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Quasi-Continuous Sliding Mode Flight Control for a Fixed-Wing UAV

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***Abstract*— This paper addresses the problem of controlling the attitude and the airspeed of a fixed wing Unmanned Aerial Vehicle (UAV). A full dynamical model including aerodynamic model that represents the behavior of the UAV is obtained. Furthermore, quasi-continuous sliding mode approach is considered thanks to its attractive features, such as robustness and finite-time convergence. In order to implement such controller a robust differentiator is required to estimate the time derivatives of the sliding surface. Additionally, this control approach is capable of controlling the UAV under external disturbances and coupled dynamics. Simulation results are presented to illustrate its performance.**

Keywords: Robust control, UAVs, Flight control.

I. INTRODUCTION

Unmanned Aerial Vehicles (UAVs) are defined as those vehicles without human crew, where the flight control is performed by an automatic pilot. The UAVs have shown benefits in multiple civil applications as traffic monitoring, surveillance, wheatear research, mapping, inspection of power electric lines, oil pipelines, etc. (Valavanis, 2007). The fixed-wing classification, in contrast to rotary wing or flapping wings, consist on the typical aircraft design for manned operations. The flight performance of this aircraft is affected by the aerodynamic parameters as well as physical external conditions like altitude, wind, payload variation and limited resources (Austin, 2010). Since the fixed-wing UAV dynamical model is nonlinear and strongly coupled. In addition external disturbances like wind gusts. Regarding the attitude and airspeed control design this task can becoming a big challenge, since the control strategies must be robust against model uncertainties and external perturbations.

In order to tackle the flight control problem, several approaches have been proposed. For instance, based on linear approximations, a linearization in an equilibrium point has been proposed for trajectory tracking (Stengel, 2004). However, this methodology lacks of robustness as the exact cancellation of nonlinearities is not ensured. Several nonlinear control techniques have been proposed to flight control. For instance, those based on feedback linearization techniques, nonlinear dynamic inversion (Enns et al., 1994), adaptive backstepping. In (Zhang et al., 2012), an adaptive backstepping scheme based on invariant manifolds has been designed to track angle of attack, sideslip angle and roll angle. Nonetheless, this control is designed based on a linearized aerodynamic model. Furthermore, there are techniques based on invariant manifolds (Karagiannis et al., 2010), where an energy function is used to design a controller robust in presence of aerodynamic moments with unknown coefficients. However, this energy function is not easily established, requiring an important analytical effort. Moreover, this controller needs exact measurements, limiting its implementation.

On the other hand, Active Disturbances Rejection Control (ADRC) approach offers a robust controller that does not require an exact model. In (Hua et al., 2011), ADRC with nonlinear feedback is proposed for designing attitude and airspeed controllers under wind turbulence conditions.

Sliding mode control techniques had attracted the attention of many research groups. This technique allows to design robust control laws that are insensitive to uncertainties (Levant, 2008). Regarding the stability properties of the closed-loop system, the above methods can only ensure asymptotically stability properties. However, using the sliding mode techniques the robustness and insensitive to uncertainties can be guarantee, and the converges in finite time is ensured.

The high order sliding mode approach is an interesting alternative to overcome such problems, thanks to its properties of robustness against uncertainties in the model and external disturbances.

A. Contribution

In this work, an extension of (Castañeda et al., 2011) is presented. In that previous paper a controller is implemented with some assumptions in order to reduce the fixed-wing UAV model. However, results are constrained to steady flight. Now, using the same quasi-continuous sliding mode approach, a full attitude and airspeed control of a fixed wing UAV is designed for tracking of desired trajectories. Furthermore, to implement such controller a robust differentiator is designed to estimate time derivatives of sliding surface. This control scheme is robust against external perturbations and model uncertainties.

B. Paper structure

This paper is organized as follows: In section 2, the mathematical model of the fixed wing UAV is presented. In section 3, a quasi-continuous high order sliding mode control design is introduced. In section 4, attitude and air-speed controls for tracking desired trajectories are proposed. Simulation results are given to show the performance of the proposed controller in section 5. Finally, some conclusions are given.

