Automatic Landing System for Civil Unmanned Aerial System

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Abstract

Taking ground effect into account a longitudinal automatic landing system is designed. Such a system will be tested and implemented on board by using the Preceptor N3 Ultrapup aircraft which is used as technological demonstrator of new control navigation and guidance algorithms in the context of the "Research Project of National Interest" (PRIN 2008) by the Universities of Bologna, Palermo, Ferrara and the Second University of Naples.

A general mathematical model of the studied aircraft has been built to obtain non–linear analytical equations for aerodynamic coefficients both Out of Ground Effect and In Ground Effect. To cope with the strong variations of aerodynamic coefficients In Ground Effect a modified gain scheduling approach has been employed for the synthesis of the controller by using six State Space Models. Stability and control matrices have been evaluated by linearization of the obtained aerodynamic coefficients. To achieve a simple structure of the control system, an original landing geometry has been chosen, therefore it has been imposed to control the same state variables during both the glide path and the flare.

1 General Introduction

In spite of a number of potentially valuable civil UAS applications The International Regulations prohibit UAS from operating in the National Air Space. Maybe the primary reasons are safety concerns. In fact their ability to respond to emergent situations involving the loss of contact between the aircraft and the ground station poses a serious problem. Therefore, to an efficient safe insertion of UAS in the Civil Air Transport System one important element is their ability to perform automatic landing afterwards the failure. At the present, a few number of UAS is fully autonomous from takeoff to landing [1]. Moreover, the mathematical model of ground effect is usually neither included in the model of the aircraft during takeoff and landing nor in the design requirements of the control system [1], [2], [3], [4]. Some authors take into account the ground effect using a mean value of down-wash angle [5]. To cope with strong variation of aerodynamic characteristics most of papers make use of two different mathematical models of the aircraft during landing: the first Out the Ground Effect (OGE) and the second In Ground Effect (IGE).

Besides for an automatic longitudinal landing control, two different control systems are used:

a glide path control system during the glide slope phase and a flare control system in order to execute the flare maneuver [2], [3], [4] [5]. Usually ,during the glide slope glide path angle, pitch attitude and air speed are controlled [2], [3], [4] .Other authors use normal acceleration, air speed and pitch rate [5]. A lot of paper employees altitude and descent velocity. Recently, because of either GPS use or the increase of sensor's performances for the angular rates measurement, pitch angle and pitch rate are often used [2], [3], [6], [7], [8]. Sometimes, instead of airspeed (V) , because of the small values of the glide slope angle, aircraft velocity along the longitudinal axes (u) and elevation are controlled and the altitude is employed to tune the control laws[9]. As the airplane gets very close to the runway threshold, the glide path control system is disengaged and the flare control system is engaged. This one controls either the vertical descent rate of the aircraft, or the air speed and altitude [2], [3], [4], [5]. To control height accurately in the presence of wind and gust the perpendicular distance and velocity from the required flight path are used to calculate a demanded maneuver acceleration, this one, by means of aircraft speed and orientation is converted to pitch rate [10].

Obviously the above mentioned approach leads to a complex structure of the control system, therefore it could give rise to significant system errors due to unmodelled ground effect. To overcome these complexities, the objective of this paper is the design of a longitudinal control system having the following characteristics:

a. The controlled variables are the same during both the glide path and the flare;
b. According to previous papers [11], [12]; the aerodynamic coefficient vs. altitude are modeled during takeoff instead of using a mean value [13], [14];
c. Indirect flight path control is carried out by controlling the velocity vector (Airspeed V and glide path angle γ);

Item a. allows to achieve a simple structure of the control system independently of the actual flight phases. Item b. permits to take into account the actual ground effect. Item c. implies that the elevator and the throttle control the velocity vector, during the whole path.

