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To cite this version:
Hector Escamilla Nuñez, Felix Antonio Claudio Mora-Camino, Hakim Bouadi. Towards 4D Trajectory Tracking for Transport Aircraft. IFAC’2017, 20th World Congress of the International Federation of Automatic Control, Jul 2017, Toulouse, France. <hal-01575758>

HAL Id: hal-01575758
https://hal-enac.archives-ouvertes.fr/hal-01575758
Submitted on 21 Aug 2017

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Towards 4D Trajectory Tracking for Transport Aircraft.

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Abstract: This paper presents a new approach to perform 4D trajectory tracking for transportation aircraft. As current systems are extensions of 3D guidance with overfly time constraints at some given points and no general control framework has been developed for 4D guidance of a transport aircraft; the main goal of the proposed approach is to introduce a new method based on the inversion of the flight dynamics while avoiding numerical issues. A six degree of freedom model for a wide body transportation aircraft was developed in Matlab to provide a numerical simulation of the proposed approach, showing satisfying results.

Keywords: Trajectory Tracking. Transportation Aircraft. Navigation, Guidance and Control. Automatic control. Non Linear Inversion.

1. INTRODUCTION

As air traffic is predicted to increase dramatically in the upcoming years, new problems and requirements are arising, among those related with the use of 4D trajectories are of most interest. NextGen FAA (2016) and SESAR EUROCONTROL (2016) control projects, where traffic capacity and safety issues are central, adopt the Trajectory Based Operations (TBO) paradigm, which supposes 4D guidance effectiveness. Then, an important enabler is automation, allowing aircraft to follow with more accuracy flight plans characterized by a 4D reference trajectory. It is expected that accurate 4D guidance will improve safety by decreasing the occurrence of near mid-air collisions for planned conflict free 4D trajectories, and then diminish the workload associated to a single flight for air traffic controllers. This leads to propose in this paper a new approach to perform 4D guidance.

Until today no general control framework has been developed for 4D guidance of a transport aircraft and current systems are extensions of 3D guidance with overfly time constraints at some given points. In that case control laws are based on frequency decoupling and different PID control layers with gain scheduling provisions. Some attempts have been performed recently Wahid et al. (October 2014), Wahid et al. (2016) using mainly Non Linear Inversion (NLI). Other authors have proposed to extent the energy-based control approach Lambregts (1983) to 4D guidance Lambregts (1996), Chudy and Rzucidlo (August 2009). However in both cases serious limitations appear.

In this study it is considered that aircraft dynamics are composed of fast dynamics related with the angular attitude of the aircraft and of slow dynamics related with the trajectory followed by the aircraft, referenced at its center of gravity. For modern transportation aircraft with Fly by Wire (FBW) technology, the autopilot is in charge of controlling the aircraft angular attitude, improving flying qualities through stabilization and generating automatic protections in dangerous piloting situations. In this paper, the autopilot will be considered to be an specific device providing in an integrated way these essential functions, so that the attention will be focused on the auto guidance system in charge of controlling the slow flight dynamics.
with the aim of providing 4D guidance. The paper is organized as follows, Section 2 provides the developed mathematical model and actuator dynamics. In Section 3 are discussed two previous approaches pointing out their limitations for 4D guidance. Section 4 describes the adopted control structure and Section 5 shows the simulation results of the control approach. Finally, conclusions are given in Section 6.

2. FLIGHT DYNAMICS

2.1 Adopted Frames and Flight Variables

The Earth reference frame is considered in this paper to describe the 4D trajectories. It is assumed that the Earth reference frame is Earth centered and is denoted by $F_E = (O_E, x_E, y_E, z_E)$. Then, a 4D reference trajectory can be defined by three parameters associated with the coordinates of the center of gravity of the aircraft supposed to follow it. These functions are parameterized by: $x_E(t), y_E(t), z_E(t), t \in [t_{init}, t_{end}]$, where $t_{init}$ is the reference flight starting time and $t_{end}$ is the reference flight ending time. A second reference frame, the body frame, is considered to represent the fast dynamics of the aircraft, it is attached to the aircraft c.g. and is defined as $F_B = (C, x_B, y_B, z_B)$. The $x_B$ axis goes from tail to nose of the aircraft while the $z_B$ axis, is considered to represent the fast dynamics of the aircraft. It is assumed that the wind frame can be mapped into the body frame by the following rotation matrix:

