

The Use of an Open Source Finite Element Code for Aeroelastic Analyses

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Abstract: *The present paper intends to evaluate the utilization of some open source finite element code for aeroelastic analyses. Usually, aeroelastic analyses use linear approximations in order to predict instabilities due to fluid-structural interactions. Nowadays, it is possible to include nonlinear aerodynamic effects on the analyses using Computational Fluid Dynamics techniques. The aerodynamic phenomena are better represented in this way, nevertheless, it is necessary a huge mesh to represent the environment. In another way, the mesh to represent the dynamics of the structure is simpler than the aerodynamic mesh. There are several codes based on the Finite Element Method which are able to solve dynamic analyses using this sort of mesh. The coupling between a huge mesh and the simple one is not trivial and the utilization of independent codes (aeroelastic and structural code) is inefficient due to excessive computational time for exchanging information through files. As an alternative, it is possible to include the Structural Finite Element solver into an aeroelastic analysis framework which already incorporates the aerodynamic code. There are traditional routines implemented in open codes for Finite Element Analyses. In order to evaluate Finite Element codes to be used, the a classical wing model was tested using the commercial FEA code and the open one. The natural frequencies and modal shapes are used to study the aeroelastic evolution of this wing.*

Keywords: *Aeroelasticity, Finite Element Method, Flutter*

1. INTRODUCTION

Flight is a faster and safer way to travel. Several physic phenomena are controllable in certain circumstances. Some of uncontrollable one needs to be avoided. In order to predict the flight performance without instabilities, models are idealized to describe the phenomena based on the aerodynamic around the aircraft, its flexibility and its dynamic behavior.

Aeroelastic instabilities, for example, are investigated since the first accidents Garrick and Reed (1981). Methods as k or pk that assume an inviscid linearised theory in the frequency domain for the unsteady aerodynamics are traditional way to predict aeroelastic instabilities. In the transonic regime with mixed subsonic and supersonic regions, linear aerodynamic theories fail due to the presence of flow nonlinearities such as shock waves and shock induced flow separation. Furthermore, minimum flutter speeds, buffeting, limit-cycle-oscillations (LCO), aileron buzz, and shock-boundary layer oscillations may be encountered (Bennett and Edwards (1998)).

With the development of increasingly powerful computers, numerical simulations of various approximations to the Navier-Stokes equations began supplementing these tools. These numerical simulation methods became known as Computational Fluid Dynamics (CFD) (Johnson et al. (2003)). When CFD is coupled with the structural dynamics in the computational process, it is generally referred to as Computational Aeroelasticity (CA). One of the difficulties with CA is that aircraft must be cleared for flutter significantly beyond cruise conditions where strong shocks, separated boundary layers, and other flow conditions which are difficult to compute, may be encountered (Bennett and Edwards (1998)).

In the past, CA has required long solution times on expensive supercomputers. The workstation-type machines have attained the performance level of the supercomputers of the previous decade and the cost of the computation has decreased by between two and three orders of magnitude. Nowadays, multidisciplinary design problems which can involve as many as 10^5 to 10^6 evaluations of the flutter eigenvalue matrix for stability. CA is impractical without further development of innovative technologies (Bennett and Edwards (1998)).

The simulation CA technology is now becoming powerful and efficient enough to begin making its impact on the actual design and certification of flight vehicles (Demasi and Livne (2008)). The three major elements of most CA methods involve preprocessing of the geometry and modal data, the execution of the CFD computer program, and the postprocessing of the output data. Of course at all stages of the process, plotting and monitoring of all data is essential both for assuring that the data are error free, and that the results are properly converged. The plotting and monitoring tasks can be very time consuming and a high level of automation is desired. For mode processing which requires interpolation from the vibration-modal grid to the aerodynamic grid, we have generally used the surface spline. For the small disturbance code that requires modal slopes as well as amplitudes, some of the limitations of this spline can be overcome for cases using calculated vibration modes by splining the modal rotations rather than using the slope of the spline. This capability is basically developed for wings only. The development of a suitable method for including fuselage or bodies may require significant effort (Bennett and Edwards (1998)).

Traditionally, due to the computation complexity and computer technology limitations, a complete aeroelasticity system was analysed using CFD and Computational Structural Dynamic (CSD) softwares separately by different working groups and then using data communication between them (Feng et al. (2008)). The algorithms of continuum mechanics usually make use of two classical descriptions of motion: the Lagrangian description and the Eulerian description. La-

