COMMAND TRACKING IN HIGH PERFORMANCE AIRCRAFTS: A NEW DYNAMIC INVERSION DESIGN

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Abstract: This paper proposes a new straight forward technique based on dynamic inversion, which is applied for tracking the pilot commands in high performance aircrafts. Pilot commands assumed in longitudinal mode are normal acceleration and total velocity (while roll angle and lateral acceleration are maintained at zero). In lateral mode, roll rate and total velocity are used as pilot commands (while climb rate and lateral acceleration are maintained at zero). Ensuring zero lateral acceleration leads to a better turn co-ordination. A six degree-of-freedom model of F-16 aircraft is used for both control design as well as simulation studies. Promising results are obtained which are found to be superior as compared to an existing approach (which is also based on dynamic inversion). The new approach has two potential benefits, namely reduced oscillatory response and reduced control magnitude. Another advantage of this approach is that it leads to a significant reduction of tuning parameters in the control design process.

Keywords: Command tracking, dynamic inversion, aircraft control, longitudinal maneuver, lateral maneuver

1. INTRODUCTION

Designing flight control systems for aircrafts (especially high performance aircrafts) is a challenging task and such control design procedures are still evolving. Dynamic inversion, Lyapunov design, Sliding mode design, Model Predictive control etc. are some of the nonlinear control design methods which appeared in literatures recently. However, because of its simplicity and elegance the Dynamic Inversion approach (Menon, 1993) has found relatively wide acceptance. In this method, which is essentially based on the technique of feedback linearization, an appropriate coordinate transformation is carried out to make the system look linear so that any known linear controller design method can be used. The major concern of Dynamic Inversion approach is the mismatch between the model used and the actual plant. Because of this, ideas like augmenting the Dynamic Inversion technique with H_∞ design (Banda, 1996), Neuro-adaptive design (Calise, 1997) etc. have been proposed recently.

Based on dynamic inversion, a new method is proposed in this paper to design the flight control system. This new method has features similar to an existing approach (Menon, 1993), where the goal is to design a controller such that the roll rate, normal acceleration and lateral acceleration commands from the pilot are tracked. One of the main advantages of new method, however, is that there is no requirement on transforming the normal and lateral acceleration commands to the pitch and yaw rate commands. An additional goal of tracking total velocity command is also considered in the new method. Note that because of their time-scale separation, the aerodynamic and thrust controls are designed separately. The aerodynamic controls which are used for tracking of roll rate, normal acceleration and lateral acceleration commands are updated at a fast rate, whereas the thrust control is updated at a slower rate, which is used to track the velocity command.

In order to demonstrate the usefulness of the proposed technique, it is used in a nonlinear Six-Degree-of-Freedom (Six-DOF) model (Roskam, 1995) of a fighter aircraft F-16 (Nguyen, 1979). The comparative simulation results are presented which shows that the proposed new method requires lower control magnitude and has better transient response (lesser overshooting and no non-minimum phase behavior), thus making it a more efficient approach (Menon, 1993).

2. AIRCRAFT DYNAMICS

Assuming the airplane to be a rigid body, the complete set of Six-Degree-of-Freedom (Six-DOF) equations of motion over a flat earth in the body frame of reference (Roskam, 1995) are:

$$\dot{U} = RV - QT - g\sin\Theta + (F_{A_x} + T)/m \quad (1)$$
$$\dot{V} = PW - RU + g\cos\Theta\sin\Phi + F_{A_y}/m \quad (2)$$
$$\dot{W} = QU - PV + g\cos\Theta\cos\Phi + F_{A_z}/m \quad (3)$$
$$\dot{P} = c_1QR + c_2PQ + c_3L_A + c_4N_A \quad (4)$$
$$\dot{Q} = c_5PR + c_6(R^2 - P^2) + c_7M_A \quad (5)$$
$$\dot{R} = c_8PQ - c_2QR + c_4L_A + c_9M_A \quad (6)$$
$$\dot{\Phi} = P + Q\sin\Phi\tan\Theta + R\cos\Phi\tan\Theta \quad (7)$$
$$\dot{\Theta} = Q\cos\Phi - R\sin\Phi \quad (8)$$
$$\dot{\Psi} = (Q\sin\Phi + R\cos\Phi)\sec\Theta \quad (9)$$
$$\dot{h} = Usin\Theta - Vsin\Phi cos\Theta - Wcos\Phi cos\Theta \quad (10)$$