II. Mathematical Model of UAV

Attitude of a rigid body moving in space can be expressed with the *roll-pitch-yaw* convention with the use of 3 Euler angles $(\phi, \theta, \psi) \in [-\pi, \pi]$, as in the Figure 1. The control of a fixed-wing UAV is represented by three control surfaces: aileron, elevator and rudders; and the thrust generated by an engine. Thus, the variables describing the state of the system are the inertial position of the aircraft $\boldsymbol{d} = [x, y, z]^T \in \mathbb{R}^3$, the attitude described by the set of the Euler angles $\boldsymbol{\Theta} = [\phi, \theta, \psi]^T$, the non-inertial expression of the linear velocity (body fixed frame coordinates) $\mathbf{v} = [u, v, w]^T \in \mathbb{R}^3$ and the non-inertial expression of the angular velocity $\boldsymbol{\omega} = [p, q, r]^T \in \mathbb{R}^3$ (for complete derivation see (Castañeda et al., 2011)).

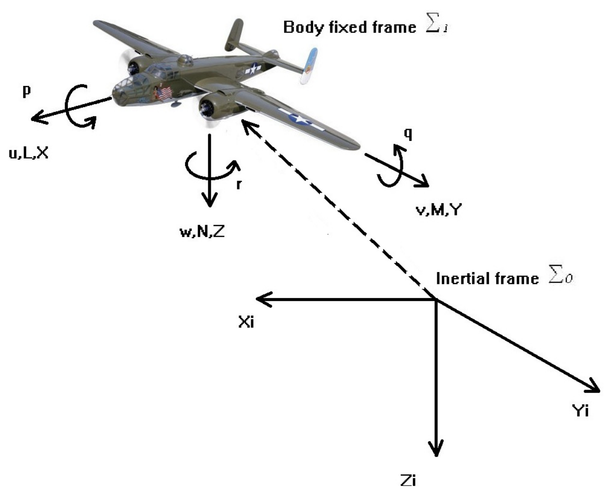

Figure 1. Referential frames.

Using Newton-Euler formulation, a full 6 degree of freedom aircraft model is given by (Stengel, 2004; Stevens et al., 2003)

$$\begin{aligned} \dot{\boldsymbol{d}} &= R(\boldsymbol{\Theta})\mathbf{v} && (1)\\ \dot{\boldsymbol{\Theta}} &= \boldsymbol{W}^{-1}(\boldsymbol{\Theta})\boldsymbol{\omega} && (2)\\ \boldsymbol{f} + \boldsymbol{T} &= m(\dot{\mathbf{v}} + \boldsymbol{\omega} \times \mathbf{v}) - mR^T(\boldsymbol{\Theta})\boldsymbol{g} && (3)\\ \boldsymbol{n} &= I\dot{\boldsymbol{\omega}} + \boldsymbol{\omega} \times I\boldsymbol{\omega} && (4) \end{aligned}$$

where the Rotation matrix $R(\boldsymbol{\Theta}) \in SO(3)$ transform body axis coordinates to inertial frame coordinates, and the operator $\boldsymbol{W}(\boldsymbol{\Theta}) \in \mathbb{R}^{3\times 3}$ maps the time derivative of the Euler angles set to the non-inertial expression of the angular velocity. Both matrices are given explicitly by

$$R(\boldsymbol{\Theta}) = \begin{bmatrix} c_\psi c_\theta & -s_\psi s_\phi + c_\psi s_\theta s_\phi & s_\psi s_\phi + c_\psi s_\theta c_\phi \\ s_\psi c_\theta & c_\psi c_\phi + s_\psi s_\theta s_\phi & -c_\psi s_\phi + s_\psi s_\theta c_\phi \\ -s_\theta & c_\theta s_\phi & c_\theta c_\phi \end{bmatrix}$$

$$\boldsymbol{W}(\boldsymbol{\Theta}) = \begin{bmatrix} 1 & 0 & -s_\psi \\ 0 & c_\psi & c_\theta s_\psi \\ 0 & -s_\theta & c_\theta c_\psi \end{bmatrix}$$

where s_x and c_y stand for the $sin(x)$ and $cos(y)$ functions with their corresponding arguments. The extrinsic active forces are given by the propeller thrust which for our study case are given uniquely along x body axis: $\boldsymbol{T} = [T_x, 0, 0]^T$ The gravity vector $\boldsymbol{g} = [0, 0, g_z]^T$ expresses the gravity acceleration in inertial coordinates while the inertia tensor $I \in \mathbb{R}^{3\times 3}$ (with x-z plane of symmetry) is constant when expressed in the non-inertial body fixed frame:

$$I = \begin{bmatrix} I_{xx} & 0 & I_{xz} \\ 0 & I_{yy} & 0 \\ I_{zx} & 0 & I_{zz} \end{bmatrix}$$