$$L_{EB} = \begin{bmatrix} c_\phi c_\psi & s_\phi c_\psi & -s_\psi \\ s_\phi s_\psi & c_\phi s_\psi & c_\psi \\ c_\theta s_\phi & s_\theta s_\phi & c_\phi \end{bmatrix}$$

where $\phi, \theta, \psi$ are the Euler angles. The difference between the wind frame and the body frame is expressed in the body frame, is given by:

$$\alpha = \arctan\left(\frac{w - V_{aw}}{u - V_{aw}}\right)$$

and

$$\beta = \arcsin\left(\frac{v - V_{aw}}{V_n}\right)$$

where $V_{aw}$ is the wind components in the body frame.

2.2 Attitude Dynamics

Concerning the attitude of the aircraft, the angular rates ($\Omega = [p, q, r]^T$) are produced by the deflection of ailerons, elevator and rudder, denoted by $[\delta_{ail}, \delta_{ele}, \delta_{rud}]^T$. The rotational equations of the aircraft are given by:

$$\dot{\Omega} = I^{-1}M_{ext} - \Omega \times (I\Omega)$$

where $M_{ext} = [L', M, N]^T$ are the rolling, pitching and yawing moments respectively, and $I$ stands for the inertia matrix, in kg·m²:

$$I = \begin{bmatrix} A & 0 & -E \\ 0 & B & 0 \\ -E & 0 & C \end{bmatrix} = \begin{bmatrix} 1,278,369.56 & 0 & -135,588.17 \\ 0 & 3,781,267.79 & 0 \\ -135,588.17 & 0 & 4,877,649.98 \end{bmatrix}$$

Thereby, introducing relations for the aerodynamic moments:

$$\frac{L'}{M} = \frac{1}{2} \rho S V_n^2 C_L \left[c_C + C_b \delta_{ail} \right] + C_b \delta_{ele} \delta_{rud}$$

where

$$C_b = \begin{bmatrix} C_{b_{init}} & 0 & b_{C_{brd}} \\ 0 & c_{C_{m_{ele}}} & 0 \\ b_{C_{n_{ail}}} & 0 & C_{b_{n_{rud}}} \end{bmatrix}$$

and $\rho$ is the air density, $S$ wing area, $b$ wingspan, $c$ mean chord, and the rolling, pitching and yawing aerodynamic coefficients ($C_L, C_m, C_n$ respectively) are denoted by:

$$C_L = \begin{bmatrix} C_{l_{\beta}} \beta + C_{l_{\psi}} \frac{\psi}{\psi} + C_{l_{\phi}} \frac{\phi}{\phi} \\ C_{m_{\alpha}} + C_{m_{\phi}} \phi + C_{m_{\psi}} \frac{\psi}{\psi} \\ C_{n_{\phi}} \beta + C_{n_{\psi}} \frac{\psi}{\psi} + C_{n_{\phi}} \frac{\phi}{\phi} \end{bmatrix}$$
Then, equation (5) can be rewritten as in Lombaerts et al. (May-June 2009) like:

\[
\dot{\Omega} = \frac{1}{2} \rho V_a^2 S T^{-1} \begin{bmatrix}
bc \\
ccm \\
bcm
\end{bmatrix}
+ C_3 \begin{bmatrix}
\delta_{ail} \\
\delta_{e\text{le}} \\
\delta_{r\text{ud}}
\end{bmatrix} - I^{-1} \Omega \times (\Omega) \quad (10)
\]