$$\begin{bmatrix} c_1 \\ c_2 \\ c_3 \\ c_4 \\ c_8 \\ c_9 \end{bmatrix} \triangleq \frac{1}{(I_XI_Z - I_{XZ}^2)} \begin{bmatrix} I_Z(I_Y - I_Z - I_{XZ}^2) \\ I_{XZ}(I_Z + I_X - I_Y) \\ I_Z \\ I_{XZ} \\ I_{XZ}^2 + I_X(I_X - I_Y) \\ I_X \end{bmatrix}$$

$$\begin{bmatrix} c_5 & c_6 & c_7 \end{bmatrix} \triangleq \frac{1}{I_Y} \begin{bmatrix} (I_Z - I_X) & I_{XZ} & 1 \end{bmatrix}$$

In the above equations U, V, W are the velocity components along the body-fixed axes. P, Q, R are the roll, pitch and yaw rates respectively about the body-fixed axes and Φ, Θ, Ψ are the Euler angles and h is the height. $F_{A_x}, F_{A_y}, F_{A_z}$ are the aerodynamic components of the external forces and T is the thrust along the longitudinal axis. Similarly, L_A, M_A, N_A are the aerodynamic components of the airplane. I_X, I_Y, I_Z, I_{XZ} represents the moment of inertias of the airplane in the body frame. m and g represents mass and acceleration due to gravity respectively (both assumed as constants in this paper). The aerodynamic forces and moments along X, Y, Z directions are given by

$$\begin{bmatrix} F_{A_X} & F_{A_Y} & F_{A_Z} \end{bmatrix}^T = \bar{q}S \begin{bmatrix} C_{X_t} & C_{Y_t} & C_{Z_t} \end{bmatrix}^T \quad (11)$$

$$\begin{bmatrix} L_A & M_A & N_A \end{bmatrix}^T = \bar{q}S \begin{bmatrix} bC_{L_t} & \bar{c}C_{M_t} & bC_{N_t} \end{bmatrix}^T \quad (12)$$

where $\bar{q}$ is the dynamic pressure and the non-dimensional aerodynamic force $(C_{X_t}, C_{Y_t}, C_{Z_t})$ and the moment $(C_{L_t}, C_{M_t}, C_{N_t})$ coefficients are expressed as multivariate nonlinear functions and are adapted from (Nguyen, 1979)

$$C_{X_t} = C_x(\alpha, \delta_e) + C_{x_q}(\alpha)\tilde{q} \quad (13)$$
$$C_{Y_t} = C_y(\beta, \delta_a, \delta_r) + C_{y_p}(\alpha)\tilde{p} + C_{y_r}(\alpha)\tilde{r} \quad (14)$$
$$C_{Z_t} = C_z(\alpha, \beta, \delta_e) + C_{x_q}(\alpha)\tilde{q} \quad (15)$$
$$C_{L_t} = C_l(\alpha, \beta) + C_{l_p}(\alpha)\tilde{p} + C_{l_r}(\alpha)\tilde{r} + C_{l_{\delta_a}}(\alpha, \beta)\delta_a + C_{l_{\delta_r}}(\alpha, \beta)\delta_r \quad (16)$$
$$C_{M_t} = C_m(\alpha, \delta_e) + C_{m_q}(\alpha)\tilde{q} + C_{Z_t}(x_{cg_{ref}} - x_{cg}) \quad (17)$$
$$C_{N_t} = C_n(\alpha, \beta) + C_{n_p}(\alpha)\tilde{p} + C_{n_r}(\alpha)\tilde{r} + C_{n_{\delta_a}}(\alpha, \beta)\delta_a + C_{n_{\delta_r}}(\alpha, \beta)\delta_r - C_{Y_t}(x_{cg_{ref}} - x_{cg})(\frac{\bar{c}}{b}) \quad (18)$$

Here $\tilde{p} = pb/2V$, $\tilde{q} = q\bar{c}/2V$, $\tilde{r} = rb/2V$. In the above equations $x_{cg_{ref}}$ is taken as same as x_{cg}. In the simulation studies, all actuators are modeled as first order systems with limits on deflections and rates. The thrust has unity time constant and rate limit of 10000 lb/sec. The aerodynamic control surfaces were assumed to have a time constant of 0.0495 sec. However, the rates limites were assumed to be $\pm$60deg/sec, $\pm$80deg/sec and $\pm$120deg/sec for elevator, aileron and rudder respectively.

