MODAL AND FLUTTER ANALYZES USING FINITE ELEMENT MODEL FOR AN AERODESIGN AIRPLANE WING

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Abstract. *In the scope of dynamics aeroelasticity, flutter stability has an important role for determination of the critical design speeds in aeronautical projects. Indeed, such phenomenon has been currently found at SAE Brazil Aerodesign Competition, leading the aircraft to catastrophic failure. The present paper performs the aeroelastic analysis of an aerodesign aircraft wing with PKNL method, already implemented in NASTRAN solution package. Doublet Lattice was adopted for aerodynamic model and the post-processing program was developed in order to visualize the V-g-f diagram. The finite element model (FEM) was validated experimentally through static and dynamic tests.*

Keywords: *experimental evaluation, finite element model, p-k method, aerodesign, flutter*

1. INTRODUCTION

Aerodesign competitions stimulate undergraduate students to project and develop a reduced scale transport aircraft. The main idea consists to challenge students to conceive the lightest aircraft able to carry the maximum payload, almost seven times heavier than the empty aircraft weight. The airplanes are, however, more susceptible to aeroelastic problems within its flight envelope. Comprehending and predicting such phenomena in complex structures are possible by simulating the coupled structural-fluid model.

The Finite Element Model (FEM) is an important tool to represent the aeronautical structures, among others, as provided by approximated solutions for complex problems, governed by differential equations (Rade, 2013). In this method, the problem domain is the finite interconnected region and, as the level of elements discretization increases, the solution becomes more accurate. The structural model is useful to verify static deflection of structures under certain loads, to predict the rupture load, and/or to characterize its modal properties.

Indeed, determining the modes shapes and their respective natural frequencies of an aircraft wing is very important into limiting the highest operational speed. According to (Fung, 1991), if the system presents a modal coupling mechanism, flutter dynamic instability can occur. In flutter critical velocity, the aerodynamic forces inputs energy to the aeroelastic system, making it self-excited. Another aeroelastic instability problem is the divergence, in which the aerodynamic loads deforms the structure until it lost its elastic restoration capability (Hodges, 2002)

Hence, for the aeroelastic stability analysis, it is necessary to include in the FEM structure an aerodynamic model. The Doublet-Lattice method (DLM) has been widely used, once the implementation of a Computational Fluid Dynamics (CFD) has a non-linear solution that implies high cost and mechanical compilations. The DLM represents the most rudimentary unsteady aerodynamic technique in practice and considers linearized aerodynamic potential flow theory, being an extension of the steady Vortex-Lattice method (Blair, 1992).

Experimental analyses are required while investigating the correlation between the FEM and the real structure for further improvements into the numerical modeling. The main objective of static and dynamic tests is to characterize experimentally the structure in terms of stiffness and mass properties. The numerical model validation is crucial for the flutter clearance process (Peeters, 2008).

For this paper purpose, an aerodesign wing, made of carbon fiber composite and balsa wood material, is modeled in Nastran®. Structural simulations and modal analysis are carried out, which are later verified through static and dynamic experimental setup. The critical speed at which aeroelastic phenomena may occur is determined with Nastran.

2. NUMERICAL ANALYSIS

2.1 Structure model

The analyzed structure for predicting aeroelastic phenomena consists the wing of Tucano Aerodesign Team's aircraft (see Fig. 1), from Federal University of Uberlândia. It has 3.1 m span, 0.6878 m² area and high aspect ratio of 14. It comprises a simple arrangement of 20 balsa ribs distributed along the laminated carbon spar. The ribs are responsible for transmitting external loads to spar and for maintaining the shape of wing section. They were design with lightness holes and openings for control runs.

Figure 1. Tucano Aerodesign aircraft.

The structural modeling of the wing is done using FEM in the software NASTRAN®. The model intends to reproduce the real structure (mass and stiffness), in terms of geometric parameters and materials properties.