Subsequently, differentiating (4a), (4b), and using (3), (14), (15), an expression rearranged for \( \Omega \) is obtained:

\[
\begin{bmatrix}
\dot{\alpha} \\
\dot{\beta}
\end{bmatrix} = \begin{bmatrix} H_{11} & H_{12} & H_{13} \\ H_{21} & H_{22} & H_{23} \end{bmatrix} \begin{bmatrix} p \\ q \\ r \end{bmatrix} + \begin{bmatrix} Q_1 \\ Q_2 \end{bmatrix} \quad (11)
\]

where

\[
H_{11} = -\left(\tan \beta C_\alpha + \frac{V_w^2 C_\alpha}{V_a C_\beta}\right)
\]
\[
H_{12} = 1 + \frac{V_w C_\alpha V_a + V_w s_n}{V_a C_\beta}
\]
\[
H_{13} = -\left(\tan \beta s_n + \frac{V_w s_n}{V_a C_\beta}\right)
\]
\[
H_{21} = s_n + \frac{V_w c_\beta + V_w s_n}{V_a C_\beta}
\]
\[
H_{22} = V_w c_\alpha s_n - V_w s_n
\]
\[
H_{23} = -\left(c_n + \frac{V_w}{V_a} c_\alpha s_n + \frac{V_w}{V_a} c_\beta\right)
\]

\[
Q_1 = \frac{1}{V_a C_\beta} \left(g_1 - \frac{1}{m} (L + F_{\text{thr}, \alpha})\right) + \frac{1}{V_a C_\beta} (V_{wz_c} s_a - V_{wz_c} c_a)
\]
\[
Q_2 = \frac{1}{V_a} \left(g_2 + \frac{1}{m} (Y - F_{\text{thr}, c\alpha})\right)
\]
\[
\left.\begin{array}{c}
g_1 = g \left(c_{\alpha \beta} c_\phi + s_a s_\theta\right) \\
g_2 = g \left(c_{\beta \phi} s_\theta + s_\beta s_\alpha c_\theta - s_\alpha s_\beta c_\phi c_\theta\right) \\
\end{array}\right\}
\]

which can be written as:

\[
R = H(R) \Omega + Q(R) \quad (12)
\]

The rotation speed components are related with the attitude angle rates by the Euler equations given by:

\[
\begin{bmatrix}
\dot{\phi} \\
\dot{\theta} \\
\dot{\psi}
\end{bmatrix} = \begin{bmatrix}
1 & 0 & 0 \\
0 & c_\phi & -s_\phi \\
0 & \frac{s_\phi}{c_\phi} & \frac{c_\phi}{c_\phi}
\end{bmatrix} \begin{bmatrix} p \\ q \\ r \end{bmatrix} \quad (13)
\]

### 2.3 Guidance Dynamics

An acceleration equation in the body frame is denoted by:

\[
\begin{bmatrix}
\dot{u} \\
\dot{v} \\
\dot{w}
\end{bmatrix} = \begin{bmatrix}
\frac{1}{m} (F_{xa} + F_{\text{thr}}) - g s_\theta + r v - q w \\
\frac{1}{m} F_{ya} + g c_\theta s_\phi + p w - r u \\
\frac{1}{m} F_{za} + g c_\theta c_\phi + q u - p v
\end{bmatrix} \quad (14)
\]

where \( m \) is the mass, \( g \) gravity, \( F_{\text{thr}} \) thrust force, and \( F_{xa}, F_{ya}, F_{za} \) the aerodynamic forces dependent on Lift \((L)\), Drag \((D)\) and Sideforce \((Y)\), which simultaneously are related with their aerodynamic force coefficients by:

\[
\begin{bmatrix}
D \\
Y \\
L
\end{bmatrix} = \frac{1}{2} \rho S V_a^2 \begin{bmatrix}
C_D \\
C_Y \\
C_L
\end{bmatrix} \quad (16)
\]

Then, an expression in the Earth frame is obtained:

\[
\begin{bmatrix}
\dot{x}_E \\
\dot{y}_E \\
\dot{z}_E
\end{bmatrix} = L_{EB} \begin{bmatrix}
F_{xa} + F_{\text{thr}} \\
F_{ya} \\
F_{za}
\end{bmatrix} + \begin{bmatrix} 1 \\ \frac{1}{m} \\ g \end{bmatrix} \quad (17)
\]