Finally, the fixed-wing UAV aerodynamics are represented by non-inertial expressions of the force vector $\boldsymbol{f} = [F_X, F_Y, F_Z]^T \in \mathbb{R}^3$ and the torque vector $\boldsymbol{n} = [F_L, F_M, F_N]^T \in \mathbb{R}^3$.

A. Aerodynamics

The aerodynamics forces and torques in (3)-(4) can be calculated by means of aerodynamic coefficients (Stevens et al., 2003) where,

$$\boldsymbol{f} = \bar{q}S\chi(\alpha, \beta)^{-1}[-C_D, C_Y, -C_L]^T$$

$$\boldsymbol{n} = \bar{q}S\,[b\,C_l, \bar{c}\,C_M, b\,C_n]^T$$

where $\alpha = \arctan(\frac{w}{u})$ and $\beta = \arcsin(\frac{u}{V})$ are respectively the angle of attack and the sideslip angle. The transformation matrix $\chi(\alpha, \beta) \in SO(3)$ maps body fixed frame coordinates Σ_1 to a virtual wind frame Σ_w, defined along the relative velocity of the aircraft; and given as (Stevens et al., 2003):

$$\chi(\alpha, \beta) = \begin{bmatrix} c_\alpha c_\beta & s_\beta & s_\alpha c_\beta \\ -c_\alpha s_\beta & c_\beta & -s_\alpha s_\beta \\ -s_\alpha & 0 & c_\alpha \end{bmatrix}$$

Note that the coefficients C_L and C_D are indeed the drag and lift coefficients for the airplane. The dynamic pressure $\bar{q} = \frac{1}{2}\rho V^2$, is function of the relative airspeed magnitude $V = \sqrt{u^2 + v^2 + w^2}$. Besides, wing surface area S, the wingspan b, the mean aerodynamic chord $\bar{c}$ and the air density ρ are considered as constant parameters, and the dimensionless coefficients in the force/moment expressions can be decomposed in the following set of equations (Stevens et al., 2003).

$$
\begin{aligned}
C_L &= c_{L0} + C_{L\alpha}\alpha + c_{L\delta e}\delta e + \frac{\bar{c}}{2V}(c_{L\dot{\alpha}}\dot{\alpha} + c_{Lq}q) \\
C_D &= c_{D0} + \frac{(c_L - c_{L0})^2}{\pi e AR} + c_{D\delta e}\delta e + c_{D\delta a}\delta a + c_{D\delta r}\delta r \\
C_Y &= c_{y\beta}\beta + (c_{yp}p + C_{yr}r)\frac{b}{2V} + c_{y\delta a}\delta a + c_{y\delta r}\delta r \\
C_l &= c_{l\beta}\beta + (c_{lp}p + c_{lr}r)\frac{b}{2V} + c_{l\delta a}\delta a + c_{l\delta r}\delta r \\
C_M &= c_{m0} + c_{m\alpha}\alpha + c_{m\delta e}\delta e + (c_{mq}q + c_{m\dot{\alpha}}\dot{\alpha})\frac{\bar{c}}{2V} \\
C_n &= c_{n\beta}\beta + (c_{np}p + c_{nr}r)\frac{b}{2V} + c_{n\delta a}\delta a + c_{n\delta r}\delta r
\end{aligned}
$$

where, δe, δa and δr represents the moving surfaces: elevator, ailerons and rudder respectively. The above expressions use also the dimensionless numbers: Oswald´s efficient number e, the Mach number M (due to velocity range of a scale airplane, this factor is neglected for this UAV), and the aspect ratio $AR = b^2/S$.

Tornado software (see (Melin, 2000) for more details) has been used to identify the coefficients using the vortex lattice method.

III. Quasi-continuous high order sliding mode

In this section, we introduce the results related to high order sliding mode (Levant, 2003) which will be considered to attitude and airspeed control of a fixed-wing UAV.

