ACTIVE AEROELASTIC CONTROL OF VARIABLE CAMBER AIRFOIL IN THE PRESENCE OF STRUCTURAL NON-LINEARITIES

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Abstract. *Modern smart materials technologies have led to the idea of aircraft that may actively change their shape to maneuver or to control aeroelastic response. In this context, it is reasonable to consider that nonlinear aeroelastic behavior represents an important issue. If an aircraft structure presents mildly or severe non-linear behavior, the active aeroelastic control by means of changing aircraft shape can represent a great challenge. This work presents an investigation on active fuzzy control of aeroelastic response using variable camber airfoils. Fuzzy aeroelastic control law modeling, where non-linear structural behavior is considered, has been aimed. Distributed lumped vortex method has been used to determine unsteady aerodynamic responses and typical structural have been also adopted for the aeroelastic simulation. Camber variation is achieved by means of time-varying polynomial description of the airfoil camber line. Fuzzy logic approach also allows easy assessment to aeroelastic control because of its relative independence on precise aeroelastic system modeling as a requirement to its design. As a preliminary work on variable geometry aeroelastic structures, typical fuzzy logic schemes have been used. The results show a robust and acceptable methodology to threat aeroelastic response control of variable shape aerodynamic structures.*
Keywords*: Aeroelasticity, variable camber, smart airfoil, fuzzy control, non-linear aeroelasticity*

1. Introduction

Generally aircrafts are optimised for specific flight conditions. When an aircraft operates away from the optimal design points, the performance will decline. Nowadays the interest in air vehicles that can operate in several conditions, such as efficiency in subsonic and supersonic cruise or high manoeuvrability is increasing. The use of adaptive technologies can be a solution to improve the performance at these conflicting conditions and increase the optimum flight envelope (Sanders, Eastep and Foster, 2003). Lifting surfaces with variable geometry can be one of these technologies. The recent development of smart materials has been pointed out as an important factor to the development of this kind of structure (Johnston, 2003).

The geometry of a lifting surface can be modified in many different ways. Traditionally it has been modified by the use of hinged surfaces, such as flaps and trim tabs that modifies camber. Other forms of geometry changes are wing warping, wing sweep, wing twist (Bowman, Sanders and Weisshaar, 2002). In general, these modifications are limited by aerodynamic and structural requirements and are typically efficient in a small range of velocities, for example. The development of lifting surfaces that can change their geometry smoothly and actively through the camber variation, twist angle and sweep angle variation, can eliminate the need for the conventional hinged control surfaces (Pern and Jacob, 1999; Sanders, Crowe and Garcia, 2004; Simpson *et al.*, 1998). This new technology has been increasing the aerodynamic and aeroelastic performance of lifting surfaces in a wide range of conditions through the active flow control and active control of aeroelastic responses. These benefits will imply in a more efficient structural design.

The smooth variation of camber has been studied by many authors and is the main interest of this work. In the 1970s the Mission Adaptive Wing program demonstrated on the F-111 a smooth camber variation in chordwise direction (Smith and Nelson, 1990). The Active Aeroelastic Wing utilizes multiple leading-edge and trailing-edge control surfaces to use possible benefits of wing twist (Pendleton *et al.* 2000). Moner (2001) and Stanewsky (2000) describe the development of a flexible flap to be used in a civil transport aircraft that allows smooth camber variations. The aerodynamic, aeroelastic and structural benefits are clearly shown by these authors.

The possible application of smart materials has been studied by some authors. Ehlers and Weisshaar (1993) developed an analytical study of a wing with variable geometry using piezoelectric patches as actuators. Several improvements in aerodynamic performance and control of static aeroelastic characteristics are observed. Sanders, Eastep and Foster (2003) developed a wind tunnel wing model that used shape memory alloy torque tubes to twist the wing and shape memory alloy wires to smoothly deform trailing-edge control surfaces. Comparisons between a conventional hinged control surface and the shape memory alloy actuated wing show the aerodynamic and aeroelastic benefits of the new technology. Previously to the experimental tests the aerodynamic and aeroelastic performance were investigated using a computational code that associates a vortex lattice method and finite element modelling. In the same line, Gern, Inman and Kapania (2002) have developed a method to study the structural behaviour and static aeroelastic response of a general wing. The structural model is based on the assumption that the wing behaves like a plate and can be modelled by the first order shear deformation theory and a compressible vortex lattice code is used for aerodynamic model. The manoeuvre, aerodynamic and aeroelastic improvements were demonstrated for a wing that can modify camber and twist angle.