In this paper the velocity vector/roll maneuver is considered. The equations of motion in wind frame, which are required to synthesize a controller for this objective, are as follows

$$\dot{V_T} = (F_{wx}/m) - gsin\gamma \quad (19)$$
$$\dot{\alpha} = Q - (Q_w/cos\beta) - Pcos\alpha tan\beta - Rsin\alpha tan\beta \quad (20)$$
$$\dot{\beta} = R_w + Psin\alpha - Rcos\alpha \quad (21)$$
$$P_w = Pcos\alpha cos\beta + (Q - \dot{\alpha})sin\beta + Rsin\alpha cos\beta \quad (22)$$
$$Q_w = -(F_{wz}/mV_T) - (g/V_T)cos\gamma cos\Phi \quad (23)$$
$$R_w = (F_{wy}/mV_T) + (g/V_T)cos\gamma sin\Phi \quad (24)$$

where V_T, α, β are the total velocity, angle of attack and side slip angle respectively. P_w, Q_w, R_w are the roll, pitch and yaw rates respectively about the wind axes and F_{wx}, F_{wy}, F_{wz} are the wind axis total forces.

The equations (1)-(6) can be written as

$$\dot{X}_V = f_V(X) + \begin{bmatrix} g_V(X) & d_V \end{bmatrix} \begin{bmatrix} U_A \\ \sigma_T \end{bmatrix} \quad (25)$$
$$\dot{X}_R = f_R(X) + g_R(X)U_A \quad (26)$$

where $X \triangleq \begin{bmatrix} V_T & \alpha & \beta & P & Q & R & \Phi & \Theta & h \end{bmatrix}^T$, $X_V \triangleq \begin{bmatrix} U & V & W \end{bmatrix}^T$, $X_R \triangleq \begin{bmatrix} P & Q & R \end{bmatrix}^T$, $U_A \triangleq \begin{bmatrix} \delta_a & \delta_e & \delta_r \end{bmatrix}^T$,

$\sigma_T \triangleq T/T_{max}$, $U_c \triangleq \left[\, U_A^T \;\; \sigma_T \,\right]^T$. The normal acceleration (n_z), longitudinal acceleration (n_x) and lateral acceleration (n_y) are defined as

$$\begin{aligned} n_z &= -(F_z/m) \\ &= UQ - VP + g\cos\Phi\cos\Theta - \dot{W} \qquad (27) \\ n_x &= (F_x/m) = \dot{U} - RV + QW + g\sin\Theta \qquad (28) \\ n_y &= (F_y/m) \\ &= UR - WP - g\sin\Phi\cos\Theta + \dot{V} \qquad (29) \end{aligned}$$

Alternately, these terms can also be written as

$$n_z = f_{n_z} + g_{n_z} U_A \qquad (30)$$

$$n_x = f_{n_x} + g_{n_x} U_A \qquad (31)$$

$$n_y = f_{n_y} + g_{n_y} U_A \qquad (32)$$

Note that from equations (25) and (26) one can write:

$$\dot{V}_T = f_{V_T}(X) + \left[\, g_{V_T}(X) \quad d_{V_T} \,\right] \begin{bmatrix} U_A \\ \sigma_T \end{bmatrix} \qquad (33)$$

$$\dot{P} = f_P(X) + g_P(X) U_A \qquad (34)$$

$$\dot{Q} = f_Q(X) + g_Q(X) U_A \qquad (35)$$

Similarly, in wind axis frame, normal acceleration (n_{wz}) can be written as

$$n_{wz} = f_{n_{wz}} + g_{n_{wz}} U_A \qquad (36)$$

3. CONTROL SYNTHESIS PROCEDURE

The objective is to design a controller such that the roll angle $P \to P^*$, normal acceleration $n_z \to n_z^*$, lateral acceleration $n_y \to n_y^*$ and total velocity $V_T \to V_T^*$ where P^*, n_z^*, n_y^*, V_T^* are commanded values from the pilot. In (Menon, 1993) it is assumed that $\dot{V} = \dot{W} = 0$ and $[\ddot{\Phi}^* \;\; \ddot{\Theta}^* \;\; \ddot{\Psi}^*]^T = 0$. In this paper, it is assumed that $\ddot{V} = \ddot{W} = 0$, a more realistic assumption compared to assuming $\dot{V} = \dot{W} = 0$. Moreover, the additional assumption $[\ddot{\Phi}^* \;\; \ddot{\Theta}^* \;\; \ddot{\Psi}^*]^T = 0$ is also not necessary.