Consider the attitude fixed wing UAV system belonging to a class of nonlinear systems represented by

$$\dot{X} = f(X) + g(X)u, \qquad X(t_0) = X_0, \tag{5}$$

where $t_0 \geq 0$, $X \in B_x \subset \mathbb{R}^{3n}$ is the *state vector*, n is the number of subsystems in the attitude of fixed wing UAV, i.e. roll, pitch and yaw motions, $u \in \mathbb{R}^n$ is the *control input vector*, the field vectors f and g are assumed to be bounded with their components being smooth function of X. B_x denotes a closed and bounded subset, centered at the origin. In order to design a finite-time convergent controller some conditions are required. Since the relative degree r of the system is assumed to be constant and known, it implies that the control explicitly appears first in r-th total time derivative of σ and

$$\sigma^{(r)} = h(t,x) + m(t,x)u \tag{6}$$

where $h(t,x) = \sigma^{(r)}|_{u=0}$, $g(t,x) = (\partial/\partial u)\sigma(r) \neq 0$. It is supposed that for some $K_m, K_M, C > 0$

$$0 < K_m \leq \frac{\partial}{\partial u}\sigma^{(r)} \leq K_M \qquad |\sigma^{(r)}|_{u=0} \leq C \tag{7}$$

which is always true at least locally. From (6) and (7), we have

$$\sigma^{(r)} \in [-C, C] + [K_m, K_M]u. \tag{8}$$

The closed differential inclusion is understood here in the Filippov sense, which means that the right-hand vector set is enlarged in a special way in order to satisfy certain and semi-continuity conditions. The inclusion only requires to know the constants r, C, K_m and K_M of system (5). These conditions allow to give a solution to this control problem. In order to design a high order sliding mode control for the system. We consider the following n-dimensional nonlinear surface defined by

$$\sigma(X - X_d) = 0 \tag{9}$$

where X_d is a desired equilibrium point of the system and each function $\sigma_i : \mathbb{R}^3 \to \mathbb{R}, i = 1, ..., n$, is a C^1 function such that $\sigma_i(0) = 0$. Then, provided that successive total time derivatives σ, $\dot{\sigma}$,..., $\sigma^{(r-1)}$ are continuous functions of the closed-system state-space variables, and

$$\sigma = \dot{\sigma} = \ldots = \sigma^{(r-1)} = 0 \tag{10}$$

is a nonempty integral set, whose motion of called r-*sliding mode*. Under the above considerations the controller is designed as follows. Let be $i = 0, ..., r-1$. Then,

$$\varphi_{0,r} = \sigma \quad N_{0,r} = |\sigma|, \quad \Psi_{0,r} = \varphi_{0,r}/N_{0,r} = sign\{\sigma\},$$

and

$$
\begin{aligned}
\varphi_{i,r} &= \sigma^{(i)} + \beta_i N_{i-1,r}^{(r-i+1)}\Psi_{i-1,r} && (11) \\
N_{i,r} &= |\sigma^{(i)}| + \beta_i N_{i-1,r}^{(r-i)/(r-i+1)} && (12) \\
\Psi_{i,r} &= \varphi_{i,r}/N_{i,r} && (13)
\end{aligned}
$$

where $\beta_i, ..., \beta_{r-1}$ are positives numbers, provided $\beta_i, ..., \beta_{r-1}$, $\alpha > 0$ are chosen sufficiently large in the list order, the controller

$$u = -\alpha\Psi_{r-1,r}(\sigma, \dot{\sigma}, ..., \sigma^{r-1}) \tag{14}$$

is r-sliding homogeneous and provided for the finite-time stability, $\sigma = 0$. Each choice of parameters $\beta_1, ..., \beta_{r-1}$ determines a controller family applicable to all systems (6) of the relative degree r.

