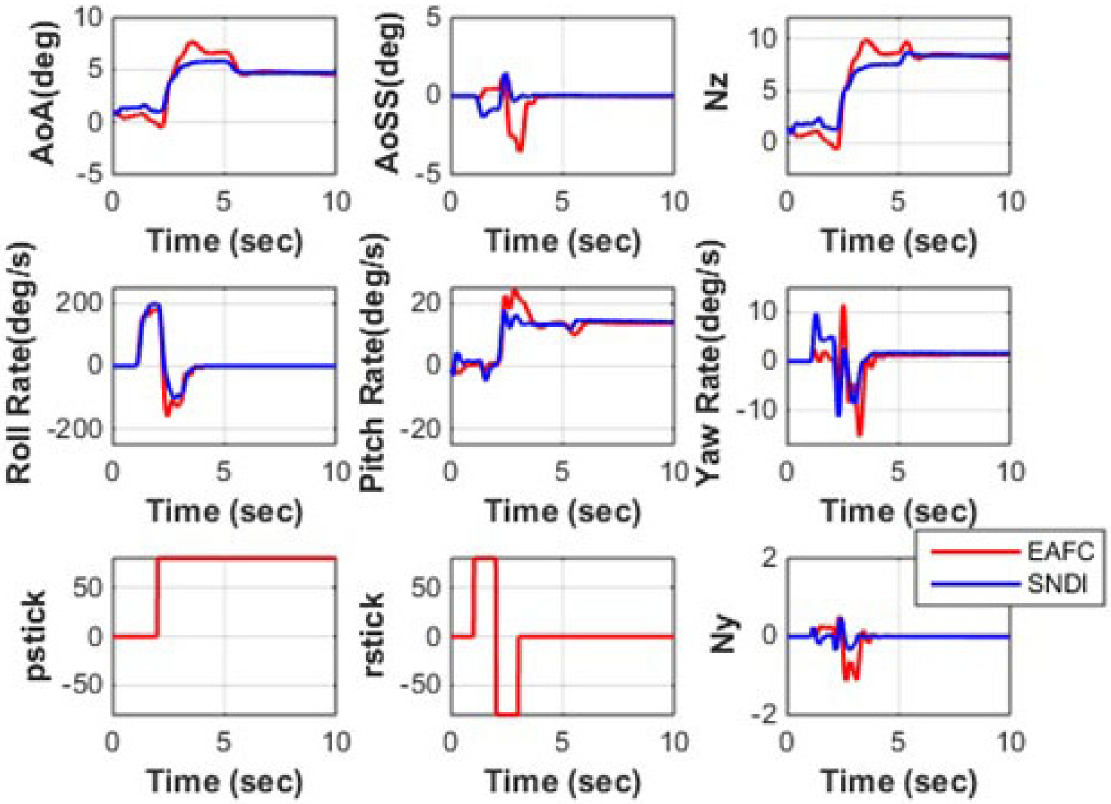


Figure 4. Nonlinear closed loop simulation responses for full pitch and roll stick inputs with discrete gust injected at 5sec for altitude 15000 ft. and Mach number 1.0.