The trailing edge and ribs are modeled as plate elements of balsa wood material, while the leading edge and spars are plate elements of carbon fiber laminate material. The thickness of the laminate material is 0.60 mm for elements between the wing root and fourth rib and for the elements up to wing tip is 0.45mm. Fig. 2 and Tab. 1 show the details of the structural grid.

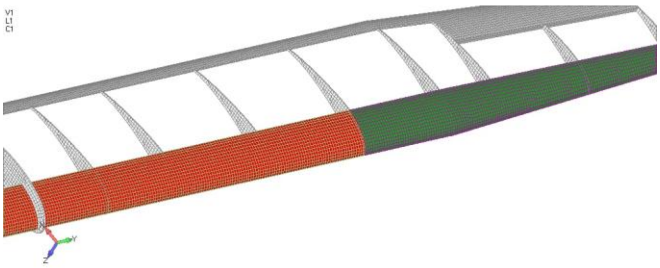


Figure 2. Detail of material distribution: gray is the balsa material, red is the carbon fiber laminate of 0.60 m thickness, green is the laminate carbon fiber of 0.45 m thickness.

Table 1 – Structural grid details.

Element type	**Quantity**	**Total elements**
Quadrangle 2D **(CQUAD4)**	**35810**	**36146**
Triangular 2D **(CTRIA3)**	**296**	
Bar **(CBAR)**	**20**	
Rigid **(RBE2)**	**18**	
Mass **(MASS)**	**2**	

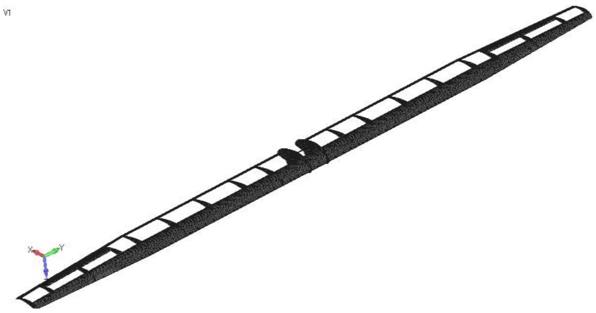
The mechanical properties of the materials are given in Tab. 2. 'EL' and 'ET' is the longitudinal and transversal oriented fiber Young modulus. 'GLT', 'GLR' and 'GTR' are the modulus of rigidity associated with the strain corresponding to the longitudinal-transversal, the longitudinal-radial, and the transversal-radial axes, respectively. For the carbon structures, the longitudinal axis is oriented parallel with respect to the spar's direction, while, for the balsa wood, its longitudinal direction is oriented perpendicular with respect to the span direction.

The static analysis was done using SOL 101 from NASTRAN, as shown in Fig.3 (b), there was applied force and boundary condition, in accord to experimental set up (Section 3.1).

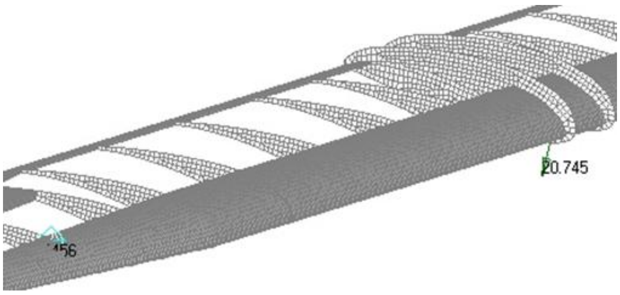
Table 2. Material properties details.

Material	EL	ET	GLT	GLR	GTR	Density
Interlaced carbon 0.60	33.5 GPa.	1.1 GPa.	4.32 GPa.	2.70 GPa.	2.70 GPa.	1534.16 Kg/m³
Interlaced carbon 0.45	22.4 GPa.	1.67 GPa.	7.51 GPa.	2.70 GPa.	2.70 GPa.	1637.00 Kg/m³
Balsa wood	4.35 GPa.	212 MPa.	16.5 MPa.	17 MPa.	17 MPa.	167.71 Kg/m³

In other hand, the modal analysis was also realized using the same software, by means SOL103, and the free-free boundary condition was adopted, as presented in Fig.3 (a). This type of dynamic analysis consists in determining the natural frequencies and their modes shapes by solving the eigenvalue problems associated to the differential equation (Rade, 2010).