A. Output Feedback Controller

In order to implement the control (14), it is necessary to know the real time exact calculation or direct measurement of $(\sigma, \dot{\sigma}, ..., \sigma^{(r-1)})$. However, the only measurable signals in the system are the angular positions $\Theta = [\phi, \theta, \psi]^T$. Combining controller (14) and the homogeneous differentiator (Levant, 2001), we have

$$
\begin{aligned}
u &= -\alpha\Psi_{r-1,r}(z_0, z_1, \ldots, z_{n-1}) \\
\dot{z}_0 &= -\lambda_r L^{\frac{1}{r}}|z_0 - \sigma|^{\frac{r-1}{r}} sgn(z_0 - \sigma) + z_1 \\
\dot{z}_k &= -\lambda_{r-k} L^{\frac{1}{r-k}}|z_k - z_{k-1}|^{\frac{r-k-1}{r-k}} sgn(z_k - z_{k-1}) \\
&+ \ z_{k+1} \\
\dot{z}_{r-1} &= -\lambda_1 L sgn(z_{r-1} - z_{r-2})
\end{aligned} \tag{15}
$$

for k = 1,..., r-2; where $z0, z1, ..., zk$ are estimates of the k-th derivatives of σ.

Remark 1: Finite time convergence of the observer allows to design the observer and the control law separately, i.e., the separation principle is satisfied. If the applied controller is known to stabilize the process, one way is to choose the differentiator dynamics fast enough to provide for the exact evaluation of the σ, $\dot{\sigma}$ and $\ddot{\sigma}$ before leaving some preliminarily chosen area, where the stabilization is assured.

IV. Attitude and airspeed controllers

In this section, attitude and airspeed controllers used to drive the fixed wing UAV flight are presented. The physical elements to control the UAV are the control surfaces. The **elevator** produces an angle δe which in turns generates a pitching motion, **rudders** provides an angle δr which in rotates results a heading motion; and **ailerons** generates an angle δa which in spin gives a rolling motion. Furthermore, **thrust** produces an acceleration in the fixed wing UAV along x-axis.

A. Attitude controller

Since, rotational motion is faster than translational motion in the fixed wing UAV. Therefore, the translational time derivatives of position and velocity can be neglected with respect to rotational dynamic (Recasens et al., 2005). Then, taking the derivative with respect to the time of (2) we have the dynamical inertial attitude motion described in state-space representation by

$$\begin{aligned}\dot{X}_1 &= X_2\\ \dot{X}_2 &= (\boldsymbol{I}\boldsymbol{W}(X_1))^{-1}[-\boldsymbol{I}N-(\boldsymbol{W}(X_1)X_2\times\boldsymbol{I}\boldsymbol{W}(X_1)X_2)\\ &+ \bar{q}S\boldsymbol{C}]+(\boldsymbol{I}\boldsymbol{W}(X_1))^{-1}\boldsymbol{B}\boldsymbol{u}+\boldsymbol{\Phi}(t) \qquad (16)\end{aligned}$$

where $\boldsymbol{\Phi}(t) \in \mathbb{R}^3$ is the disturbance, $X_1 = \boldsymbol{\Theta} = [\phi, \theta, \psi]^T$, $\boldsymbol{C} = [bC_l, cC_m, bC_c]^T$,

$$\boldsymbol{B} = \begin{bmatrix} bC_{l\delta a} & 0 & 0\\ 0 & cC_{m\delta e} & 0\\ 0 & 0 & bC_{n\delta r}\end{bmatrix},$$

$$N = \frac{d}{dt}\boldsymbol{W}(\boldsymbol{\Theta})\dot{\boldsymbol{\Theta}} = \begin{bmatrix} -C_\theta\dot{\theta}\dot{\psi}\\ -S_\theta\dot{\phi}\dot{\psi}+C_\phi C_\theta\dot{\phi}\dot{\psi}-S_\phi S_\theta\dot{\theta}\dot{\psi}\\ -C_\theta\dot{\phi}\dot{\psi}-S_\phi C_\theta\dot{\phi}\dot{\psi}+C_\phi S_\theta\dot{\theta}\dot{\psi}\end{bmatrix},$$

and $u = [\delta a, \delta e, \delta r]^T$ is the control input.

Then, we can define

$$\begin{aligned} f(X) &= (\boldsymbol{I}\boldsymbol{W}(X_1))^{-1}(-\boldsymbol{I}N-(\boldsymbol{W}(X_1)X_2\times\boldsymbol{I}\boldsymbol{W}(X_1)X_2)\\ &+ \bar{q}S\boldsymbol{C})+\boldsymbol{\Phi}(t)\\ G(X) &= (\boldsymbol{I}\boldsymbol{W}(X_1))^{-1}\boldsymbol{B}\end{aligned}$$