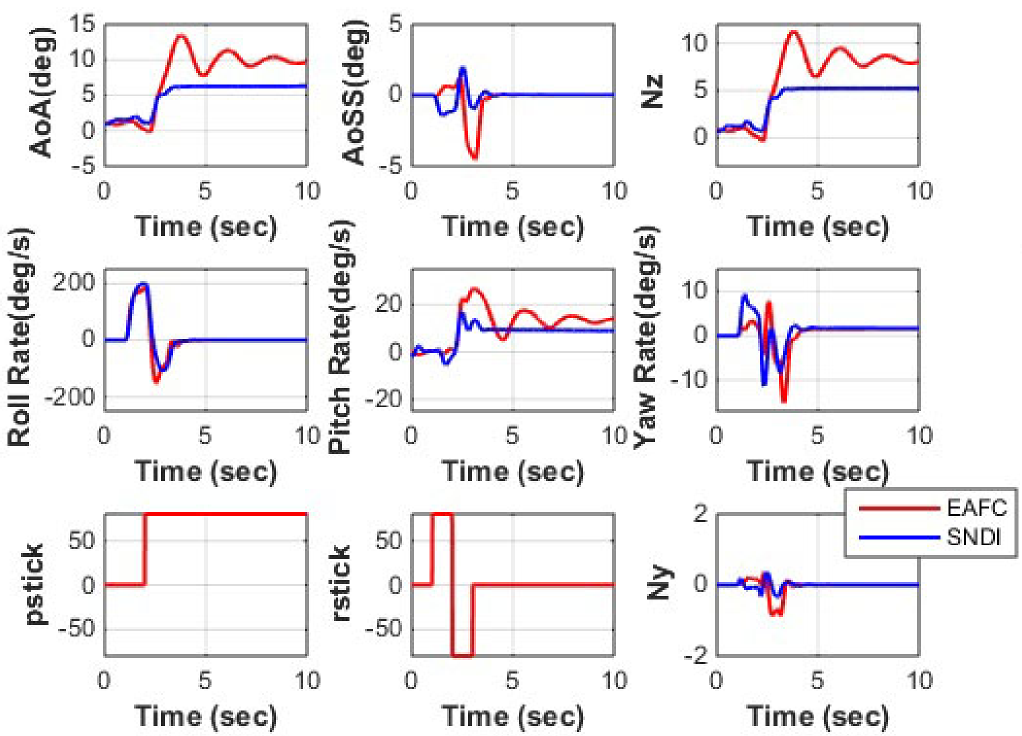
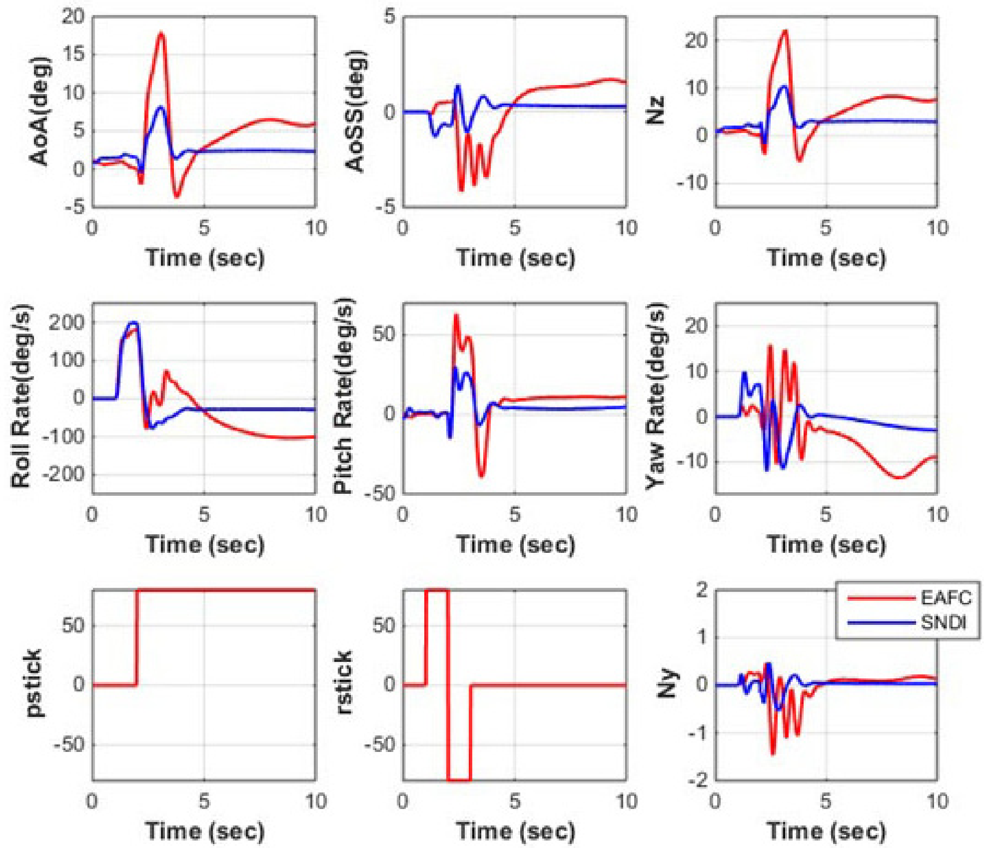