(a)



(b)

Figure 3. Structure FEM model (a) free-free configuration for modal analysis; (b) constrains for static analysis.

The stiffness matrix of FEM is adjusted by correlating the wing deflections between numerical simulation and experimental results. The model is finally validated by comparing the natural frequencies values and modes shapes.

2.2 Aerodynamic model and spline method

In NASTRAN®, the aerodynamic coordinate system adopted considers the flow direction in the positive x-axis, in which every aerodynamic element must be parallel to the mentioned coordinate system at its undeformed position. The aerodynamic model consists in panels divided in trapezoidal lift elements, called boxes. CAERO1 entry constructs the aerodynamic panel along the wing surface. The aerodynamic property is obtained with PAERO1 command (MSC Software, 2013). In both commands (CAERO1 and PAERO1), the DLM is set as aerodynamic solver method.

DLM computes the aerodynamic efforts based on the pressure difference on the induced upwash between two different positions. Elementary source solutions are placed throughout the flow domain and each one represents the potential function in coordinates x, y, z, t. The resulting potential field is a superposition of the potential arising from each source point. A source sheet is a two dimensional surface which is partitioned into differential trapezoidal areas where is assigned a point source of varying flow rate (or strength) and the pressure is assumed spatially constant within these areas (Blair, 1992).

For the present paper and due to tapered wing tip, four aerodynamic panels are considered (see Fig. 4). The panels close to root is divided into 390 boxes, while the panels close to tip into 195 boxes. The mesh between two consecutive panels is coincident in the boundary region.

Also, it is necessary to determine mean aerodynamic chord, reference fluid density and symmetry keys, defined by AERO entry. Structural and aerodynamic grids are connected by interpolation, called splining method. SPLINE1 entry references CAERO1 and SET1, which selects the structural hard and upper surface points to be linked.

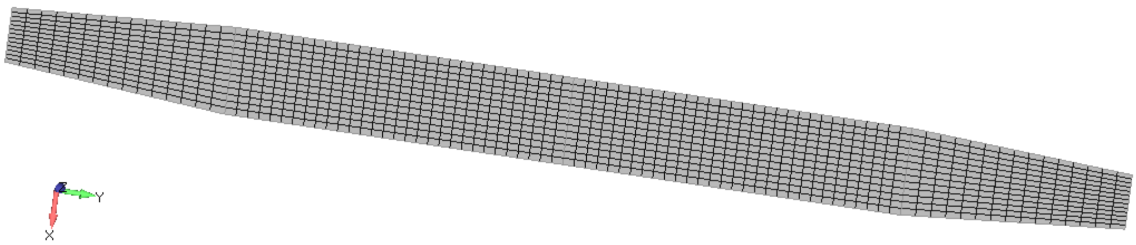


Figure 4. Aerodynamic panel

2.3 Flutter and divergence analysis

The structural and aerodynamic models, described in Section 2.1 and 2.2, respectively, are used to compute the critical speeds at which the wing suffers flutter and divergence phenomena. The aeroelastic analysis of FEM wing is done with SOL 145 and PKNL method in Nastran®. The solution procedure corresponds to PK method without looping in the flight conditions.

A key parameter for flutter analysis is the reduced frequency (k), described in terms of reference length ($c = 198\ mm$), circular frequency (ω) and flow velocity (V), as follow:

$$k = \frac{\omega c}{2V} \tag{1}$$

The aeroelastic analysis considers the aircraft at straight and steady-state flight at different velocity (flow velocity), varying from 5 to 330 m/s in the FLFACT entries. The Mach number, m, and the fluid density, ρ, are considered constants for all simulations. In the MKAERO entry, the reduced frequency is within the interval 0.01 and 3.00, calculated by Eq.(1) for the maximum and minimum ω and V values.