grangian algorithms, in which each individual node of the computational mesh follows the associated material particle during motion, are mainly used in structural mechanics. The Lagrangian description allows an easy tracking of free surfaces and interfaces between different materials. It also facilitates the treatment of materials with history-dependent constitutive relations. Its weakness is its inability to follow large distortions of the computational domain without recourse to frequent remeshing operations. Eulerian algorithms are widely used in fluid dynamics. The computational mesh is fixed and the continuum moves with respect to the grid. In the Eulerian description, large distortions in the continuum motion can be handled with relative ease, but generally at the expense of precise interface definition and the resolution of flow details (Donea and Rodriguez-Ferran (2004)). The numerical simulation of multidimensional problems in fluid dynamics and nonlinear solid mechanics often requires coping with strong distortions of the continuum under consideration while allowing for a clear delineation of free surfaces and fluid-fluid, solid-solid, or fluid-structure interfaces. A fundamentally important consideration when developing a computer code for simulating problems in this class is the choice of an appropriate kinematical description of the continuum. In fact, such a choice determines the relationship between the deforming continuum and the finite grid or mesh of computing zones, and thus conditions the ability of the numerical method to deal with large distortions and provide an accurate resolution of material interfaces and mobile boundaries (Donea and Rodriguez-Ferran (2004)). The simultaneous solution of nonlinear fluid, fluid-mesh, and structural equations of motion is computationally intensive. It has raised some concerns about the feasibility and practicality in production environments of the three-field formulation of nonlinear aeroelastic problems (Bennett and Edwards (1998)).

This paper intends to evaluate the performance of FEM open-source routines in order to use the extracted mode shapes as an input of a commercial software dedicated to aeroelastic analyses. This first step expects to obtain the aeroelastic evolution behaviour of wing used for educational purposes at Instituto Tecnológico de Aeronáutica.

2. Theoretical Background

2.1 Finite Element Method

Describing the dynamic wing problem by Lagrangian Formalism in function of generalized coordinates \mathbf{q} as in (Meirovitch, 1980):

$$\frac{d}{dt} \left(\frac{\partial L}{\partial \dot{\mathbf{q}}} \right) - \frac{\partial L}{\partial \mathbf{q}} = \mathbf{F} \quad , \quad (1)$$

where the Lagrangian L is expressed in terms of Kinetic energy T and potential energy U as Eq. 2 and the vector \mathbf{F} is generalized effort submitted to the wing:

$$L = T - U \quad . \quad (2)$$

Replacing the T and U in terms of interpolated shape function \mathbf{N} , satisfying the Dirichlet boundary conditions, it is possible to write the equation of motion using the Finite Element Method, according Cook et al. (1989)

$$[\mathbf{M}]\ddot{\mathbf{q}} + [\mathbf{K}]\mathbf{q} = \mathbf{F} \quad , \quad (3)$$

where the mass matrix \mathbf{M} can be expressed, now, in terms of function of Shape Function \mathbf{N} and Inertia components Matrix \mathbf{In} of each discretized element. In similar way stiffness matrix with the Constitutive Material Properties Matrix \mathbf{C} are

$$\mathbf{M} = \int_{Vol} \mathbf{N}^T [\mathbf{In}] \mathbf{N} dVol \quad \text{and} \quad \mathbf{K} = \int_{Vol} \mathbf{N}^T [\mathbf{C}] \mathbf{N} dVol \quad . \quad (4)$$

The shape functions are defining according the element topologies. The beam element has unidimensional shape functions and plate element has bi-dimensional shape functions, for example. The shape function are, normally, interpolation polynomial satisfying the boundary conditions of each nodes.

2.2 Aeroelastic Analyses

Aerodynamic forces \mathbf{F}_a acting on the flexible wing dependent on the wing displacements. The dynamic system described in Eq. 3 can be expressed as

$$[\mathbf{M}]\ddot{\mathbf{q}}(t) + [\mathbf{K}]\mathbf{q}(t) + \mathbf{F}_a(\mathbf{q}) = \mathbf{F}_e \quad , \quad (5)$$

where \mathbf{F}_e is the exogenous force.

The aerodynamic forces \mathbf{F}_a may be expressed in terms of dynamic pressure q_∞ and, in unsteady aerodynamic potential approach

$$\mathbf{F}_a = q_\infty \mathbf{Q}(i k) \mathbf{q} \quad . \quad (6)$$

The unsteady aerodynamic potential matrix $\mathbf{Q}(i k)$ contents the Aerodynamic Influence Coefficient Matrix relating the aerodynamic influence coefficients $[AIC(i k)]$ among the discretized mesh pannels , mesh-structure spline interpolated matrix \mathbf{G} :

$$\mathbf{Q}(i k) = \mathbf{G}^T [AIC(i k)] \mathbf{G} \quad . \quad (7)$$

This relationship is a function of reduced frequency, k , a dimensionless number that relates the physics dynamic properties of fluid structure interactions and defines the degree of unsteadiness of the wing:

$$k = \frac{\omega b}{V_\infty} \quad , \quad (8)$$

where ω is the circular frequency submitted to structure, b is the semichord reference and V_∞ is the flow velocity.