The new structures obtained with the new technologies will certainly have inherent non-linearities, as is already verified in real structures. Different types of structural and aerodynamic non-linearities are commonly encountered in aeronautical engineering. Lee, Price and Wong (1999) show a comprehensive study of these different types of non-linearities present on aeroelastic systems. Non-linear aerodynamic effects are more difficult to analyse since the fluid motion is governed by equations where analytical solutions are practically non-existent. Structural non-linearities arise from worn hinges of control surfaces, loose control linkages, material behaviour and various other sources. Aging aircraft and combat aircraft that carry heavy external stores are more likely to be influenced by effects associated with non-linear structures. This type of non-linearity can be treated as a concentrated non-linearity, and usually can be approximated by one of the three classical structural non-linearities, namely, cubic, bilinear and hysteresis.

The consequences of the non-linear aeroelastic responses and their control have been widely studied. Block and Strganac (1998) explain that fighter aircrafts have experienced limit cycle oscillations for certain attached wing store configurations. The mechanism that leads to these LCOs is not well understood, but explanations under study include aerodynamic and/or structural non-linearities. Stiffness tests show evidence of a spring-hardening non-linearity in the wing torsional mode. This type of non-linearity will lead to LCO behavior similar to the described in this work. Block and Strganac (1998); Singh and Yim (2003); Zhang and Singh (2001); Ko, Kurdila and Strganac (1997) among others have developed control strategies for non-linear aeroelastic phenomena suppression.

This article presents the study of non-linear aeroelastic response of an airfoil. The non-linear aeroelastic response is then controlled by means of variable geometry lifting surfaces, more specifically variation of camber in chordwise direction. Non-linear structural behaviour in torsion is adopted and an aeroelastic model is accomplished using lumped vortex method. A fuzzy logic controller emulated with a neural network is used as a preliminary tool in order to test the control actuation concept.

2. Aeroelastic Model

A two-dimensional airfoil having two degree of freedom, as depicted in Fig. 1, is investigated.

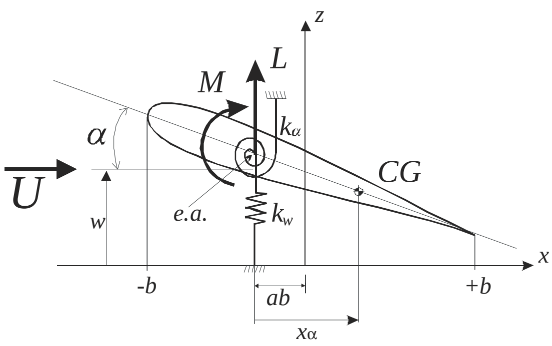


Figure 1. Aeroelastic model.

The bending and torsional variables are denoted as w and α, respectively, and the equations of motion for this typical section are obtained as (Fung, 1993),

$$\begin{bmatrix} m & -mx_\alpha b \\ -mx_\alpha b & I_\alpha \end{bmatrix} \begin{Bmatrix} \ddot{w} \\ \ddot{\alpha} \end{Bmatrix} + \begin{bmatrix} k_w & 0 \\ 0 & k_\alpha(\alpha) \end{bmatrix} \begin{Bmatrix} w \\ \alpha \end{Bmatrix} = \begin{Bmatrix} L \\ M \end{Bmatrix} \tag{1}$$

In Eq. (1), m the mass of the system; I_α is the mass moment of inertia about the elastic axis; x_α is the non-dimensionalized distance between the centre of mass and elastic axis; L and M are the aerodynamic lift and moment; b is the semichord of the wing and the two structural spring forces are represented by k_w and k_α for bending and torsion, respectively.