Because of high angles of attack during landing, a nonlinear mathematical model of the aircraft should be used for designing the controller [15], [16]. As a consequence, to obtain satisfactory performance, nonlinear controllers should be developed [17]. To overcome the difficulties due to the use of nonlinear models of the aircraft in ground effect, a gain scheduling flight control system has been designed using the following approach:

– The Landing flight path has been divided into two segments: the glide path for aircraft altitudes $h >$ *of the* wing span b (OGE) and the flare for $h <= b$ (IGE);
– The flare manoeuvre starts for $h = b$;
– An acceptable number of linear models has been obtained by means of linearization of the original nonlinear model in various flight conditions: one in OGE condition (from 300 ft to $h = b$) and five in IGE conditions (during flare). (These ones are necessary to employ the linearization through the small disturbance theory).
– A modified gain scheduling approach has been employed for the synthesis of the controller. Initially, by using the obtained linear models, various PID controllers have been designed. Afterwards the obtained PID gains have been modelled by using analytical equations,taking into account the hyperbolic variations of the aerodynamic coefficients. Finally, by linearization of the obtained equation for the gains a set of control gains matrix has been calculated.
– A flight control system has been implemented consisting of the above PID controllers and a supervisor which schedules one of them to be inserted online, depending on the actual flight condition.

The contributions of this paper are: the general model of the aerodynamic coefficients in the whole range of altitude from OGE to IGE condition, the original landing geometry, the simple structure of the control system. Therefore, the system is easily configurable since to control the velocity vector only a small set of sensors are necessary. In fact, by using both Inertial Measurement Unit (IMU) and air data boom, pitch attitude (ϑ) ,airspeed (V) and angle of attack (AOA, α) are easy obtainable. Otherwise a low rate GPS may be used to obtain glide path angle (γ) airspeed (V) and vertical ground speed (V_Z).

2 Flight Control Research Laboratory

The studied research aircraft is used for the Italian National Research Project PRIN2008.

The subject vehicle is an unpressurized 2 seats, 427 kg maximum take of weight aircraft. It features a non retractable, tail wheel, landing gear and a power plant made up of reciprocating engine capable of developing 60 HP, with a 60 inches diameter, two bladed, fixed pitch, tractor propeller. The aircraft stall speed is 41.6 kts, therefore it is capable of speeds up to about 115 kts (Sea level) and it will be cleared for altitudes up to 10.000 ft. (Fig. (1))

Because of it is used as a Flight Control Research Laboratory (FCRL) the studied aircraft is equipped with a research avionic system composed by sensors and computers and their relative power supply subsystem. In particular the Sensors subsystem consists of :

- Inertial Measurement Unit (three axis accelerometers and gyros)
- Magnetometer (three axis)
- Air Data Boom (static and total pressure port, vane sense for angle of attack and sideslip)
- GPS Receiver and Antenna
- Linear Potentiometers (Aileron, Elevator, Rudder and Throttle Command)
- RPM (Hall Effect Gear Tooth Sensor)
- Outside air temperature Sensor

Geometrical characteristics of the subject vehicle are:

- Wing area S: 120 ft^2
- Wing chord c: 3.934 ft
- Wing span b: 30.5 ft

Fig. 1 Flight Research Laboratory L.A.U.R.A.

3 UAS Mathematical Model

When an aircraft flies close to the ground, this imposes a boundary condition which inhibits the downward flow of air associated with the lifting action of wing and tail. The reduced downwash mainly reduces both the downwash angle ε and the aircraft induced drag, therefore it increases both the wing-body and the tail lift slope. Therefore, the lift increases, the neutral point shifts, the pitching moment at zero lift varies. So, stability derivatives In Ground Effect must be used during take-off and landing for aircraft altitudes similar to the wing span b.

Because of these effects, stability derivatives have to be modified and so it is very important a mathematical model which afford to evaluate their behavior in ground effect.

Therefore, for ground distance h ≤ b it's necessary to evaluate the h-derivatives.

In previous researches [11], [12], a mathematical general methodology has been tuned up to evaluate the aerodynamic characteristics variation laws due to the altitude. Such a methodology permits the calculation of aerodynamic coefficients either OGE or IGE. It has been found that aerodynamic coefficients can be expressed by means of hyperbolic equations [11].