First, we define new variables a_z, a_z^* and a_y, a_y^* as

$$a_z \triangleq n_z + \dot{W}, \quad a_z^* \triangleq n_z^* + \dot{W} \qquad (37)$$

$$a_y \triangleq n_y + \dot{V}, \quad a_y^* \triangleq n_y^* + \dot{V} \qquad (38)$$

The new method relies on the key observation that $([n_z \;\; n_y]^T \to [n_z^* \;\; n_y^*]^T) \Leftrightarrow ([a_z \;\; a_y]^T \to [a_z^* \;\; a_y^*]^T)$; this is because of the one-to-one correspondence between them. From equations (27), (29), (37) and (38), it can be seen that

$$a_z = UQ - VP + g\cos\Phi\cos\Theta \qquad (39)$$

$$a_y = UR - WP - g\sin\Phi\cos\Theta \qquad (40)$$

Taking derivatives of both sides with respect to time and using Equations (7)-(9) and equations (25)-(26), we get

$$\dot{a}_z = f_{a_z}(X) + \left[\, g_{a_z}(X) \quad d_{a_z} \,\right] \begin{bmatrix} U_A \\ \sigma_T \end{bmatrix} \qquad (41)$$

$$\dot{a}_y = f_{a_y}(X) + \left[\, g_{a_y}(X) \quad d_{a_y} \,\right] \begin{bmatrix} U_A \\ \sigma_T \end{bmatrix} \qquad (42)$$

3.1 Longitudinal Maneuver

In the longitudinal maneuver case, goal is $X_T \to X_T^*$ and $V_T \to V_T^*$, where $X_T \triangleq [P \quad n_z \quad n_y]^T$, $X_T^* \triangleq [P^* \quad n_z^* \quad n_y^* = 0]^T$. Defining $\hat{X}_T \triangleq (X_T - X_T^*)$, a controller is designed such that the stable error dynamics has the following structure.

$$\dot{\hat{X}}_T + K\hat{X}_T = 0 \qquad (43)$$

Here the gain matrix K is a positive definite matrix and is selected as $K = diag(1/\tau_P, 1/\tau_{n_z}, 1/\tau_{n_y})$. Carrying out the necessary algebra, an expression for the controller reduces to

$$\begin{aligned} U_A &= A_U^{-1} b_U \qquad (44) \\ \textit{where } A_U &\triangleq [g_P^T \;\; g_{a_z}^T \;\; g_{a_y}^T]^T + K[0^T \;\; g_{n_z}^T \;\; g_{n_y}^T]^T \\ b_U &\triangleq -[f_P \;\; f_{a_z} \;\; f_{a_y}]^T \\ &\quad - K[(P - P^*) \;\; (f_{n_z} - n_z^*) \;\; (f_{n_y} - n_y^*)]^T \end{aligned}$$

Now define $\hat{V}_T \triangleq (V_T - V_T^*)$, which is considered as slow variable as compared to $\hat{X}_T$ and error dynamics can be defined as

$$\dot{\hat{V}}_T + K_{V_T}\hat{V}_T = 0 \qquad (45)$$

where the gain matrix K_{V_T} is selected to be a positive definite matrix. By solving the equation (33) an expression for thrust can be expressed as:

$$\sigma_T = d_{V_T}^{-1} c_{V_T} \qquad (46)$$

$$c_{V_T} = \{(f_{V_T} + g_{V_T} U_A) - \dot{V}_T^* + K_{V_T}(V_T - V_T^*)\} \qquad (47)$$

The control vector $[\delta_a \quad \delta_e \quad \delta_r]$ is updated after every time step dt while σ_T is updated after every five time steps $5dt$, as it is slow variable as compared to $\delta_a, \delta_e, \delta_r$.