Consider the second order SISO quasi-continuous high order sliding mode to drive each attitude angle, given by following equation

$$u_{qsi} = -\kappa_i\frac{\dot{\sigma}_i+\varsigma_i \mid \sigma_i \mid^{1/2} sgn(\sigma_i)}{\mid \dot{\sigma}_i \mid + \varsigma_i \mid \sigma_i \mid^{1/2}} \qquad (17)$$

for $i = \phi, \theta, \psi$, where $u = [\delta a, \delta e, \delta r]^T = [u_{qs\phi}, u_{qs\theta}, u_{qs\psi}]^T$. Furthermore, to estimate $\dot{\sigma}$ from this system, we use the first order differentiator, given by

$$\begin{aligned}\dot{z}_{0,i} &= v_{0,i}\\ v_{0,i} &= -\lambda_{1,i} \mid z_{0,i}-\sigma_i \mid^{1/2} sgn(z_{0,i}-\sigma_i)+z_{1,i}\\ \dot{z}_{1,i} &= sgn(z_{0,i}-\sigma_i) \qquad (18)\end{aligned}$$

where $z_{0,i}$ and $z_{1,i}$ are the estimations of σ_i and $\dot{\sigma_i}$, for $i = \phi, \theta, \psi$; respectively. On the other hand, the sliding surface is defined as

$$\sigma = [\sigma_1, \sigma_2, \sigma_3]^T = [X_1 - X_1(d)] = [\phi-\phi_d, \theta-\theta_d, \psi-\psi_d]^T \qquad (19)$$

which gives, from (16)

$$\ddot{\sigma} = \frac{d}{dt}[\dot{X}_1 - \dot{X}_1(d)] = \Big(f(X) - \ddot{X}_1(d)\Big) + G(X)u + \boldsymbol{\Phi}(t) \qquad (20)$$

Obviously, $\sigma = 0$ describes the required system dynamic. The controller is designed to ensure the system trajectories reach an arbitrarily small vicinity of the origin in finite time and remain there in spite of bounded disturbances. The tuning for this controllers is given by the following gains $\kappa_{1\phi} = 0.06$, $\kappa_{1\theta} = 0.08$ and $\kappa_{1\psi} = 0.6$, $\varsigma_\phi = \varsigma_\theta = \varsigma_\psi = 0.001$, $\lambda_{1\phi} = \lambda_{1\theta} = \lambda_{1\psi} = 5$ and $L_\phi = L_\theta = L_\psi = 9$ for roll, pitch and yaw controllers respectively. Note that according with the definition of $\boldsymbol{W}(X_1)$ this matrix has a singularity on $\theta = \pm\pi/2$. However, the mapping between local and inertial velocities is always kept. Therefore, this work not considers flight with the aforementioned values.

Proposition 1: Consider the UAV attitude dynamics given by (16) in closed loop with second order quasi continuous sliding mode controllers (17) together a robust exact differentiators (18) via output feedback. Then, attitude $X(t)$ tracks attitude desired $X_d(t)$ in finite time, under internal and external uncertainties.

B. Airspeed Controller

Now, to command the aircraft velocity V by means of trust T_x, an airspeed controller is designed. From (3) and using a hybrid system of coordinates wind axes frame and body axes, the airspeed is given by (see (Stengel, 2004)):

$$\dot{V} = \frac{D}{m} - gsin(\theta-\alpha) - \frac{cos(\alpha)cos(\beta)}{m}u \qquad (21)$$

where, D drag, α the angle of attack, β sideslip angle, m aircraft mass, g gravity, and $u = T_x$ is the input control. Then, in order to design a controller to drive the airspeed, we define $\sigma = V - V_d$ as the sliding surface. Note that a first high order sliding mode controller is enough to control the airspeed. However, in order to reduce the chattering, it is virtually increased the relative degree r of the airspeed system. Then, a second order quasi-continuous controller is applied using the single second order controller (17) and the differentiator (18) respectively. The gains of this controller are $\kappa_V = 8$, $\varsigma_V = 0.001$, $\lambda_{1V} = 5$ and $L_V = 1$.