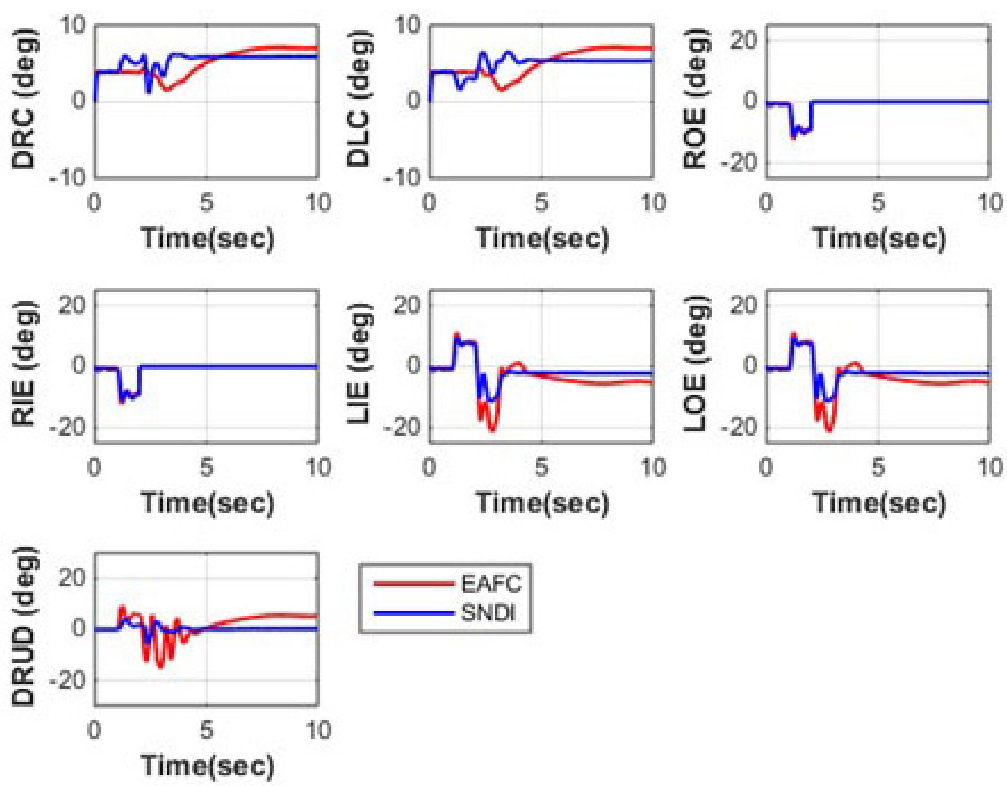
Figure 5. Nonlinear closed loop simulation responses for coupled full pitch and roll stick inputs with mass and inertia variation (nominal+50 %) for altitude 15000 ft. and mach number 1.0.

channels, while the stability axis yaw rate loop is considered to be disturbance rejection channel.

The outer loop Eqn. (α,β) defined in Reference 3 are simplified with the assumption of small (β) i.e., the $(p_s \, \mathrm{Tan}\beta)$ term can be dropped. For the ease of carrying out simulations with stability axis roll and yaw rates, (μ) angle is replaced by (ϕ) angle. Due to space limitations, the comparison of results with (μ) and (ϕ) angle is given in the supplementary material.



(a)



(b)

Figure 6. (a) Nonlinear closed loop simulation responses for coupled full pitch and roll stick inputs with right inboard and outboard elevon actuator failure at 2 sec. for altitude 15000 ft. and mach number 1.0.; and (b) Control surface responses for coupled full pitch and roll stick inputs with right inboard and outboard elevon actuator failure at 2 sec for altitude 15000 ft. and mach number 1.0

Hence, the outer loop Eqn. arere-written as:

$$\dot{\alpha} \cong q - \frac{g}{V}\left(N_Z - \cos\phi \cos\gamma\right) \tag{5}$$

$$\dot{\beta} \cong -r_S + \frac{g}{V}\left(\sin\phi \cos\gamma - N_y\right) \tag{6}$$

$$q_{cmd} \cong K_\alpha\left(\alpha_{cmd} - \alpha\right) + \frac{g}{V}\left(N_Z - \cos\phi \cos\gamma\right) \tag{7}$$

$$r_{Scmd} \cong -K_\beta\left(\beta_{cmd} - \beta\right) + \frac{g}{V}\left(\sin\phi \cos\gamma - N_y\right) \tag{8}$$

The AoA outer loop gain is K_α= 2.5 rad/s/rad and the AoSS outer loop gain is K_β= -2.5 rad/s/rad. The values chosen for K_α and K_β provide desired AoA and AoSS responses. It may be noted that sufficient TSS is ensured between inner to outer loops.

The gravity correction terms are simplified in terms of measured signals thus avoiding the need to carry onboard the entire aerodynamic database. The gravity correction terms mentioned in Eqns. (7, 8) are included in the longitudinal and directional channels of flight controller. The inertia coupling terms mentioned in Eqns. (2, 3) are included in all the three channels of flight controlleras they should not be ignored especially when large roll rate maneuvers are performed.

4. PERFORMANCE COMPARISON OF THE SNDI BASED CONTROLLER AND EAFC

In this section, the controller performance evaluation for SNDI based controller for ADMIRE is presented first. The performance comparison of SNDI based controller with the EAFC is discussed next with supporting nonlinear closed loop simulation results.The EAFC taken from open literature[7-8] follows conventional controller design process and uses a Control Selector (CS) to distribute the three control channels (roll, pitch and yaw) to the seven control actuators. The CS for pitch channel is obtained using a method proposed in[3].