3.2 Lateral Maneuver

During lateral maneuver, the objectives are to drive $P \to P^*, n_y \to n_y^*, h \to h^*$ *and* $V_T \to V_T^*$. Note that the appropriate n_z^* (such that $h \to h^*$)is automatically computed in this process. An error expression is defined as $\hat{h} \triangleq (h - h^*)$ and a stable height-error dynamics is formulated as

$$\dot{\hat{h}} + (1/\tau_h)\hat{h} = 0 \qquad (48)$$

where τ_h is the desired time constant. Substituting the value for $\dot{h}$ from equation (10), this can be expanded as

$$\begin{aligned} &[U\sin\Theta - V\cos\Theta\sin\Phi - W\cos\Theta\cos\Phi] \\ &\quad - \dot{h}^* + (1/\tau_h)(h - h^*) = 0 \qquad (49) \end{aligned}$$

The variable Θ is solved from equation (49) and denoted as Θ^*. Next, a stable first-order error dynamics is enforced for the pitch angle as follows

$$\dot{\hat{\Theta}} + (1/\tau_\Theta)\hat{\Theta} = 0 \tag{50}$$

where $\hat{\Theta} \triangleq (\Theta - \Theta^*)$ and $\tau_\Theta > 0$ is the desired time constant. Substituting for $\dot{\Theta}$ equation from equation (8) and assuming Θ^* to be constant at each instant of time (quasi-steady assumption), an expression for Q (and denote it as Q^*) can be obtained as

$$Q^* = (1/\cos\Phi)[R\sin\Phi - (1/\tau_\Theta)(\Theta - \Theta^*)] \tag{51}$$

Since U_A appears in the $\dot{Q}$ equation (35), it facilitates control computation as follows. Defining $X_T \triangleq [P \;\; Q \;\; n_y]$, $X_T^* \triangleq [P^* \;\; Q^* \;\; n_y^* = 0]$ and $\hat{X}_T \triangleq (X_T - X_T^*)$, the objective is to synthesize a controller such that equation (43) is satisfied. In this case, the gain matrix is selected as $K = diag(1/\tau_P, 1/\tau_Q, 1/\tau_{n_y})$. Following the steps outlined before and carrying out the necessary algebra, an expression for control can be written in the following form.

$$\begin{aligned} U_A &= A_U^{-1} b_U \qquad (52)\\ \text{where } A_U &\triangleq [g_P^T \;\; g_Q^T \;\; (g_{a_y}^T + (1/\tau_{n_y})g_{n_y}^T)]^T \\ b_U &\triangleq -[f_P \;\; f_Q \;\; f_{a_y}]^T \\ &\quad - K[(P - P^*) \;\; (Q - Q^*) \;\; (f_{n_y} - n_y^*)]^T \end{aligned}$$

σ_T can be calculated by using the same expression as used in section (3.1).

3.3 Combined Longitudinal and Lateral Maneuver

In this case, goal is $X_T \to X_T^*$ and $V_T \to V_T^*$, where $X_T \triangleq [P \;\; n_z \;\; n_y]^T$, $X_T^* \triangleq [P^* \;\; n_z^* \;\; n_y^* = 0]^T$. Here P_w^* command is given about velocity vector and further P^* is calculated from P_w^*. The significance of this maneuver is obvious, for it allows the pilot to quickly slew and point the aircraft's nose using a presumably "fast" roll maneuver, without pulling g's and turning. Using equations (20), (22) and (23), P^* can be expressed in control affine form as

$$\begin{aligned} P^* &= f_{P^*} + g_{P^*} U_A \qquad (53)\\ \text{where } f_{P^*} &= (1/cos\alpha(cos\beta + tan\beta sin\beta))(P_w^* \\ &\quad - Rsin\alpha(sin\beta tan\beta + cos\beta) \\ &\quad - tan\beta(f_{n_{wz}}/V_T) + (g/V_T)cos\gamma cos\Phi) \\ g_{P^*} &= (-tan\beta/cos\alpha(cos\beta \\ &\quad + tan\beta sin\beta))(g_{n_{wz}}/V_T) \end{aligned}$$

Now, a controller is designed such that the stable error dynamics has the following structure.