Non-linear effects due to aerodynamics, damping or structural dynamics can be incorporated to this model. In this work, the structural damping effects are neglected and the source of non-linearities is assumed to be in the pitching stiffness. The polynomial form of Eq. (2) can be an approximation of this kind of non-linearity (Ko, Strganac and Kurdila, 1998).

$$k_\alpha(\alpha) = k_{\alpha_0} + k_{\alpha_1}\alpha + k_{\alpha_2}\alpha^2 + k_{\alpha_3}\alpha^3 + k_{\alpha_4}\alpha^4 + \cdots \tag{2}$$

An aerodynamic model has to be assumed to represent the terms on the right side of Eq. (1). The solution of an aeroelastic problem is strictly connected to the quality of the aerodynamic model. The unsteady air load over the airfoil for the incompressible potential flow is solved with the lumped vortex method (Katz and Plotkin, 2001). The applied numerical method was modified to make possible the time variation in airfoil camber. The method seems to be appropriate for the calculation of the air load on the airfoil with variable geometry, but it is computationally more demanding if compared with that for the rigid airfoil case. Within every time step a new position of point vortices and collocation points must be evaluated.

The airfoil is described by a set of discrete vortices on the camberline, Γ_j, as observed in Fig. 2. When the airfoil's circulation changes, the vortex wake elements, Γ_W, are shed at the trailing edge and the wake is modelled using the same vortex element. Two boundary conditions are defined: the zero normal velocity across de body's solid boundaries and the flow disturbance, due to body motion through the fluid, should diminish far away from the body. Also, Kelvin condition and Kutta theorem have to be respected.

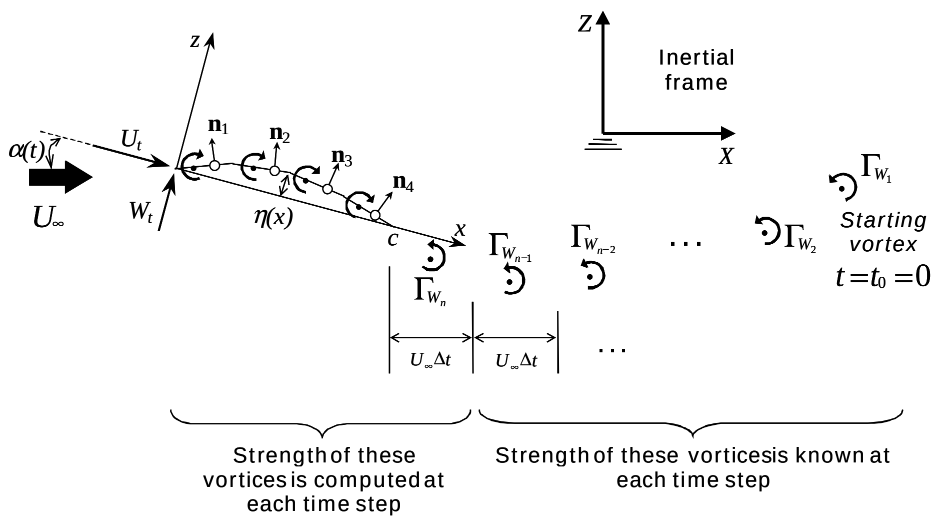


Figure 2. Discrete vortex model for the unsteady air load on the airfoil calculation (Katz and Plotkin, 2001).

The induced velocities, *u* and *w*, at an arbitrary point *(x, z)* due to a vortex element Γ_j located at (x_j, z_j) is given by Eq. (3), where $r_j^2 = (x - x_j)^2 + (z - z_j)^2$.

$$\begin{Bmatrix} u \\ w \end{Bmatrix} = \frac{\Gamma_j}{2\pi r_j^2} \begin{bmatrix} 0 & 1 \\ -1 & 0 \end{bmatrix} \begin{Bmatrix} x - x_j \\ z - z_j \end{Bmatrix} \tag{3}$$

Considering that the airfoil camberline is divided into *N* panels. The vortex points (x_j, z_j) are placed at the quarter chord of each planar panel and the zero normal flow boundary condition can be fulfilled on the camberline at the three quarter point (collocation point) of each panel. The normal vector $\mathbf{n}_i$ at each of these collocation points is found in the body's frame from the surface shape $\eta(x)$ is,

$$\mathbf{n}_i = \frac{(-d\eta/dx, 1)}{\sqrt{(d\eta/dx)^2 + 1}} = (sin\alpha_i, \cos\alpha_i) \tag{4}$$

As the airfoil's geometry changes with time in the case considered by this paper, vortex and collocation points calculations have to be done at each time step loop.