According to [18], to evaluate the influence of ground effect on aerodynamic coefficients the variation of either angle of attack (α), or downwash angle (ε) or aspect ratio (A) due to flight altitude have been modeled by using classical methodologies [13], [19] by:

$$\Delta\alpha = -F_{tv}\left[\frac{9.12}{A} + 7.16\frac{c_r}{b}\right]c_{L_{wf}} - \frac{A}{2c_{L_{\alpha_{wf}}}}\frac{c_r}{b}\left[\frac{L}{L_0} - 1\right]c_{L_{wf}}r_g \qquad (1)$$

where:

- F_{tv} represents the vortex effect on the lift;
- A represents the aspect ratio;
- c_r represents the cord in the wedge wing section;
- $c_{L_{wf}}$ represents the lift coefficient, in this case it is the value in the equilibrium position;
- $\frac{L}{L_0} - 1$ represents a correction factor that take into account of the vortex non linear effects on the lift;
- r_g represents the non linear correction factor for taking into account that the wing is finite

$$\Delta\varepsilon = \frac{b_{eff}^2 + 4(H_h - H_w)^2}{b_{eff}^2 + 4(H_h + H_w)^2} \qquad (2)$$

where:

- H_h e H_w represent the tail and wing height from the ground;
- b_{eff} represents a non linear term that links contributes due to the wing span in IGE condition. It can be expressed by:

$$b_{eff} = \frac{c_{L_{wf}} + \Delta c_L}{\frac{c_{L_{wf}}}{b_w^I} + \frac{\Delta c_L}{b_f^I}}$$

where:

- $c_{L_{wf}}$ represents the lift coefficient in OGE condition;
- Δc_L represents the lift increase in OGE condition;
- b_w^I is calculated by $b_w^I = b\left(\frac{b_w^I}{b}\right)$ and the $\frac{b_w^I}{b}$ ratio is known in literature;
- b_f^I is calculated by $b_f^I = b\left(\frac{b_f^I}{b_w^I}\right)\left(\frac{b_w^I}{b}\right)$ and the $\frac{b_f^I}{b_w^I}$ ratio is in literature.

$$-\frac{1}{\pi A_e} = \frac{c_{L_{\alpha_{IGE}}}}{c_{L_{\alpha_{OGE}}}}\frac{1}{c_{L_{\alpha_{airfoil}}}}\left(1 - \frac{c_{L_{\alpha_{airfoil}}}}{\pi A_{OGE}}\right) - \frac{1}{c_{L_{\alpha_{airfoil}}}} \qquad (3)$$

By using Eq. (1), Eq. (2) and Eq. (3) the longitudinal stability derivatives may be determined as:

$$
\begin{aligned}
c_{L_\alpha} &= 3.7931 + 0.01091\left(\frac{h}{b}\right)^{-1.416}\\
c_{M_\alpha} &= -0.8798 - 0.03147\left(\frac{h}{b}\right)^{-1.405}\\
c_{L_{\dot\alpha}} &= 0.9452 - 0.05758\left(\frac{h}{b}\right)^{-1.406}\\
c_{M_{\dot\alpha}} &= -2.4528 - 0.1501\left(\frac{h}{b}\right)^{-1.403}\\
c_{L_q} &= 4.5086\, rad^{-1}\\
c_{M_q} &= -7.3647\, rad^{-1}\\
c_{T_V} &= -0.0896\, rad^{-1}\\
c_{L_h} &= -0.006653\left(\frac{h}{b}\right)^{-2.463}\\
c_{D_h} &= -0.0001002\left(\frac{h}{b}\right)^{-2.685}\\
c_{M_h} &= 0.001339\left(\frac{h}{b}\right)^{-2.717}
\end{aligned}
\tag{4}
$$

4 Flight Path Model

As it is known, the landing procedure is constituted of a slope segment and a flare. Obviously the glide slope phase start out of ground effect, afterwards ground effect is to take into account. Therefore because of, as previous stated, strong variation of aerodynamic coefficients are due to ground effect, a non linear model of the studied aircraft should be used. To overcome this difficulty an original landing geometry has been chosen:

- The first part starts when UAS is at h=300ft and it is a constant speed descent until *h=b* with a glide slope angle *γ=-6°* (because there are not passengers, and so it's not necessary take into account some wealth requirement, it's possible don't consider flight path angle γ limitations);
- the second part is a constant speed flare which start when *h=b* (UAS in IGE condition).

Therefore it has been divided the second condition in five steps based on the values of the h/b ratio. Each one of these steps have a 0.2 h/b ratio size. So the first step is from h/b=1 to h/b=0.8 and so on (in this way there are not strong variation of flight altitude in each step).