$$\dot{\hat{X}}_T + K\hat{X}_T = 0 \tag{54}$$

where the gain matrix K is selected to be a positive definite matrix. Using equations (34), (42), (41) and (53), and carrying out the necessary algebra, an expression for the controller reduces to

$$\begin{aligned} U_A &= A_U^{-1} b_U \qquad (55)\\ \text{where } A_U &\triangleq [g_P^T \;\; g_{a_z}^T \;\; g_{a_y}^T]^T + K[-g_{P^*}^T \;\; g_{n_z}^T \;\; g_{n_y}^T]^T \\ b_U &\triangleq -[f_P \;\; f_{a_z} \;\; f_{a_y}]^T \\ &\quad - K[(P - f_{P^*}) \;\; (f_{n_z} - n_z^*) \;\; (f_{n_y} - n_y^*)]^T \end{aligned}$$

Note that after designing the aerodynamic controller, the thrust control σ_T is calculated by using the same expression as used in section (3.1).

4. NUMERICAL RESULTS

4.1 Numerical Values Selection

All numerical data used in simulations for F-16 are taken from NASA report (Nguyen, 1979). A fourth-order Runge-Kutta technique with fixed step size 50msec was used for numerical integration.

Trim Condition: The trim condition for steady level flight is calculated by minimizing the cost function (J) (Russell, 2003), which is given as

$$\begin{aligned} J = 5\dot{h}^2 + W_\Phi\dot{\Phi}^2 + W_\Theta\dot{\Theta}^2 + W_\Psi\dot{\Psi}^2 + 2\dot{V_T}^2 \\ + 10\dot{\alpha}^2 + 10\dot{\beta}^2 + 10\dot{P}^2 + 10\dot{Q}^2 + 10\dot{R}^2 \quad (56) \end{aligned}$$

where $W_\Phi = W_\Theta = W_\Psi = 10$. The trim condition values found at specified velocity $V_{T_0} = 580 \; (ft/sec)$ and altitude $h_0 = 10{,}000 ft$ are: $\alpha_0 = 1.497 \; deg$, $\beta_0 = 0 \; deg$, $\Phi_0 = 1.497 \; deg$, $\Theta_0 = 0 \; deg$, $\Psi_0 = 0 \; deg$, $\delta_{a_0} = 0 \; deg$, $\delta_{e_0} = -1.81 \; deg$ $\delta_{r_0} = 0 \; deg$ and $\sigma_{T_0} = 0.09$

Selection of Control Design Parameters: After some trial and error the values selected for the time constants are: $\tau_P = 0.3$, $\tau_{n_z} = 2.5$, $\tau_{n_y} = 2$, $\tau_{V_T} = 3$, in the longitudinal case and $\tau_P = 0.3$, $\tau_{n_y} = 2$, $\tau_{V_T} = 3$, $\tau_\Theta = 0.2$, $\tau_Q = 0.15$, $\tau_h = 5$ in the lateral case. In order to compare the performance of the modified formulation with the existing version (Menon, 1993), gain values of $k_1 = k_3 = 1$, $k_2 = k_4 = 30$ were selected for the command augmentation system. Similarly in the attitude orientation system, parameter values of $k_{v_i} = 2\zeta_i\omega_{n_i}$, $k_{p_i} = \omega_{n_i}^2$ with $\zeta_1 = 1.5$, $\zeta_2 = 0.9$, $\zeta_3 = 0.9$ and $\omega_1 = 2$, $\omega_2 = 5$, $\omega_3 = 5$ rad/sec ($i = 1,2,3$) were selected for each of the attitude angle error dynamic channels. It is important to point out that in the new approach only five design parameters are needed for the longitudinal mode and only seven are needed for the lateral mode compared to the requirement of eleven and twelve parameters in the existing approach. This significantly less number of design parameters without compromising in performance is clearly a potential advantage of the new approach.

4.2 Analysis of Results

In our numerical studies, the goal was to track the reference commands for 90 sec in longitudinal case and 60 sec in lateral case. In all plots, the solid lines represent the results from the new approach, whereas the dashed lines represent the results from the existing approach (Menon, 1993).