The influence coefficient a_{ij} is defined as the velocity component induced by the airfoil's unit strength Γ_j element, normal to the surface at the collocation point *i*. The set of algebraic equations shown in Eq. (5) are obtained for each collocation point. In Eq. (5), the right-hand side (*RHS*) terms are known at each time steps and are composed by the kinematic velocities due to the motion of the airfoil plus the velocity components induced by wake vortices, except the latest one. As the airfoil's geometry changes with time, the influence coefficient calculation has to be done at each time step loop.

$$\begin{bmatrix} a_{11} & a_{12} & \cdots & a_{1N} & a_{1W} \\ a_{21} & a_{22} & \cdots & a_{2N} & a_{2W} \\ \vdots & \vdots & \ddots & \vdots & \vdots \\ a_{N1} & a_{N2} & \cdots & a_{NN} & a_{NW} \\ 1 & 1 & \cdots & 1 & 1 \end{bmatrix} \begin{Bmatrix} \Gamma_1 \\ \Gamma_2 \\ \vdots \\ \Gamma_N \\ \Gamma_{W_t} \end{Bmatrix} = \begin{Bmatrix} RHS_1 \\ RHS_2 \\ \vdots \\ RHS_N \\ \Gamma(t-\Delta t) \end{Bmatrix} \tag{5}$$

The Kutta condition is not stated explicitly for the lumped vortex method and the Kelvin condition is represented by

$$\Gamma(t) - \Gamma(t-\Delta t) + \Gamma_W = 0 \tag{6}$$

The pressures and loads are then computed by using the unsteady Bernoulli equation. The total lift and moment are obtained by integrating the pressure difference between the camberline upper and lower surfaces along the chordline. At this point, all the terms necessary to the solution of Eq. (1) are known and it can be solved with the Runge-Kutta method, for example.

3. Deformable camber and control law

Airfoils with variations in geometry can be applied in two main categories. The first one can be described as small and slow alterations of airfoil's geometry that makes possible the optimisation of shape according to the flight conditions. In the second one, faster and major changes will damp vibrations and aeroelastic phenomena, for example (Moner, 2001; Stanewsky 2000). In this work, an airfoil with camber variation is used to damp non-linear aeroelastic responses. The actuators necessary to change the airfoil's camber are considered ideal.

The original, or non-deformed airfoil, is a symmetric one and its straight camber line is represented in Fig. 3. The variation in airfoil's camber is approximated by a third-order polynomial shown in Eq. (7), where *G(t)* is the amplitude of the camber at each time step. An example of camber variation approximated by a third-order polynomial is shown in Fig. 3.

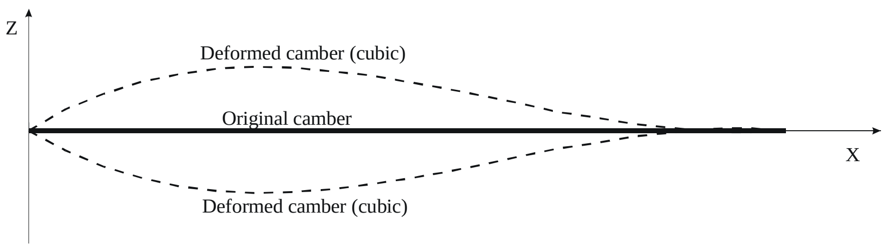


Figure 3. Representation of the original mean camber line and the deformed camber line represented by a third-order polynomial.

$$camber(x,t) = (A_0 x^0 + A_1 x^1 + A_2 x^2 + A_3 x^3) \times G(t) \tag{7}$$

In order to suppress aeroelastic responses, linear or non-linear, a controller has to be designed. A fuzzy logic controller based on the Mamdani-type fuzzy model is applied. Membership functions and a set of rules are defined based on the previous experience obtained from simulations performed with the non-linear aeroelastic model and a decision surface is obtained. The decision surface represents, therefore, the control law on a unitary discourse universe. The control law models the consequent control action like a PD-type controller.