This subdivision has been necessary to employ the linearization at several equilibrium flight conditions over the desired flight path; then the small disturbance theory may be applied. In this way difficulties due to the use of nonlinear models of the aircraft in ground effect have been overcome and, in particular, it has been obtained six stability matrices and six control matrices (one for OGE condition and five for IGE condition for each kind of matrices).

The studied flight path is governed by the two following equations:

$$
\begin{aligned}
&h = -0.1051x + 91.44\\
&\text{for } 0 \le x \le 766.8\, m\\
&h = \tfrac{1}{2}\left(3952.4-\left(3952.4^2 - 4(x^2 - 1927.6x + 932880)\right)^{\frac{1}{2}}\right)\\
&\text{for } x > 766.8\, m
\end{aligned}
\tag{5}
$$

where x=0 h= 300 ft represents the beginning of the landing path.

5 Automatic Landing System

Because of the small disturbance theory permits to decouple longitudinal and lateral motion only longitudinal equations have been used in the present study.

$$
\begin{aligned}
\dot V &= \frac{T}{m}cos(\alpha_T+\alpha) - \frac{D}{m} - g sin(\alpha_T+\alpha)\\
\dot\alpha &= \frac{T}{mV} sin(\alpha_T+\alpha) - \frac{L}{mV} + \frac{g}{V}cos(\alpha_T+\alpha) + q\\
\dot q &= \frac{M}{I_y}
\end{aligned}
\tag{6}
$$

Considering Eq. (4), stability derivatives have been calculated in one value of *h/b* inside each of the five steps considered. Because of altitude variation inside each step is small, it has been possible to consider derivatives constant inside each step.

The above hypothesis bring to the following linear state space equation:

$$\dot{x}(t) = A(h)x(t) + B(h)u(t) \qquad (7)$$

where the state vector is

$$x(t) = [\Delta V\ \Delta\alpha\ \Delta q\ \Delta\theta\ \Delta h]^T \qquad (8)$$

and the control vector is

$$u(t) = [\delta_{el}\ \delta_{th}] \qquad (9)$$

Air speed and flight path angle has been controlled during the whole procedure. In particular deflection of elevator is used to control the speed while the deflections of elevator and throttle together are used to control the flight path angle.

Besides following requirements have been imposed:

- maximum tracking error less than 10% during the studied flight path;
- rise time smaller than 5 seconds;
- settling time smaller than 15 seconds.

Because of during the landing flare it is necessary to correlate the vertical speed to the instantaneous distance from ground, an altitude control has been effected through an external P feed forward control loop. Such a system is engaged from flight altitude from 6 m to touch down. Elevator deflections are employed to control flight altitude. Obviously such a system improves the precision of the controller in flight path following.

Because of, as stated in the previous paragraph, six linear models of the studied aircraft have been obtained six multiple PID controllers have been designed by using time domain specifications.

In particular the elevator PID controllers are MISO systems, whereas the throttle PID controllers are SISO systems.

Each one of the obtained sets of gains belongs to one value of the h/b ratio.

Afterwards a supervisor, has be implemented. This one, by using the actual flight altitude, schedules the set of gains to be inserted online, depending on the real flight condition.

In this way it is possible to take into account the ground effect.