Longitudinal Maneuver: In figures 1 and 2 simulation results for a longitudinal maneuver are shown. The initial conditions are same as trim values. The upper value of n_z^* for high performance aircrafts (Keviczky and Balas, 2006) is 2.0g. The following sequence of command signals are input: $[\Phi^* \;\; n_y^* \;\; V_T^*] = [0 \;\; 0 \;\; V_{T_0}]$ throughout the maneuver. $n_z^* = 0.9965g$ for $t = 0-1$ sec, $2.0g$ for $t = 1-15$ sec, $0.5g$ for $t = 15-65$ sec, $1.0g$ for $t = 65-90$ sec.

The new approach offers several improvements. In

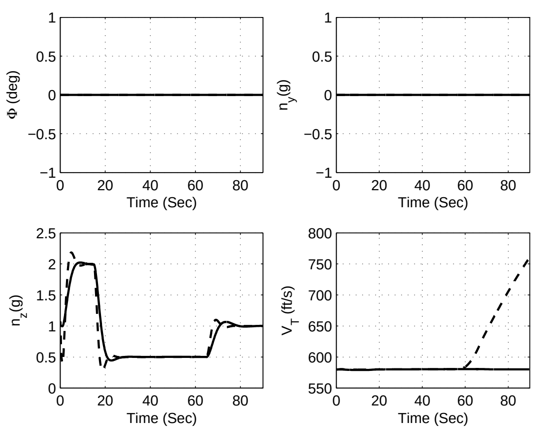

Fig. 1. Roll angle, Normal acceleration, Lateral acceleration and Total velocity in longitudinal maneuver

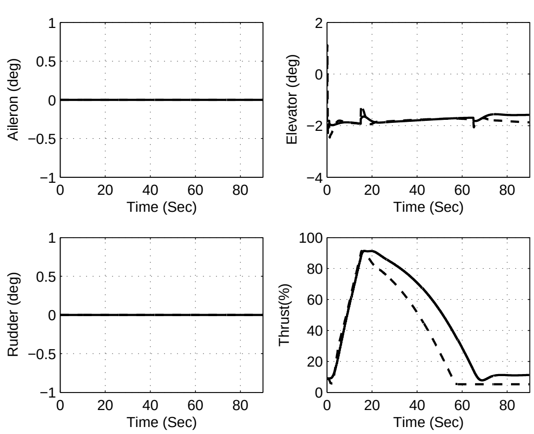

Fig. 2. Aileron, Elevator and Rudder deflections and Thrust level in longitudinal maneuver

figure 1, the transient oscillations have much smaller overshoot and the frequency of oscillation is less. Normal acceleration is eventually tracked in the existing approach successfully, but initially it shows a non-minimum phase behavior. Moreover the final elevator deflection requirement is less in this new approach. Moreover, the existing approach exhibits some high-frequency oscillations in the elevator as compared to this new approach. The main difference here is that, in existing method thrust saturation starts very early as compared to this new method which is shown in figure 2, and it goes below the lower limit of thrust (Nguyen, 1979), and after saturation velocity deviates from its desired goal, but in this method it is able to recover and the desired goal is achieved, which provides better tracking.

Lateral Maneuver: Simulation results for a lateral maneuver from the trim condition are presented in figures 3 -4. The sequence of command signals applied consisted of $P^* = -10\ deg/sec$ for $t = 0-7$ sec, $10\ deg/sec$ for $t = 7-14$ sec and $0\ deg/sec$ for $t = 14-60$ sec. Throughout the maneuver, it was

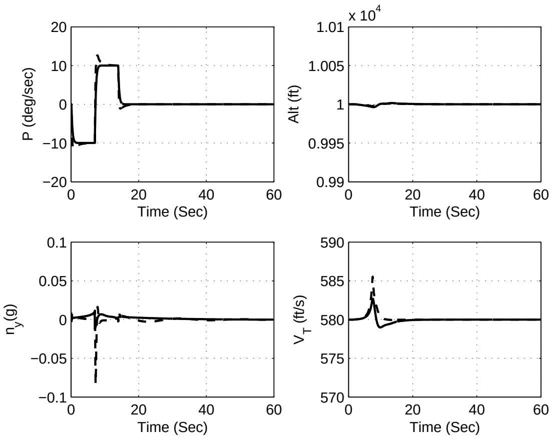

Fig. 3. Roll rate, height, Lateral acceleration and Total velocity in lateral maneuver