DLM method evaluates $Q_{hh}(m,k)$, the complex generalized aerodynamic influence matrix, as a function of Mach number and reduced frequency. For PKNL method, the aerodynamic matrices are split into real ($Q^R{}_{hh}$) and imaginary parts ($Q^I{}_{hh}$), corresponding to modal aerodynamic stiffness and damping matrices, respectively. These matrices contain only real terms.

The fundamental equation of the PKNL method is given in terms of modal parameters and flight condition, according to Eq. (2).

$$\left[M_{hh} p^2 + \left(B_{hh} - \frac{1}{4}\rho c V\, Q^I_{hh} / k \right) p + K_{hh} - \frac{1}{2}\rho V^2 Q^R_{hh} \right] \{u_h\} = 0 \tag{2}$$

The u_h corresponds to modal amplitude and γ represents the transient decay rate coefficient. The M_{hh}, B_{hh} and K_{hh} are the modal mass, damping and stiffness matrices. For achieving accurate aeroelastic results, the ten first modes are taking into account in the simulation. However, only the three first modes are considered for validation in the dynamic tests.

From the eigenvalue problem, described by Eq.(2), the complex root $p = \omega(\gamma \pm i)$ is determined. The dynamic characteristics of the aeroelastic system, such as frequency and damping, are computed as follow:

$$\omega = \mathrm{Im}(p) \tag{3}$$

$$g = 2\gamma = 2\,\mathrm{Re}(p) / \mathrm{Im}(p) \tag{4}$$

Based on the values of ω and g, it is possible to verify the stability of the dynamic system (stable for negative values of g). The critical flutter and divergence speeds are obtained in the vicinity of the unstable responses (g=0). While flutter is characterized by the coalescence between two vibration modes (fluid-structure problem), the divergence is determined when the natural frequency tends to zero.

3. EXPERIMENTAL VALIDATION

3.1 Static test

The static tests were conducted to validate the structural finite element model developed in Nastran®, by comparing the deflections measured in the static set up (Fig. 5) and the position of the elastic axis with the output results from numerical analysis.

Concerning the measurements of the wing deflections, the apparatus was mounted pinning the wing at the seven rib counted from the wing root in both sides and applying the load in the middle of the structure. The deflections were measured along the wing semi span direction in nine points, utilizing a dial indicator.

With respect to the validation of elastic axis, the load was applied at different points along the chord direction measuring the displacement at leading edge and trailing edge to find the exact position where the structure is twist free.

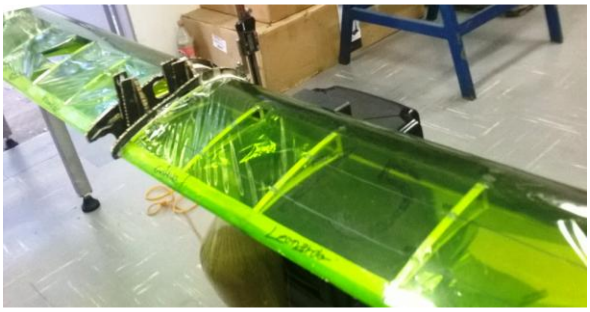


Figure 5. Static test assembly

3.2 Dynamic test

This section describes the experimental approach developed for validating the modal analysis of the reduced scaled aircraft wing.

Elastic threads, attached to the wing chord root (see Fig. 6), supports the wing weight. The rigid body modes are not interesting and they are neglected from the experimental results. For FEM modal analysis (according to Section 2.1) the real structure is modeled by a free-free dynamic configuration.