When fuzzy controllers are considered the controller inputs are defined as an error and the variation of this error. The error is the difference between the feedback variable and a reference value. The torsional and bending displacements, α and w of Eq. (1), are used to calculate the controller input signals. The error defined in this work is a composed one. It is the summation of the bending error and the torsional error. The torsional error is the difference between the torsional angle and the zero value and the bending error is the difference between the bending displacement and the zero value. The camber amplitude *G(t)* will be the output of the controller and will change the airfoil's shape.

The input and output signals are normalized by individual gains. These gains are obtained manually and they guarantee stable and efficient response control. The fuzzy membership functions are also obtained and tuned manually. Figure 4 shows the rule basis and the respective decision surface, where *Z* is zero, +*S* is negative small, - *M* is negative medium, -*B* is negative large, +*S* is positive small, +*M* is positive medium, +*B* is positive large.

	Variation of error							
		-L	**-M**	**-L**	**Z**	**+S**	**+M**	**+L**
Error	**-L**	-L	-L	-L	-L	-M	-S	Z
	-M	-L	-L	-M	-M	-S	Z	+S
	-S	-L	-M	-S	-S	Z	+S	+M
	Z	-L	-M	-S	Z	+S	+M	+L
	+S	-M	-S	Z	+S	+S	+M	+L
	+M	-S	Z	+S	+M	+M	+L	+L
	+L	Z	+S	+M	+L	+L	+L	+L

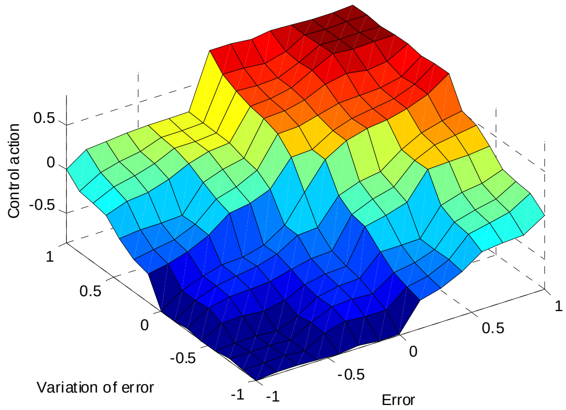


Figure 4. Rule basis and decision surface of the fuzzy controller.

4. Aeroelastic response control and LCO avoidance

4.1. Linear analysis

This subsection presents a study on aeroelastic linear problem of the two-dimensional airfoil section. The set of dimensionless parameters are taken as those in Fung (1993) and shown in Tab. 1.

Table 1. System parameters obtained from Fung (1993), pp. 219.

b = 0.127 m	ω_α = 64.1 rad/s
a = - 0.15	ω_w = 55.9 rad/s
μ = 76	x_α = 0.25
r_α^2 = 0.388	ρ = 1.225 kg/m^3

In Tab. 1 the dimensionless parameters are defined as

$$r_\alpha^2 = \frac{I_\alpha}{mb^2}\,,\ x_\alpha = \frac{S_\alpha}{mb}\,,\ \omega_w^2 = \frac{k_w}{m}\,,\ \omega_\alpha^2 = \frac{k_\alpha}{I_\alpha}\,,\ m = \mu\pi\rho b^2$$

where ω_w and ω_α are the bending and torsional frequencies; r_α is the radius of gyration; μ is the mass ratio; S_α is the static moment; *a* is non-dimensionalized distance from the midchord to the elastic axis.

Fung (1993) assumes a harmonic motion of the airfoil and model the unsteady aerodynamics using Theodorsen's function. The critical flutter speed of 27.5 *m/s* and flutter frequency of 9.5 Hz are determined solving the flutter determinant (Fung, 1993). This critical speed is calculated neglecting the effects of structural damping. The aeroelastic model shown in Eq. (1) also neglects the effect of structural damping.