6 Results

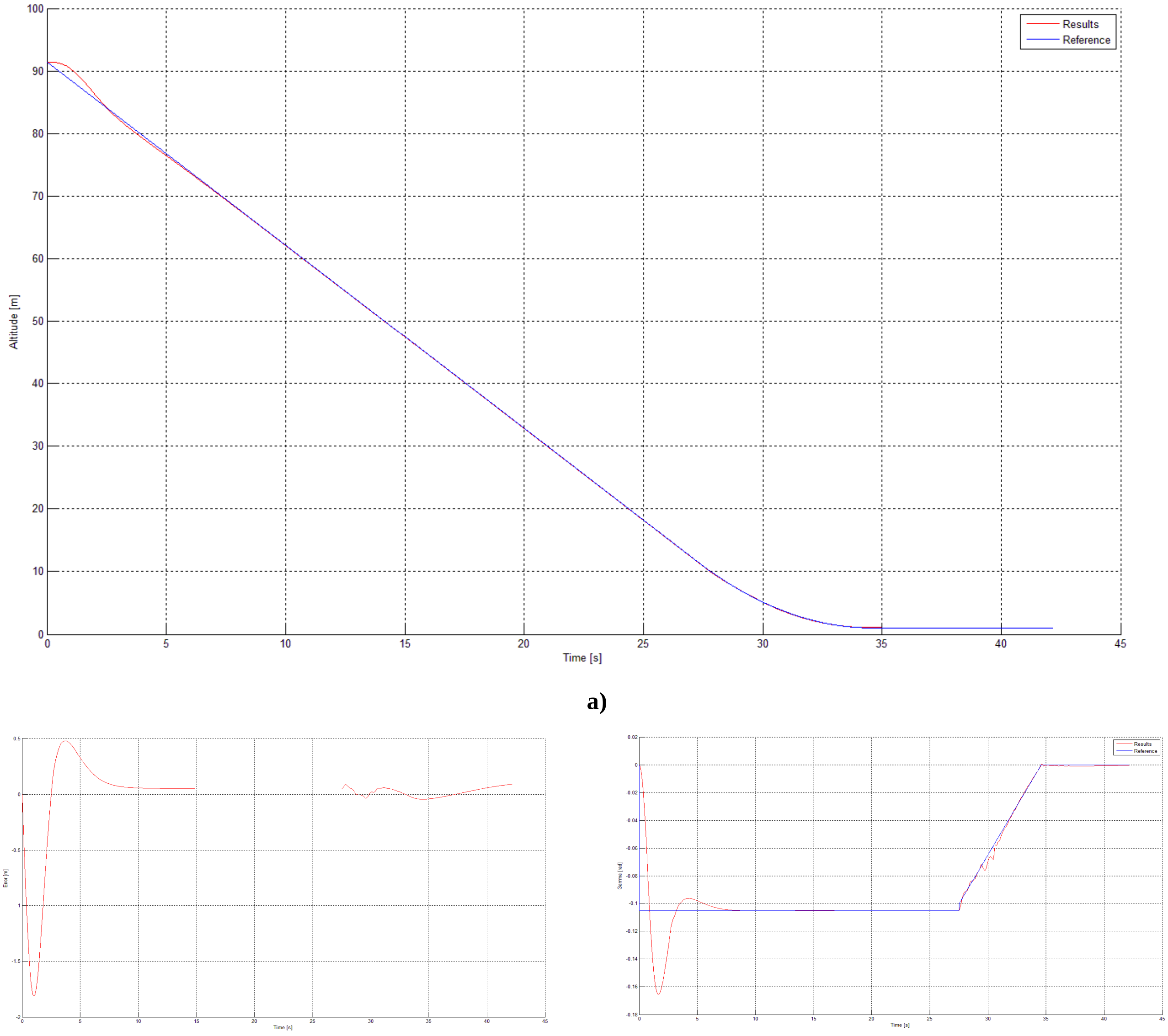
Before its implementation on board the Flight Control System has been tested in MATLAB Simulink environment, Fig. (1-a) shows the flight path of the center of mass (notice that because of the presence of the landing gear the touchdown corresponds to h=1.1 m). As previous stated, the landing has been divided in descent (that starts when UAS is at h=300ft) and flare covered both at constant speed. According to JAR VLA the approach speed is equal to 1.3 times stall speed.

To test the system ability to perform autonomous landing in case of remote control failure, it has been decided to use a horizontal flight condition as equilibrium condition. The selected initial conditions are:

$$h = 300\text{ft}$$
$$V = 54.08 \text{ kts}$$
$$\alpha_e = 0.072 \text{ rad}$$
$$q = 0$$
$$\vartheta_e = \alpha_e$$

The beginning of Fig. (2-a) shows transition from horizontal flight to descent with a little knee and it's possible to see such a transition also in Fig. (2-c) where the glide path angle is shown.