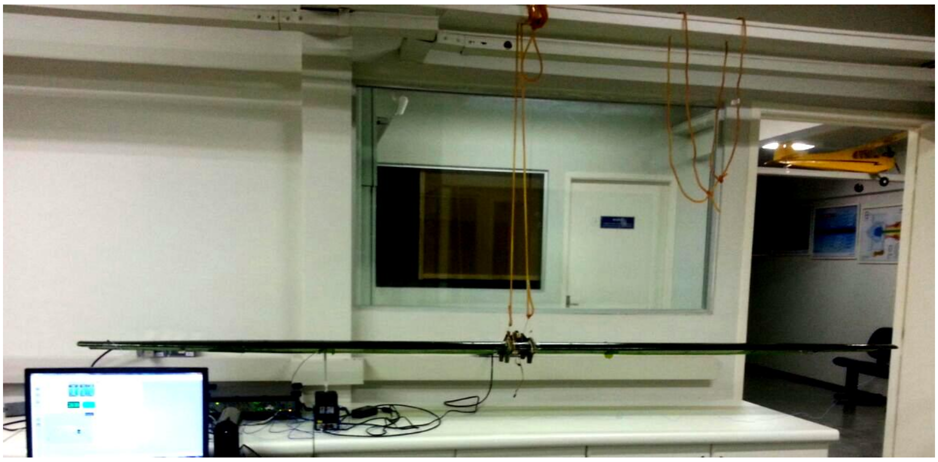
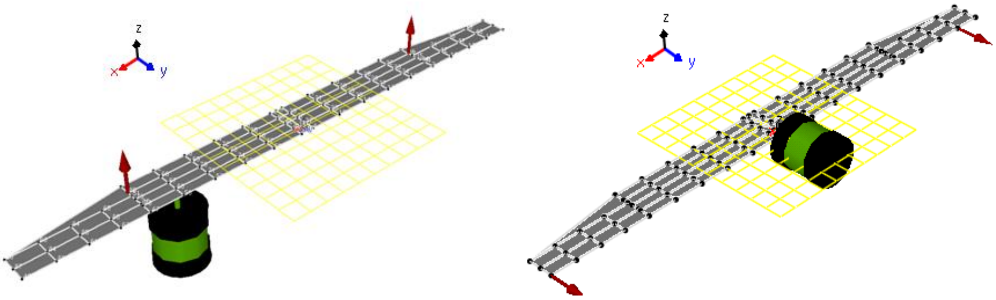


Figure 6. Dynamic experimental setup configuration

In this setup, the K2007E01 TMS electromagnetic shaker excites the structure with a white noise signal of constant amplitude (9 $m\mathrm{V}_{rms}$), based on the signal generated from PULSE LAbShop®. The 208C03 PCB Piezotronics load cell mounted in the shaker measures the effective force level transmitted to the structure; while the 352C22 PCB Piezotronics accelerometer (positioned on the structure) measure the acceleration signals.

The position and orientation of the electromagnetic shaker and accelerometers are defined according to Fig. 7, determined based on previous FEM modal analysis. Eighty-four points along the whole wing are select to measure the vibration in the vertical direction (z-axis) and thirty-eight points distributed along the wing span direction (x-axis).



(a) (b)

Figure 7. Measurement system positioning (a) z-direction test; (b) y-direction test

At each measurement point, the experimental frequency response function (FRF) was obtained by considering the linear averaging from forty data. The frequency range of the FRF's is from 0 up to 100 Hz for the z-direction measurements and from 0 up to 50 Hz for the y-direction measurements. A low-pass filter was adjusted in the signal processing, to ensure signal coherence at low frequency. The cutoff frequencies are set as 100Hz and 50 Hz from measurements along out-of-plane direction and in-plane direction, respectively. In both cases, the frequency range is divided along 800 lines.

Afterwards, the FRF data was exported to software Me'scope® for the post-treatment of experimental results. The Complex Mode Indicator Function (CMIF) method was applied to identify (based on the imaginary part of the FRFs) a modal database for the mechanical system. This method is based on singular value decomposition of multiple reference FRF measurements (Leurs, 1996). Also, it was used a polynomial interpolation of order six to fit the data curve in the area delimited by red traces, shown in Fig. 9.

4. RESULTS

Table 3 comprises the model validation in terms of displacements, where the measured points are shown in Fig. 8.

Table 3. Static test and numerical analysis results.