Regarding to airspeed in the wind frame, after differentiating (4c), and using (3), (14), (15), it is obtained that:

\[
V_{a} = g_3 + \frac{1}{m} \left(F_{\text{thr}, \alpha} c_\beta - D\right) + p \left(V_{wz_c} s_n - V_{wz_c} c_n\right)
\]
\[
+ q \left(V_{wz_c} s_n - V_{wz_c} c_n\right) + r \left(V_{wz_c} c_n - V_{wz_c} s_n\right)
\]
\[
- V_{wz_c} c_n c_\beta - V_{wz_c} s_n - V_{wz_c} s_n c_\beta
\]

with

\[
g_3 = g \left(-c_n c_\beta s_\theta + s_\beta c_\phi s_\theta + s_\alpha c_\beta c_\phi c_\theta\right)
\]

Then, neglecting wind disturbances and taking into account a longitudinal decoupled motion, (18) can be reduced to:

\[
V_{a} = \frac{1}{m} \left(F_{\text{thr}, \alpha} c_\beta - D - m g s_\gamma\right) \quad (19)
\]

Introducing the Flight Path Angle \((\gamma = \theta - \alpha)\), the vertical motion is given by:

\[
\dot{\gamma} = \frac{1}{m V_a} \left(F_{\text{thr}, \alpha} + L - m g \cos \gamma\right) \quad (20)
\]

Considering that transport aircraft perform through their yaw stabilizer equilibrated turns, the heading rate is given by:

\[
\dot{\psi} = \frac{g}{V_a} \tan \phi \quad (21)
\]

#### 2.4 Actuator Dynamics

Let a first-order model be adopted for the aerodynamic actuators, writing \( \delta_i^i \) \((i = \text{ail, ele, rud})\) as the commanded positions of the control surfaces, and \( \delta_i \) as the current positions of the control surfaces:

\[
\dot{\delta}_i = \frac{1}{\xi_i} (\delta_i^i - \delta_i) \quad (22)
\]

where \( \xi_i \) are the time-constants. Also, the resultant thrust produced by the engines is supposed to behave as a first-order system, denoted by

\[
F_{\text{thr}} = \frac{1}{\xi_T} (F_{\text{thr}} - F_{\text{thr}}) \quad (23)
\]

where the \( F_{\text{thr}} \) is the desired thrust and \( F_{\text{thr}} \) the current thrust. Besides, the time-constants of the actuators keep the relation: \( \xi_T >> \xi_i \).
3. PROPOSED CONTROL APPROACHES

In this part are briefly discussed two previous control approaches that resulted in the design of 4D guidance devices. The first one is related to energy-based control while the second is related with direct Non Linear Inversion control. Limitations of both methods are pointed out.

3.1 Total Energy Control Approach

In this method, the concept of a potential flight path angle denoted by: \( \gamma_p = \gamma + \frac{V_a}{g} \) is considered. It indicates the potential path angle that can be achieved by bringing the acceleration to zero by applying elevator until \( \gamma \) becomes \( \gamma_p \). This potential angle is found to be related with the specific total energy rate of the aircraft, therefore, the approach consists in controlling the behaviour of the energy rate by controlling \( \gamma_p \) using the thrust and elevator. However, the control of this variable is performed with non available plant dynamics while introducing empirically an energy rate distribution variable \( \dot{L} = \gamma - \frac{V_a}{g} \) to distribute the energy rate between the flight path angle and acceleration. This distribution variable is controlled directly by the elevator, bypassing the fast dynamics and arising difficulties to integrate other autopilot functions. Furthermore, the speed and altitude dynamics in this approach are expected to be identical, bringing to light a problem for some 4D trajectories where this is not true. Finally, no integral term has been proposed to force online position error to zero.

3.2 Direct Non Linear Inverse Control Approach

This method distinguishes between fast and slow dynamics. The variables used to create the link between these dynamics are the components \( p, q, r \) of the rotational speed. Therefore, two layers of inversion are considered for this approach. The first regards to the fast dynamics and allows to determinate the necessary position of the actuators (deflections of aileron, elevator and rudder) as functions of the desired angular velocities, based on equations (10),(12),(18), and (22). The second layer takes care of the slow dynamics, and allows to obtain the angular velocities and Thrust required to follow the 4D commanded trajectory.