4.1 Performance of SNDI Based Controller

The SNDI based controller, explained in previous section, is developed in Matlab/Simulink and integrated to ADMIRE flight dynamic model taken from the open literature[7-8]. As stated already, the SNDI flight controller results in AoA, roll rate command following and the ‘S’ matrix ensures appropriate usage of canards, elevons and rudder for control. This controller is capable of meeting the HQ requirements discussed in Section 2. Highest possible inner loop gains ensure robustness to mass, C.G. and aerodynamic uncertainties without exciting the position and rate limits of the actuators. The performance comparison of both SNDI based ADMIRE controller and EAFC is discussed next.

Due to space limitation, the results corresponding to subsonic and supersonic speeds, especially the low speed and high altitude case, are presented in Supplementary Material.

The comparison of ADMIRE nonlinear closed loop simulation responses obtained using EAFC and SNDI based controller is shown in Figs. 4 to 6 for three cases: vertical gust input with coupled pitch and roll input; parameter (mass and inertia) variation with coupled pitch and roll input; and right elevon actuators fail with coupled pitch and roll input. These results highlight the performance and robustness of proposed controller with respect to atmospheric disturbances, parameter variation and failure scenario for coupled pitch and roll stick input cases.

4.2 Command Following

The nonlinear simulation is carried out with coupled full pitch and roll stick inputs at 15000 ft. altitude and 1.0 Mach number. The discrete vertical gust input is injected at 5 sec. The simulation studies are carried out with SNDI based controller and EAFC. The comparison of simulation responses is shown in Fig. 4. It is observed that the roll rate has not exceeded 200 deg/sec while ensuring control decoupling with very small variations in N_y and AoSS for roll stick input.

4.2 Parameter Variation

A study has been carried out to see the effect of mass and inertia variations. The nonlinear simulation is carried out with coupled full pitch and roll stick inputs at 15000 ft. altitude and Mach number1.0 along with the mass and inertia variation (+50 % with respect to nominal values). The simulation studies are carried out with SNDI based controller and EAFC. The comparison of simulation responses is shown in Fig. 5. The AoSS, pitch rate, roll rate, yaw rate, N_z and N_y excursions for SNDI based controller are found to be lower compared to EAFC.

4.3 Actuator Failure

A study has been carried out to demonstrate the efficacy of control allocation scheme when right inboard and outboard elevon actuators are failed. The nonlinear simulation is carried out with coupled full pitch and roll stick inputs at altitude 15000 ft. and 1.0 Mach number. The right elevon actuators are failed at 2 sec. The simulation studies are carried out with SNDI

based controller and EAFC. The comparison of simulation responses is shown in Fig. 6(a) and Fig. 6(b). The AoSS, pitch rate, roll rate, yaw rate, N_z and N_y excursions for SNDI based controller are found to be stabilized effectively compared to those of EAFC.

It is observed that the simulation responses obtained with SNDI based controller are more robust, provide similar performance and better decoupling compared to those of obtained with EAFC for the three cases considered. The proposed SNDI controller is achieved in one month whereas EAFC which is based on conventional control design takes typically three to four months

5. CONCLUSIONS

The controller design for ADMIRE aircraft using the SNDI framework proposed in this paper can be summarised in the following steps:

- Initially, the open loop stability, control and performance parameters for the aircraft are studied. The optimal mix of the control power to be used to achieve the desired performance is also indicated as part of the study
- The approach to arrive at the control allocation matrix ‘S’, which includes ARI gain for lateral-directional control decoupling, is demonstrated with appropriate control power distribution
- For the SNDI based controller, closed loop gains are

chosen keeping in view the HQ requirements as per MIL specifications, limits on control surface positions, actuator bandwidths and actuator rates for large amplitude maneuvers
- Comparison of simulation responses of SNDI based controller with EAFC demonstrated similar performance and better decoupling with the SNDI based controller.

An orderly controller design process prevents many iterative back and forth designs while meeting the performance goals. The proposed controller is capable of addressing the full flight envelope in one sweep and enables the designers to conduct piloted evaluations of candidate configurations during the preliminary design phase itself. The SNDI controller design approach has also been successfully applied to three different aircraft, viz., a cranked arrow delta wing aircraft with separate elevators, a tailless double delta wing aircraft with elevons and rudder and atrapezoidal wing aircraft with multiple controls resulting in a significant reduction in design cycle times.

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    doi: 10.14429/dsj.71.15347

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