The critical flutter speed obtained with the aeroelastic model described in Section 1 (considering 20 panels) is 26.0 *m/s* and the flutter frequency is 8.4 Hz, what is close to Fung's results. The open-loop linear aeroelastic responses for bending and torsional displacements at critical flutter speed are shown in Fig. 5. The results obtained in Fig. 5 could validate the aeroelastic model developed in this work.

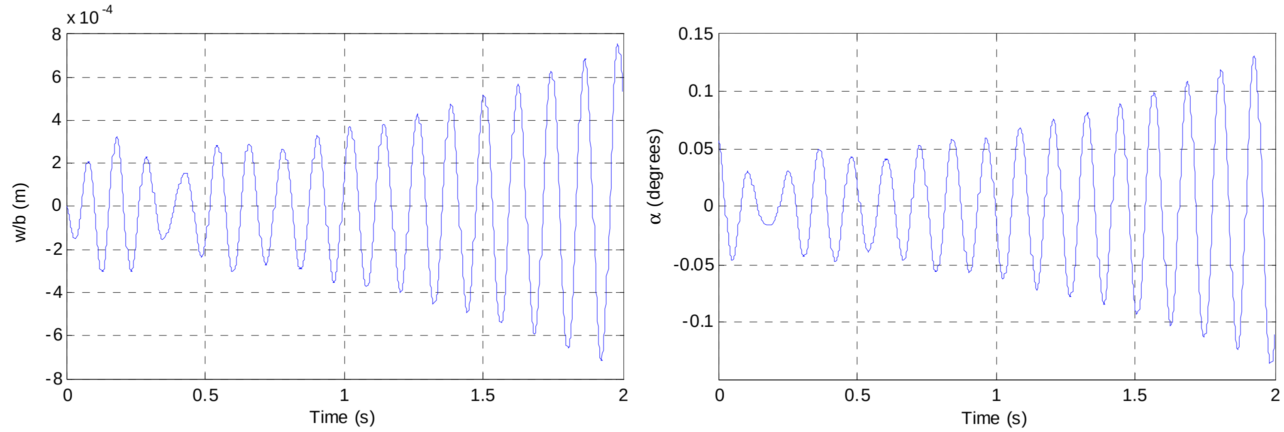


Figure 5. Open-loop linear aeroelastic responses for bending and torsional displacements.

4.2 Non-linear analysis and control

This subsection presents the analysis of non-linear aeroelastic response of the two-dimensional airfoil section. The source of non-linearities is assumed to be the torsional spring, so, the non-linear torsional stiffness is shown in Eq. (8). This equation was obtained from a curve fitting performed with some points chosen near linear torsional moment curve with respect to the angle of attack.

$$k_\alpha(\alpha) = 6000\alpha^2 + 0.49 \tag{8}$$

To verify the non-linear aeroelastic behavior and the performance of the Fuzzy controller the numerical simulations are performed with a freestream velocity of 31.5 *m/s*, that exceeds the linear critical flutter velocity ($U_{cr} \cong 26$ *m/s*). The parameters shown in Tab. 1 are retained. The open-loop non-linear responses with the initial conditions w = 0.015 *m* and α = 0.3 *degrees* are shown in Fig. 6. Figure 7 shows the phase trajectories for open-loop bending and torsional responses. The limit cycle oscillations behaviour is observed in these responses.