However, in the second layer, this approach presents as main drawback a singularity when inversion is performed to get the input parameters to the slow dynamics. This problematic appears after introducing the control inputs \([p, q, F_{thr}]^T\) in the equation of the outputs, which is made by obtaining the jerk vector of the positions in \( F_E \) by differentiating (17). The control inputs appear in the matrix \( \dot{L}_{EB} \). This is possible thanks to the Euler property and (13), which allows to rewrite \( \dot{L}_{EB} \) and the skew-symmetric matrix of the angular velocities \( \left( \dot{L}_{EB} = L_{EB} \Omega \right) \). The resultant equation using the vector \( F_{thr} = [F_{xa} + F_{chh}, F_{ya}, F_{za}]^T \) and considering that the mass rate of change is very small compared to the aircraft total mass, has the form:

\[
\begin{bmatrix}
\dot{r}_{(3)}^E \\
\dot{\gamma}_{(3)}^E \\
\dot{\xi}_{(3)}^E
\end{bmatrix} = \frac{1}{m} M_{pqT} \begin{bmatrix}
p \\
q \\
F_{thr}
\end{bmatrix} + \frac{1}{m} M_r F_{ab} F_{za} F_{ya} \begin{bmatrix}
c_0 c_\psi c_\phi \\
c_0 s_\psi s_0 - s_\psi s_0 c_\phi \\
s_\phi c_\theta \end{bmatrix}
\]

where

\[
M_{pqT} = \begin{bmatrix}
M_p F_{ab} & M_p F_{ab} & c_0 c_\phi s_\psi \\
0 & s_0 & -s_0 c_\phi \\
-M_{pqT} & -M_{pqT} & 0
\end{bmatrix}
\]

with

\[
M_p = \begin{bmatrix}
0 & c_0 s_\phi c_\psi + s_\phi s_\psi - s_\phi s_\phi c_\psi & c_0 s_\psi s_0 - s_\phi s_\phi c_\psi \\
0 & c_0 s_\phi c_\psi + s_\phi s_\psi - s_\phi s_\phi c_\psi & -c_0 s_\psi s_0 \\
0 & c_0 s_\phi c_\psi + s_\phi s_\psi - s_\phi s_\phi c_\psi & 0
\end{bmatrix}
\]

and

\[
M_r = \begin{bmatrix}
-c_\phi s_\psi c_\phi - s_\psi s_0 & 0 & c_\phi s_\psi s_0 - c_\phi c_\psi \\
-c_\phi s_\psi c_\phi - s_\psi s_0 & 0 & -c_\phi s_\psi s_0 \\
0 & c_\phi s_\psi c_\phi + s_\psi s_0 & 0
\end{bmatrix}
\]

Note that to develop a control law using NLI or any other model-based-approach relying on equation (24), the inverse of (25) needs to be computed.

In the case in which \( \phi \approx \psi \approx 0 \), the determinant of this matrix is given by:

\[
|M_{pqT}| = s_\phi F_{xa} + (s_\phi (F_{xa} + F_{thr}) - s_\phi F_{xa}) (s_\phi F_{xa}) + c_\phi (c_\phi (F_{xa} + F_{thr}) + s_\phi F_{xa}) F_{za}
\]

and, when considered \( \theta \approx 0 \), a singularity appears for \( F_{xa} + F_{thr} = 0 \). This is the case when the plane is cruising at constant speed. Therefore, taking into account that the cruise phase of a flight is essential, and that the airplane will go through this condition very often, any algorithm using this control approach should be discarded. This result is not surprising, realizing that when a cruise flight with the Euler angles near zero is performed, the matrix \( L_{EB} \) will remain constant, so its derivative is expected to tend to zero. Furthermore, it is worth to say that even if the first derivative of the Euler angles along with the Thrust are considered as the control inputs, \([\dot{\phi}, \dot{\theta}, \dot{F}_{thr}]^T \) for instance, like in Wahid et al. (2016), the corresponding matrix (similar to (25)) will be described by (28) and will also present singularities during the cruise phase.
Equations (19), (20), and (21) can be rewritten as inverting the flight dynamics. This will be made possible when the singularity issues while implementing in the simulations is based on an inversion of the fast dynamics, like in Escamilla-Núñez and Kim (November 2003), Ali et al. (2010), Mattei and Monaco (October 2014). In this work, the autopilot version of the fast dynamics, like in Escamilla-Núñez et al. (January 2017) (see Section 7).