Point	Load [kg]	NASTRAN® Displacement [m]	Experimental Displacement [m]	Error [%]
1	2.12	0.0101123	0.01018	0.66
2		0.009834	0.00934	5.29
3		0.0093397	0.00916	1.96
4		0.0090478	0.00877	3.17
5		0.0059518	0.00613	2.91
6		0.0055632	0.00539	3.21
7		-0.0036263	-0.00398	8.89
8		-0.0036626	-0.00351	4.35
9	5.13	0.025909	0.02580	0.42

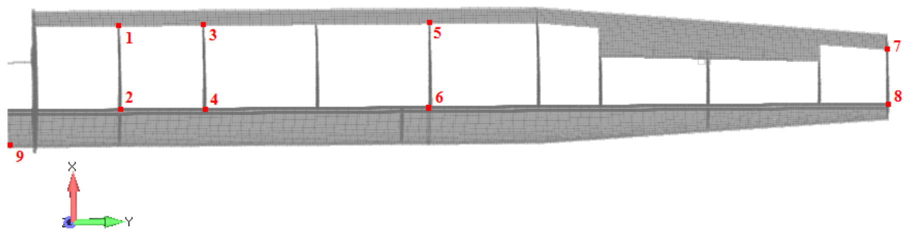
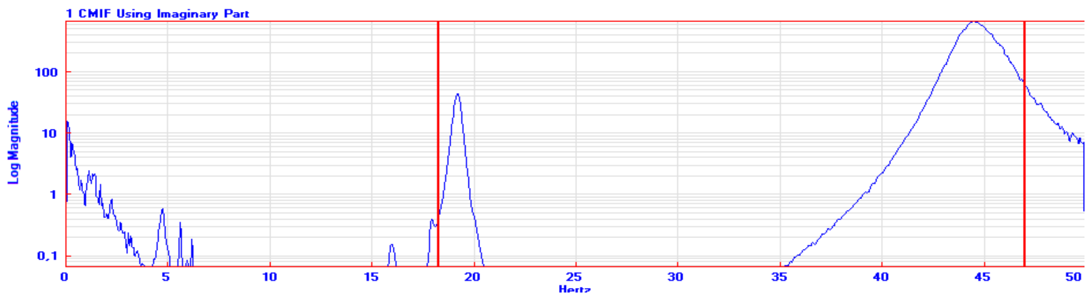
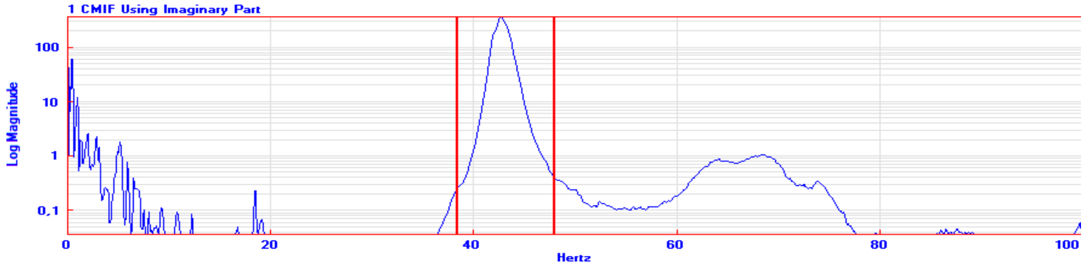


Figure 8. Points measured localization in static analysis.

The CMIF method identified three modal shapes: two bending modes on the z-direction test and one in plane mode on the y-direction test, as presented in Fig. 9. Regarding the deformation patterns at the natural frequencies, the modal shapes are obtained by interpolating each point measured with the five nearest points.