Proposition 2: Consider system (3)-(21) in closed-loop with quasi continuous controller (17) using the estimations of the robust differentiator (18). Then, the airspeed $V(t)$

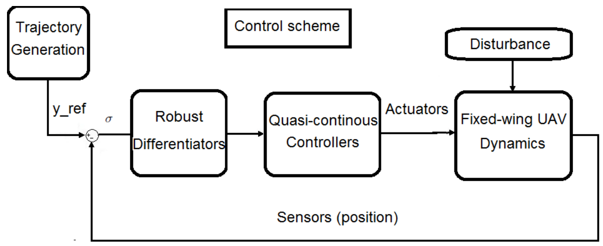

Figure 2. Simulation diagram

track desired references $V_d(t)$ in finite-time via output feedback, under parametric uncertainties and external disturbance.

Remark 2: The separation of the rotational and translational dynamics implies that it is neglected the coupled effect of rotational dynamics into translation motion. Regarding the control, this effect represents a zero dynamics whose stability depends of the shape of the fixed-wing and the distribution of the forces into the aircraft. In this work we assume that the Mitchell B-25 has stable zero dynamics.

V. Results and discussion

Simulations results are obtained using the full nonlinear system described by (1)-(4). This model takes into consideration the complexity of the aerodynamic forces/torques. Furthermore, the controllers and observers were developed in Matlab/Simulink with a sampling time of $0.001s$, using the Runge-Kutta solver. In addition, the Figure 2 illustrates the control scheme. Finally, disturbances represented by wind external currents with magnitude $x = 4m/s$ at $t = 30s$, $y = 4m/s$ at $70s$ and $z = 1m/s$ at $t = 105s$ have been applied in order to verifying robustness of the proposed approach.

Simulations focus on test of the quasi-continuous high order sliding mode control on a UAV to track a desired trajectory under uncertainties and external perturbations. Furthermore, the proposed control is compared with Active Disturbance Rejection Control, this technique is based on the extended state observer, which estimate and compensates lumped the unknown dynamics and external disturbances. Then, the closed loop is completed by means of a PD controller (see for more details (Han, 2008)).

Figure 1 shows the UAV, which represents a scale model of the B-25 Mitchell, where the main parameters are given in Table I. Figure 3 shows desired trajectory, which consist in a circle followed by an oval with changes of velocity from $15m/s$ to $20m/s$ of airspeed and altitude variations.

Attitude responses are plotted in Figure 4, where a good tracking of roll, pitch and yaw angles is achieved by both controllers. Tracking error is illustrated in Figure 6, where high order sliding mode controller has been shown more accuracy. This signals represents a measurement of performance for tracking. Control signals are plotted in Figure 5. This signals represents the deflection angle on the physical control surfaces in the aircraft, the position of this

TABLE I

Parameters of the B-25 Mitchell UAV.

Parameter	Value	Unit
Weight	8	Kg
Span	2.05	m
Wing surface	0.55	m^2
Mean aerodynamic chord	0.28	m
Length	1.6	m
Inertia moment I_{xx}	0.5528	Kgm^2
Inertia moment I_{yy}	0.6335	Kgm^2
Inertia moment I_{zz}	1.0783	Kgm^2
Inertia moment I_{xz}	0.0015	Kgm^2

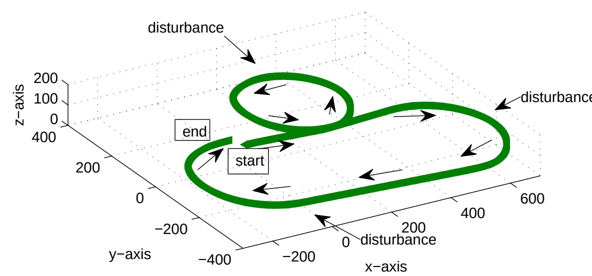

Figure 3. Desired trajectory

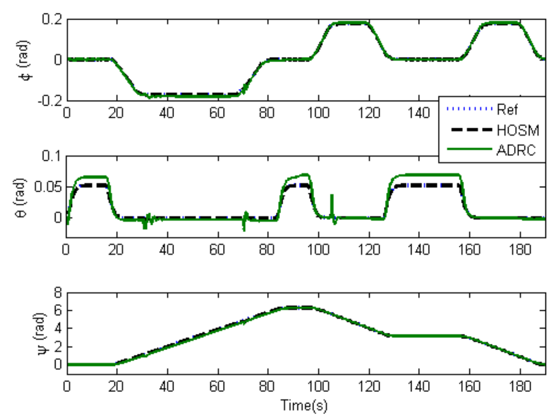

Figure 4. Attitude (rad).