So another method should be searched in order to avoid this singularity.

4. PROPOSED 4D CONTROL APPROACH

The main objective for the proposed 4D trajectory tracking control strategy is to avoid singularity issues while inverting the flight dynamics. This will be made possible by considering three phases:

(1) Control of the longitudinal and lateral motions, θ and φ, with stabilisation in yaw ψ. We will consider that thanks to the relation α = θ − γ, to command θ when γ is known, is equal to command the AoA. The control of θ and φ while stabilizing yaw motion can be performed using classical control techniques implemented in the simulations is based on an inversion of the fast dynamics, like in Escamilla-Núñez et al. (January 2017) (see Section 7).

(2) Control of Speed and the Flight Path Angle as functions of AoA and Thrust using the guidance equations (19), (20).

(3) Adjustment of the heading angle given by (21), targeted speed, and desired flight path angle as functions of the positions errors generated by a 4D reference trajectory.

Proposing desired dynamics for the speed \( V_a^d \), heading \( \psi^d \), and flight path angle \( \gamma^d \) as first order linear responses, they are such as:

\[
V_a = \frac{1}{\tau_V} (V_a^d - V_a)
\]

\[
\gamma = \frac{1}{\tau_\gamma} (\gamma^d - \gamma)
\]

\[
\psi = \frac{1}{\tau_\psi} (\psi^d - \psi)
\]

where \( \tau_V, \tau_\gamma, \tau_\psi \) are time-constants.

Equations (19), (20), and (21) can be rewritten as

\[
\begin{bmatrix}
M_{\theta} F_{\theta b}
M_{\theta} F_{ab}
\end{bmatrix}
\begin{bmatrix}
c_\theta c_\psi
-c_\theta s_\psi
\end{bmatrix}

\[
M_{\theta} = \begin{bmatrix}
-s_\theta c_\psi & c_\theta s_\psi & c_\theta c_\psi
-s_\theta s_\psi & c_\theta s_\psi & c_\theta s_\psi
-c_\theta & -s_\theta & c_\theta
\end{bmatrix}
\]

So another method should be searched in order to avoid this singularity.

\[
\begin{align*}
J &= \begin{bmatrix}
\tau_V \left(V_a^d - V_a\right) + m g s_\gamma & F_{thr} c_\alpha - D(p, \alpha, V_a) \\
\frac{m V_a}{\gamma} \left(\gamma^d - \gamma\right) + m g c_\gamma & F_{thr} s_\alpha + L(p, \alpha, V_a) \\
tg^{-1} \left(\frac{V_a}{g s_\psi} (\psi^d - \psi)\right) = \phi
\end{bmatrix}
\end{align*}
\]

Therefore, in order to obtain the values of the thrust and AoA required to follow their proposed dynamics, we solve the nonlinear set of equations (31a), (31b) for \( F_{thr} \) and \( \alpha \), and apply saturations if the results are out of the thrust \( [F_{thr}^{min}, F_{thr}^{max}] \) and angle of attack \( [\alpha^{min}, \alpha^{max}] \) limits. Note that the Jacobian of the right hand side of equations (31a), (31b) is

\[
\begin{bmatrix}
c_\alpha - \left(F_{thr} s_\alpha + \frac{\partial D(p, \alpha, V_a)}{\partial s_\alpha}\right) \\
F_{thr} c_\alpha + \frac{\partial D(p, \alpha, V_a)}{\partial c_\alpha} \end{bmatrix}
\]

and its determinant

\[
|J| = F_{thr} + \frac{\partial L(p, \alpha, V_a)}{\partial \alpha} c_\alpha + \frac{\partial D(p, \alpha, V_a)}{\partial \alpha} s_\alpha > 0
\]

assuring that inversion is possible at all times. The \( \phi \) value required to follow the lateral dynamics, it is obtained directly from (31c) and is also saturated within the limits \( [\phi^{min}, \phi^{max}] \).