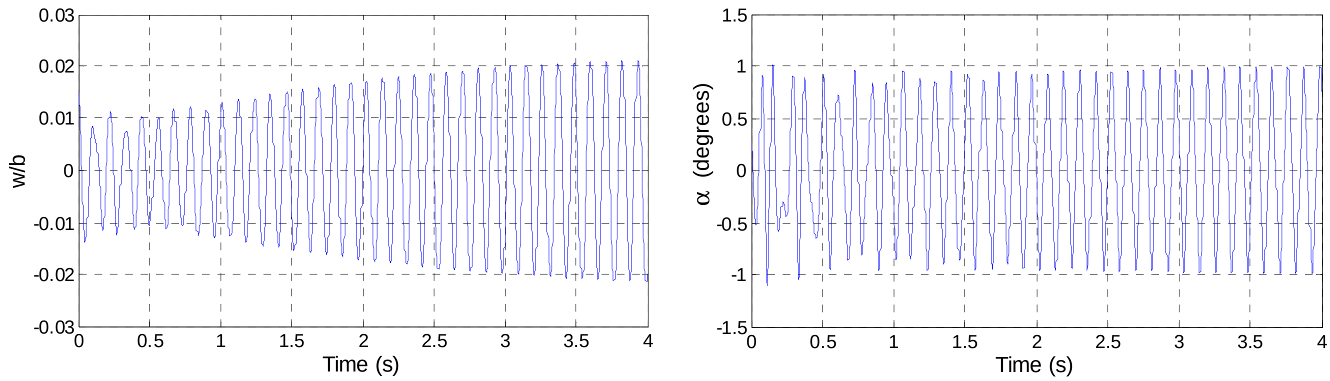


Figure 6. Non-linear open-loop bending and torsional responses (U = 31.5 *m/s*).

The parameters of the third-order polynomial, Eq. (7), that approximate the airfoil's camber variation are set to A_0 = 0, A_1 = 0.875, A_2 = 1.875 and A_3 = 1. The controller gains used to normalize the input and output signals of the fuzzy controller are set to $Gain_{\mathrm{error}} = 10^{-3}$, $Gain_{\mathrm{Verror}} = 10^{-4}$ and $Gain_{co}$ = 0.35. Figure 8 shows the closed-loop bending and torsional responses. The loop is closed at time 1.5 seconds.

The closed loop bending and torsional responses in Fig. 8 converge to zero. The settling time for stabilization of non-linear bending and torsional responses is of the order of 1.2 seconds.

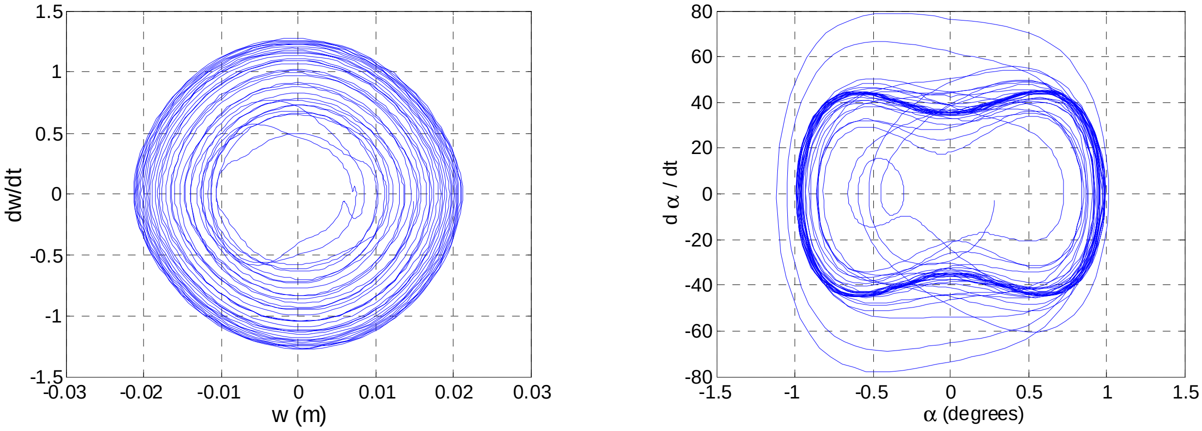


Figure 7. Phase trajectory of bending and torsional.