(a)



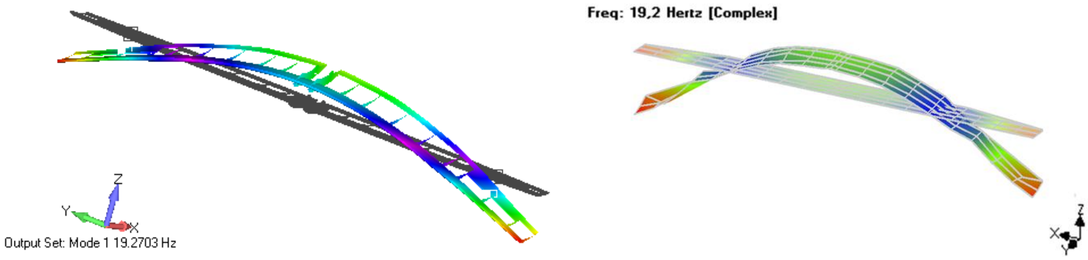
(b)

Figure 9. Interval for curving fitting in CMIF (a) bending-pitching test; (b) in plane test.

Figures 10, 11 and 12, in addition to Tab. 4 presents the mode shapes comparison obtained in FEM and Dynamic test.

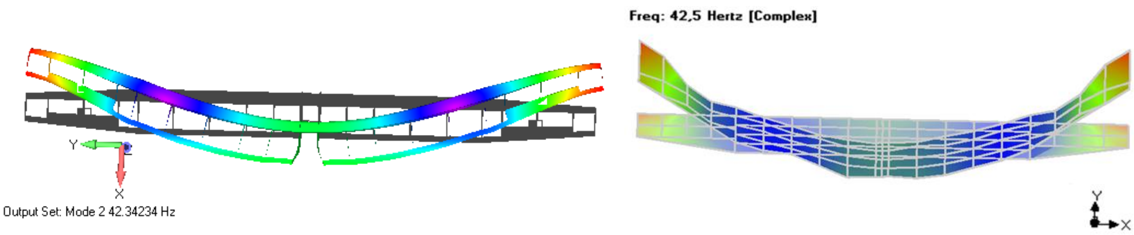
Table 4. Dynamical test and numerical analysis results.

NASTRAN® Natural Frequency [Hz]	Experimental Natural Frequency [Hz]	Error [%]
19.2703	19.2	0.37
42.3423	42.5	0.37
44.5680	44.5	0.15



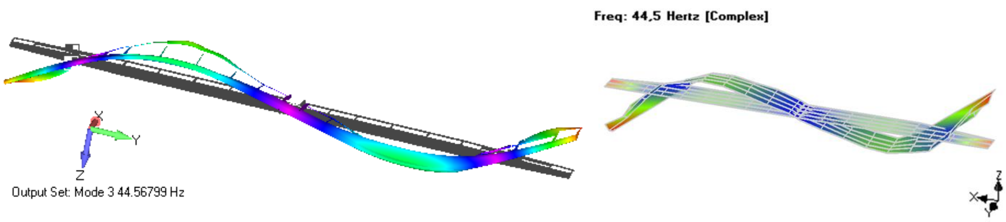
(a) (b)

Figure 10. 1^{st} bending shape (a) FEM Nastran® - 19.2703 Hz; (b) experimental Me'scope® - 19.2 Hz.



(a) (b)

Figure 11. In plane shape (a) FEM Nastran® - 42.3423 Hz; (b) experimental Me'scope® - 42.5 Hz.



(a) (b)

Figure 12 . 2^{nd} bending shape (a) FEM Nastran® - 44.5680 Hz; (b) experimental Me'scope® - 44.5 Hz.

The aeroelastic analysis obtained from Nastran® (.f06 text file) were post treated and the graphical results of the v-g and v-f diagrams are given in terms of Equivalent Airspeed, according to Fig. 13. It is shown that the aeroelastic instability occurs at 101.7 m/s EAS, once the damping of Mode 1 turns to a positive value. Simultaneously, the natural frequency of Mode 1 goes to zero characterizing a divergence behavior.