Now, to assure that the 4D trajectory error tends to zero after some perturbation, the reference values for \( V_a, \gamma, \) and \( \psi \) must be adapted. The proposed adaptive scheme is the following:

\[
\begin{align*}
V_a^d &= V_a + \delta V_a \\
\gamma^d &= \gamma + \delta \gamma \\
\psi^d &= \psi + \delta \psi
\end{align*}
\]

where

\[
\begin{align*}
\delta V_a &= \frac{1}{\tau_x} (x_R - x_E) \\
\delta \gamma &= \frac{1}{\tau_\gamma} (y_R - z_E) \\
\delta \psi &= \frac{1}{\tau_\psi} (y_R - y_E)
\end{align*}
\]

and \( \tau_x, \tau_\gamma, \tau_\psi \) are time-constants such that: \( \tau_x >> \tau_\gamma; \tau_y >> \psi; \tau_z >> \tau_\gamma \).

The term \( \delta \psi \) is depicted in figure 1.
5. SIMULATION RESULTS

A six degree of freedom model for a wide body transportation aircraft has been developed in Matlab to provide simulation results with the adopted control law. The aircraft parameters were chosen close enough to emulate a Boeing 737-200, flying in an International Standard Atmosphere model with uniform gravity. The time response for engines is 4 s, and for actuators used to move the control surfaces is 50 ms. To obtain the values of the aerodynamic coefficients, two-layer feed-forward neural networks with sigmoid hidden neurons and linear output neurons were trained using data bases obtained from the United States Air Force (USAF) Stability and Control Digital DATCOM (Data Compendium) PDAS (a). This is a Public Domain Aeronautical Software that computes the static stability, control and dynamic derivative characteristics of fixed-wing aircrafts using the methods contained in the USAF Stability and Control DATCOM. Furthermore, the data sets were compared and refined using the JSBSim open source Flight Dynamics Models (FDM) PDAS (b), used in several open source simulators and also employed to drive the motion-base research simulators of many universities worldwide. The training, validation and test sets were 70, 15 and 15 percent respectively of the available data for each aerodynamic coefficient. The training algorithm used was Bayesian Regularization and the number of hidden neurons was selected by trial and error trying to improve the performance as much as possible.

When the control surfaces are jammed at zero degrees, and enough constant Thrust to hold a longitudinal flight is set, the behaviour of the aircraft is a stable, with decreasing phugoid mode with a 100s period. In order to test the proposed 4D trajectory tracking approach for the longitudinal motion, a 100m amplitude sinusoidal reference trajectory for $z_R(t)$ is proposed. The results are depicted in figure 2. The required thrust is computed and affected by the first order response of its actuator (eq. (23)). Also, the required AoA is computed and then used to calculate a desired pitch angle (using the Flight Path Angle). The commanded $(x_R(t), z_R(t))$ trajectory is followed with a small delay.

For the lateral motion, another 100m sinusoidal reference trajectory for $y_R(t)$ is proposed, the response of the controller is depicted in figure 3. For practical purposes, the initial heading of the airplane is zero degrees.

6. CONCLUSION

This paper presents a new approach to perform 4D trajectory tracking for transportation aircraft. The proposed approach avoid the numerical difficulties encountered by direct non linear inverse control approaches by limiting inversion in the guidance dynamics. Contrarily to the energy-based methods, frequency decoupling is preserved, avoiding blind setting of control parameters. The proposed
approach has been tested through numerical simulation of the flight of a large transportation aircraft, providing satisfying results. Future studies should consider the effect of introducing wind estimates and the establishment of 4D trajectory tracking performance as a parameter in order to improve air traffic management.

7. APPENDIX A. FAST DYNAMICS INVERSION

The jerk vector of the angular velocities is obtained by differentiating one more time equation (10) and including (22). Also, taking into account the aerodynamic moment coefficients dynamics due to their close relation with \( Ra = [\alpha, \beta, V_a] \) by differentiating (9), wind information can be included in the model. The control surfaces moment coefficients dynamics are neglected. Yielding:

\[
\begin{bmatrix}
\dot{p} \\
\dot{q} \\
\dot{r}
\end{bmatrix} = \frac{1}{2} \rho ST^{-1} \left( V_a^2 C_a \xi \begin{bmatrix}
\delta a_{ail} - \delta a_{ele} \\
\delta e_{ele} - \delta e_{ele} \\
\delta r_{rud} - \delta r_{rud}
\end{bmatrix} + V_a^2 C_a \dot{Ra}
\right)
+ 2 V_a \dot{V}_a \left( \begin{bmatrix}
bC_{i1} & 0 & 0 \\
0 & bC_{e1} & 0 \\
bC_{n1} & 0 & bC_{n1}
\end{bmatrix} + C_d \begin{bmatrix}
\delta a_{ail} \\
\delta e_{ele} \\
\delta r_{rud}
\end{bmatrix} \right)
+ I^{-1} \left( \frac{1}{\rho S V_a C_a - I_n} \right) \dot{\Omega}
\]

(36)

where

\[
I_n = \begin{bmatrix}
-E_q & (C - B)r - E_p & (C - B)q \\
(A - C)r + 2 E_p & (A - C)p - 2 E_r & 0 \\
(B - A)q & (B - A)p + E_r & E_q
\end{bmatrix}
\]

\[
\xi = \begin{bmatrix}
1 \\
0 \\
0 \\
0 \\
0 \\
0 \\
0 \\
0 \\
1
\end{bmatrix}
\]

\[
C_c = \begin{bmatrix}
0 & bC_{i3} - \frac{\sigma^2}{\rho S} (C_{i3} p + C_{i3} r) & 0 \\
0 & -\frac{\sigma^2}{\rho S} (C_{e3} q) & 0 \\
0 & 0 & -\frac{\sigma^2}{\rho S} (C_{n3} p + C_{n3} r)
\end{bmatrix}
\]

\[
C_l = \begin{bmatrix}
\delta^2 C_{i3} & 0 & \delta^2 C_{i3} \\
0 & \delta^2 C_{e3} & 0 \\
0 & 0 & \delta^2 C_{n3}
\end{bmatrix}
\]

The vector \( \dot{Ra} \) can be taken from (12) and (18), and the vector \( \dot{\Omega} \) from (10). Consequently a NLI leads to an attitude control input denoted by:

\[
\begin{bmatrix}
\delta a_{ail} \\
\delta e_{ele} \\
\delta r_{rud}
\end{bmatrix} = \frac{1}{V_a^2} e^{-C_{i3} t} \begin{bmatrix}
2 \tau_p \\
2 \tau_q \\
2 \tau_r
\end{bmatrix} \begin{bmatrix}
\frac{1}{\rho S} & 0 & 0 \\
0 & \frac{1}{\rho S} & 0 \\
0 & 0 & \frac{1}{\rho S}
\end{bmatrix} \frac{1}{\rho S} \left( \frac{1}{4} \rho S V_a C_a - I_n \right) \dot{\Omega}
\]

\[
-2 V_a \dot{V}_a \left( \begin{bmatrix}
bC_{i1} & 0 & 0 \\
0 & bC_{e1} & 0 \\
bC_{n1} & 0 & bC_{n1}
\end{bmatrix} + C_d \begin{bmatrix}
\delta a_{ail} \\
\delta e_{ele} \\
\delta r_{rud}
\end{bmatrix} - V_a^2 C_a \dot{Ra} \right) + I^{-1} \frac{1}{\rho S V_a C_a - I_n} \dot{\Omega} \]

(37)

where the wind effects appear in the terms involving \( \dot{Ra} \), and:

\[
\begin{align*}
\tau_p &= -k_1 (p - \dot{p}^*) - k_2 (\dot{p} - \ddot{p}^*) + \dddot{p}^* \\
\tau_q &= -k_3 (q - \dot{q}^*) - k_4 (\dot{q} - \ddot{q}^*) + \dddot{q}^* \\
\tau_r &= -k_5 (r - \dot{r}^*) - k_6 (\dot{r} - \ddot{r}^*) + \dddot{r}^*
\end{align*}
\]

(38)

For \( k_i > 0 \ (i = 1,...,6) \) are gains chosen in order to assure asymptotical convergence of the variables to their desired values. The feasibility of this approach depends on the singularity of \( C_a \), the matrix involving the aerodynamic coefficients (assumed to be known thanks to experimental data, airflow simulations, or any other method) due to the control surfaces, aspect that can be handled.

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