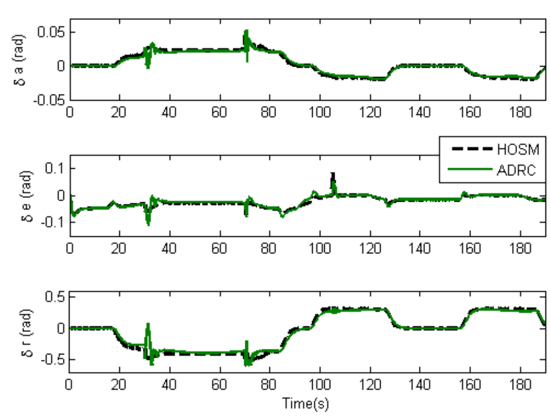

Figure 5. Attitude control (rad).

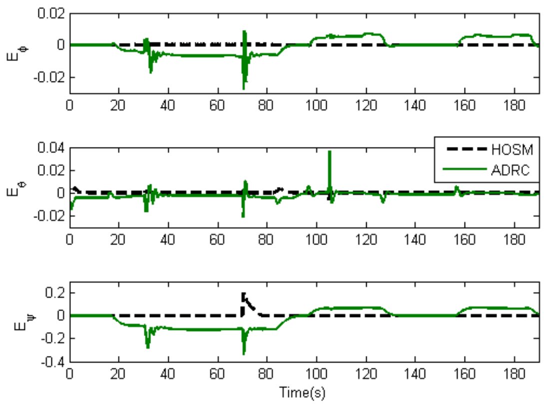

Figure 6. Tracking error.

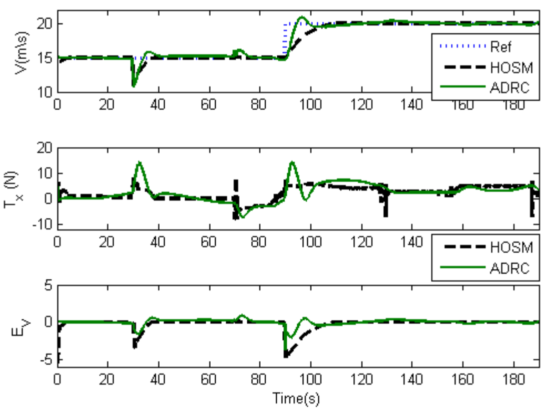

Figure 7. Airspeed.

angles determines the fixed wing attitude. The responses shows a small transients when disturbances appear at $t = [30, 70, 105]s$, where quasi continuous controller presents better behavior than ADRC.

Airspeed results are shown in Figure 7, in the first sub graphic it is possible to see the airspeed convergence to desired signal, at middle sub graphic, it is the thrust that must be generated by the propeller, where there is a constant K_f of proportion between the provided thrust and voltage applied to motors, this constant depends of the propeller. Finally, in the bottom sub graphic, the tracking error for airspeed is illustrated, where high order sliding mode controller is better.

The quasi-continuous high order sliding mode control has been implemented to drive a fixed-wing UAV, the tuning for this application has been a difficult task. Since this fact represents the appropriate reduction of chattering, for this reason it cannot see this phenomenon in control signals (see Figure 5). With respect to flight control, the chattering effect can be dangerous for the stability of the aircraft. Therefore, the tuning has been done by means of iterations to achieve smooth responses. On the other hand, there exist a compromise between the sampling time and the obtained results, i.e., when sample period is very small, the results are better. Regarding the ADRC, this controller shows good results and it does not needs full model information. Therefore, it represents a robust controller. Nonetheless, by the fact that it calculates disturbances and unknown dynamics into a extended state, this result in a controller very sensitive to the noisy and increase the difficulty for tuning.

VI. Conclusion

In this work, an attitude and airspeed control for a Fixed Wing Unmanned Aerial Vehicle, has been proposed to track a desired reference, this control is based on quasi-continuous high order sliding mode control techniques. In order to implement this controller a robust differentiator is designed to estimate time derivatives of the sliding surface. Furthermore, a comparative study has been done, by showing the performance Active Disturbance Rejection Control and the proposed control scheme. The simulation results have been shown that the presented controller is capable to achieve the tracking objective against parametric uncertainties and external disturbances.

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