In the center-right side of Fig. (2-a) it's possible to note that the studied control system tracks the desired path with minimal error. In fact in Fig. (2-b) it's possible to see that the maximum error is about 1.9 m ,when the UAS altitude is 94m: This error is due to the selected equilibrium condition. Then, during descent, the tracking error is constantly about 0.05 m. When the flare start, at about 26 seconds, tracking error comes to 0.09 m but it quickly come back to values less than 0.05 m. So it's possible to say that the FCS permits to track the desired trajectory with noticeable precision.



a)

b)

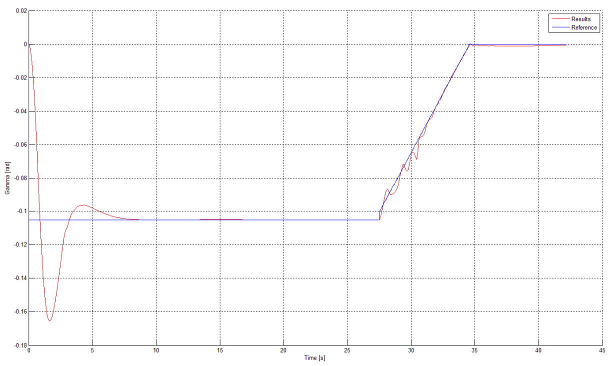
c)

Fig. 2 Indirect flight path following:

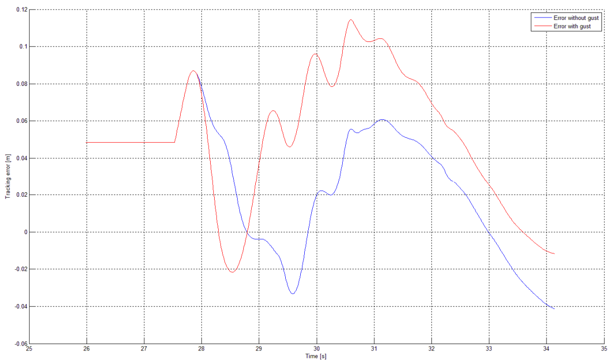
a) desired and controlled flight path, b) tracking error, c) desired and controlled flight path angle.

This affirmation is comforted also with the Fig (2-c) observation, in fact it's possible to note that desired flight path angle and real flight path angle are overlapped except for few. seconds when the descent starts and when the flare starts.

Obviously, to design the Automatic Landing System the classical hypotheses of air at rest has been made. To verify either robustness properties of the controller or its ability to reject disturbances several operating situations such as flight in the presence of gust, rear or front wind and atmospheric turbulence have been considered. Some of the most relevant results are shown in Fig. (3). It refers to a vertical gust that modifies the aircraft angle of attack of $\Delta\alpha=-1$ deg. The vertical gust is inserted into the model at the start of the flare maneuver. Despite of the angle of attack reduction, due to the gust, it is interesting to note that the aircraft performs the flare with a noticeable precision. In fact either the desired flight path or the flight path angle (Fig. 3-a) are followed with negligible errors. Figure (3-b) shows a comparison between tracking errors. It is possible to note that the maximum error is 0.08 m. The mean value of the tracking error is 0.049 m during the whole flare phase. Near the touch down the stabilized tracking error is about 0.02 m.



a)



b)

Fig. 3 Indirect flight path following with gust:

a) desired and controlled flight path angle, b)tracking error comparison

7 Conclusion

The obtained results show the effectiveness of the designed Automatic Landing System. Therefore these ones show a good accuracy of the control system for trajectory tracking in ground proximity. In fact the UAS follows the desired flight path with a noticeable precision. The following original contributions can be highlight: 1) the obtained model of the aerodynamic coefficient In Ground Effect that afford to evaluate stability and control derivatives variations during the landing 2) the use of airspeed and glide path angle as controlled variables during the whole landing 3) the landing geometry.

Further developments of the present research will be the extension of the designed control system to the take-off phase.

Afterwards the aircraft model will be improved by evaluating both lateral stability derivatives variations In Ground Effect and the bank angle derivatives (ϕ derivatives). Since the present methodology will be employed to design a Lateral Automatic Landing System.

At the present flight tests are performing to verify the effectiveness of the designed Automatic Longitudinal Landing System by means of the above described Flight Control Research Laboratory. The obtained results could be used later on, with the purpose to realize a fully autonomous UAS.

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