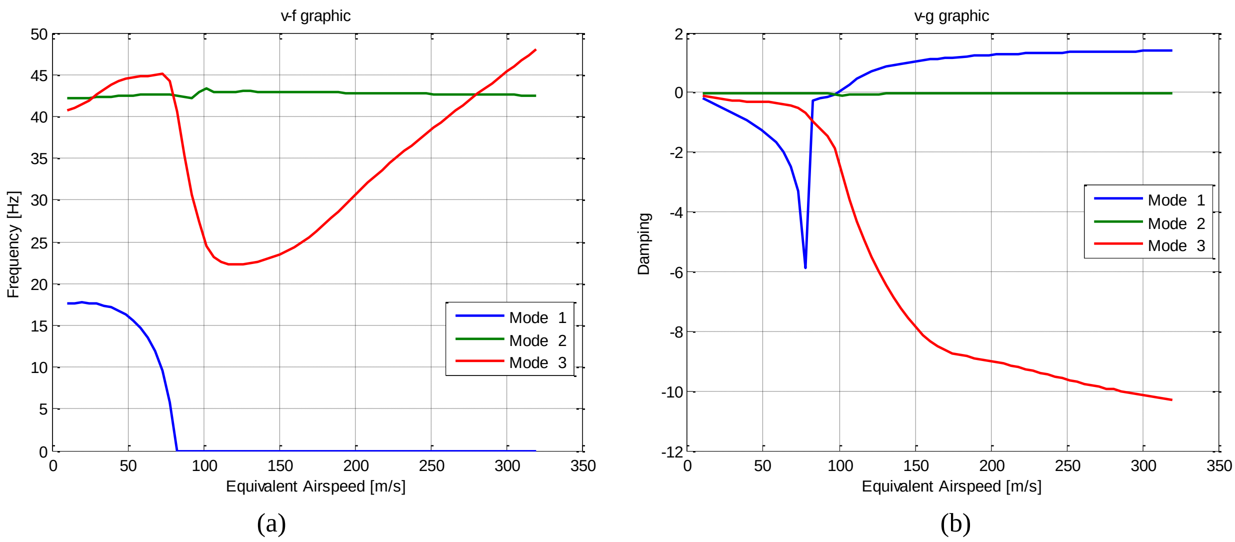


Figure 13 –Nastran® SOL145 results.

5. CONCLUSIONS

The initial goal of the presented study was to analyze the aeroelastic behavior of composite aerodesign wing. It was necessary, at first moment, to validate the structural numerical model (FEM) with the experimental static and dynamic analyses. The main material properties were adjusted to minimize the error between numerical and experimental model in both setups. A good agreement between both results were obtained, which the maximum error of 0.37% for the natural frequencies.

With numerical model adjusted, then an aeroelastic analysis was carried out. The unstable wing behavior (divergence phenomenon) was predicted to occur at 105 m/s (true airspeed). Therefore, the aircraft is assured to be out of aeroelastic problems, since its maximum operating speed is 30m/s.

6. ACKNOWLEDGEMENTS

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7. REFERENCES

Bisplinghoff, R. L., Ashley, H, Halfman, R.L., 1996. *Aeroelasticity*. Dower Publications, USA.

Blair, M., 1992. *A Compilation of the Mathematics Leading to a Doublet Lattice Method*. Flight Dynamic Directorate, Airforce Wright Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, USA.

Hodges, D.H., Pierce G.A., 2002. *Introduction to Structural Dynamics and Aeroelasticity. Cambrigde*. University Press

Leurs, w., Deblauwe, F. and Lembregts, F., 1996. *Modal Parameter on Complex Mode Indicator Functions*. LMS International Interleuvenlaan 68, Leuven, Belgium.

MSC Software, 2013. *MSC Nastran Quick Reference Guide*. USA.

Peeters, B., Climent, H., Diego, R., Alba, J., Ahlquist, J.R., Carreño, J.M., Hendricx, W., Rega, A., García, G., Deweer, J., Debille, J., 2008. *Modern solutions for Ground Vibration Testing of large aircraft*. LMS International Interleuvenlaan 68, Leuven, Belgium.

Rade, D.A., 2013. *Introdução ao Método de Elementos Finitos*. Federal University of Uberlândia, School of Mechanical Engineering, Brazil.

Rade, D.A., Steffan, V., 2010. *Structural Dynamics and Modal Analysis.* Federal University of Uberlândia, School of Mechanical Engineering, Brazil.

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