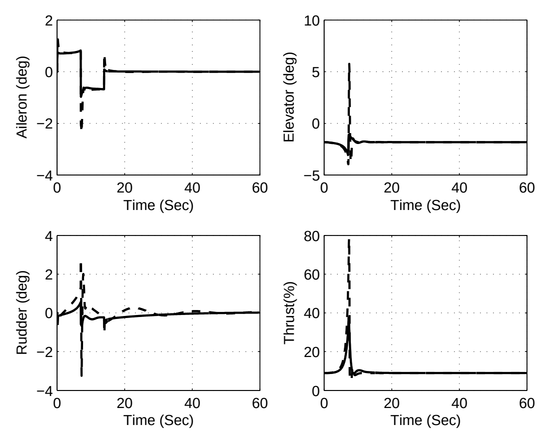

Fig. 4. Aileron, Elevator and Rudder deflections and Thrust level in lateral maneuver

assumed that $V_T^* = V_{T_0}, \;\; h^* = h_0$ (the initial condition values). Note that the lateral accelerations in both approaches remain close to zero, which is a requirement for the maneuvers. However, in the existing approach at 7 sec, aileron, elevator and rudder deflections are also more, but the elevator deflection is quite high. Besides, in the existing approach the elevator and rudder deflection histories show high frequency transient oscillations. These trends are absent in the performance of the modified approach. From the graph, it can be seen that the thrust required in existing method is also more (i.e. around double) as compared to new method, which is again an advantage. Simulation studies for a large number of cases did not show instability in any of the cases.

Combined longitudinal and lateral maneuver: In most of the papers (Qian and Stengel, 2005), (Pachter, 1996) velocity vector roll is performed at constant angle of attack, which is predominantly a lateral maneuver. But in this paper, normal acceleration command is given with velocity vector roll, which is difficult task and this design works well for this combined maneuver. Simulation results for combined longitudinal and lateral maneuver are presented in figures 5 and 6. In this maneuver the pilot commands are normal acceleration, velocity vector roll rate, lateral acceleration and total velocity. Note that results are also verified with constant angle of attack which are not presented here because of space restrictions.

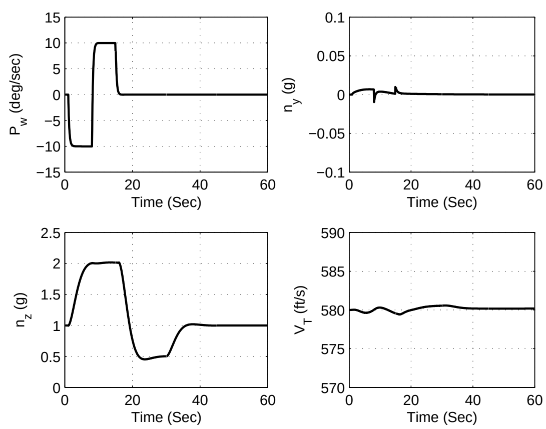

Fig. 5. Roll rate, Lateral acceleration, Normal acceleration and Total velocity in combined maneuver

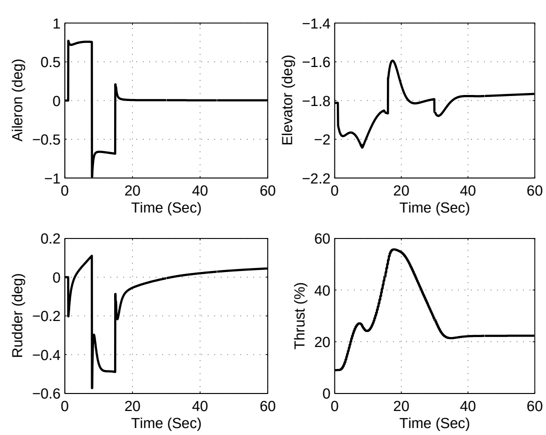

Fig. 6. Aileron, Elevator and Rudder deflections and Thrust level in combined maneuver

5. CONCLUSIONS

A new approach based on dynamic inversion technique is presented in this paper for implementation of pilot commands in high performance aircrafts. An important advantage of this approach over an existing approach is that a fewer number of design parameters are needed. The comparison studies support the view that the new approach has a better transient response and demands lower magnitudes of control. These are again desirable features in a controller. Also there is no need of integral control. The new approach demands lesser control magnitudes and leads to better transient response.

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