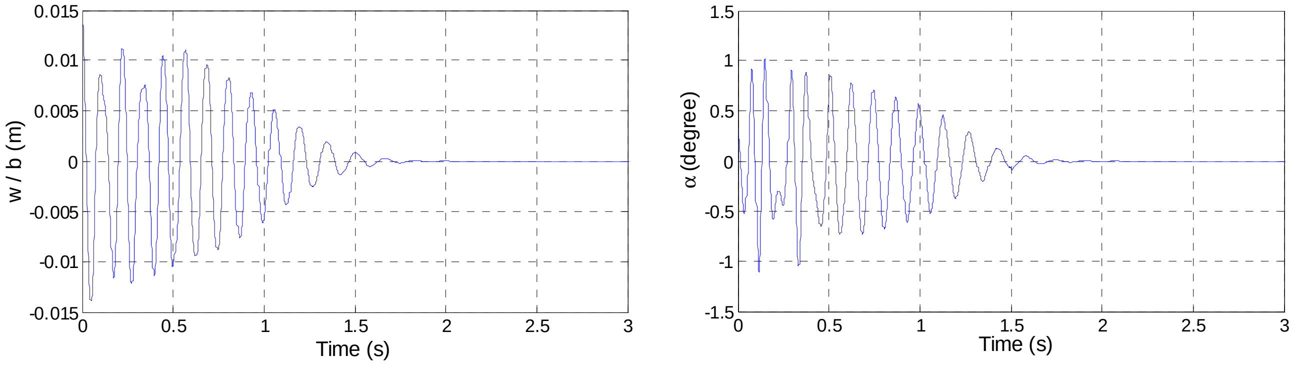
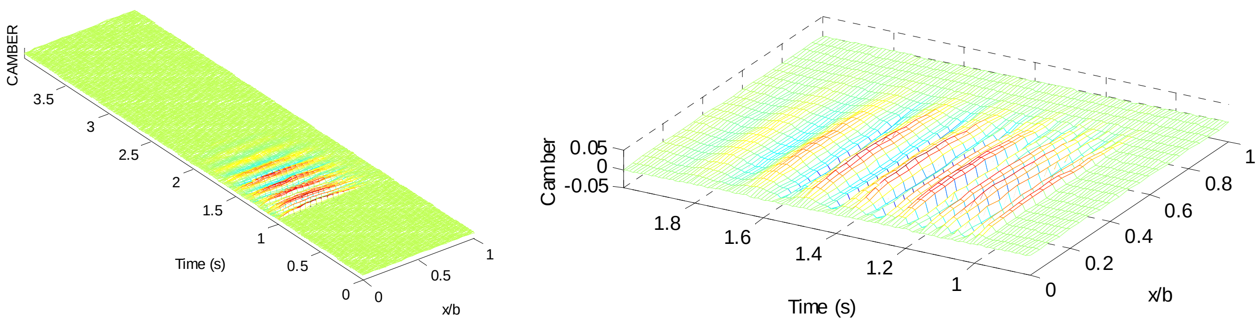


Figure 8. Closed-loop bending and torsional responses.

Figure 9 shows the airfoil's camber variations during the suppression of the LCO behaviour. Figure 9(a) shows the camber variation during the entire LCO control simulation. Figure 9(b) shows a detailed view of the region where the largest camber amplitudes (± 0.02 *m*) were necessary for control action, between 0.4 to 2.0 seconds.



(a) (b)

Figure 9. Variation of camber line during the closed-loop simulations.

5. Conclusions

An aeroelastic model was developed to study the non-linear aeroelastic responses and their control by means of active variable camber. The unsteady aerodynamics was modeled with a lumped vortex method. The linear analysis performed showed a good agreement with results presented in standard text (Fung, 1993) and was also used to perform non-linear investigations. A cubic polynomial non-linearity was assumed for the structural pitch moment. The authors determined this polynomial based on the knowledge obtained during the linear simulations and, even with this approximation the non-linear aeroelastic responses are quite coherent.

The use of a lifting surface with variable camber can be considered an important alternative for future developments of adaptive aircrafts. This kind of actuation associated with the designed fuzzy controller has shown to be a good combination in the suppression of non-linear aeroelastic responses, specifically for limit cycle oscillations avoidance. The time-varying polynomial description of the airfoil camber line has a great influence in the performance of the controller. During the simulations some different polynomials were tested. It is clear that an optimised one exists and a more detailed investigation is necessary.

Ideal actuators were considered in this initial investigation. Figure 9(b) shows that small amplitudes in camber were enough to control the non-linear responses. The results achieved in the control of non-linear aeroelastic responses through the camber variation show that future investigations considering the actuation with of smart materials are pertinent.

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