In order to satisfy the structural stability boundary condition of the Eq. 5, when $\mathbf{F}_e = \mathbf{0}$, in frequency domain and modal space

$$[-\omega^2 \bar{\mathbf{M}} + \bar{\mathbf{K}} - q_\infty \bar{\mathbf{Q}}(i k)] \bar{\mathbf{q}} = \mathbf{0} \quad ; \quad (9)$$

with Modal Mass Matrix, $\bar{\mathbf{M}}$, Modal Stiffness $\bar{\mathbf{K}}$, Modal Aerodynamic Generalized Force $\bar{\mathbf{Q}}(i k)$:

$$\bar{\mathbf{M}} = \Phi^T \mathbf{M} \Phi \quad , \quad \bar{\mathbf{K}} = \Phi^T \mathbf{K} \Phi \quad , \quad \bar{\mathbf{Q}}(i k) = \Phi^T \mathbf{Q}(i k) \Phi \quad ; \quad (10)$$

an artificial structural damping g_s is added to this equation.

$$[-\omega^2 \bar{\mathbf{M}} + (1 + i g_s) \bar{\mathbf{K}} - q_\infty \bar{\mathbf{Q}}(i k)] = \mathbf{0} \quad . \quad (11)$$

When this artificial damping change from negative to positive values, the dynamic system expressed in equation becomes unstable, indicating the flutter phenomena.

There are several methods to define where this equation is stable. Essentially, they search values to reduced frequency k that the artificial damping is negative for all considered modes. The main modes to be account in these solution can be extracted from a normal mode Finite Element Analyses. ZAERO-III, a dedicate aeroelastic solver, for example can use output format from NASTRAN, ANSYS or Abaqus, as normal mode input (Zona Technology Inc. (2007)).

3. METHODOLOGY

In order to evaluate the open source Finite Element solver for aeroelastic analyses, a wing with ballast on the tip is used as presented in the Fig. 3.2. The wing consists a simple aluminium beam with rectangular section and 0.8 mm thick, it has 320.0 mm span and 44.0 mm chord. The wing is clamped at root and a 90.0 mm cylindrical brass ballast with 8 mm diameter is fixed on the tip. This ballast is misaligned d far from elastic wing axis. This distance determines how the bending torsion wing modes are coupled. Three d values presented on Tab. 1 are used in order to compare the modal solution obtained by different open source Finite Element solvers and commercial ones with results extracted from experimental modal analyses, EMA.

3.1 Experimental Modal Analysis - EMA

In order to extract the natural frequencies for all ballast conditions without instrumentation influences, an impact test was perform according Ewins (1986) using a dynamometric hammer. The wing is extremely thin, then two laser vibrometer are employed to capture the velocity signals, instead of accelerometers. The Frequency Response Function (FRF) Matrix is estimated using a signal acquire system during the test.

The natural frequencies and mode shapes are obtained in Frequency Domain through the Global Rational Fraction Polynomial Method, GRFP, described in more details in Maia et al. (1997). This method adjust complex polynomials with order n to fit the FRFs curves. The stables poles, from one polynomial order to the other indicate the natural frequencies existence.

3.2 Finite Element Analysis

The wing is discretized using plate element and the ballast as beam element. Two open-source FEM codes (Calculix version 2.8 and Nastran NAS95) and two commercial one (AbaqusTM and NX-NASTRANTM) are used to calculate the eigenvalue problem and determine the natural frequencies and respective modal shapes. The plate element and beam element are described with different Finite Element approaches. The elements considered are isotropic and the material properties are expressed in the Tab. 2 .

The beam element and plate element, unidimensional and bidimensional one, used in Calculix and Abaqus are transformed in a three dimensional one. The beam element B32 (Dhondt (2015)) considered in the code is three master uniaxial node. These node are replaced to 20 nodes. The four node shell element SR4 (Dhondt (2015)) relates to the plate elements are also replaced by 20 nodes. The elements used in the Nastran in order to represent the ballast is CBAR, also unidimensional element, but modelled with two nodes. The element used to represent the wing is CQUAD4 with four nodes. The eigenvalues and eigenvectors are calculated using Lanczos algorithm in all finite element solvers.

Tabela 1: Position of ballast relate to the elastic wing axis

Position	d ₀	d ₁	d ₂	d ₃
distance [mm]	0.00	5.00	10.00	15.00

Tabela 2: Material properties of aluminium and brass

	ρ kg/m^3	E GPa	ν 1
Aluminium	2700	69.0	0.3
Brass	8450	97.0	0.3

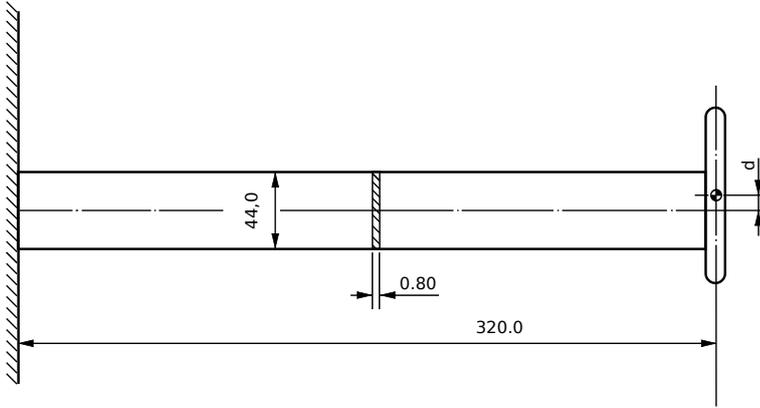


Figura 1: A Simple rectangular wing with a ballast on the tip.

3.3 Aeroelasticity analysis

The aeroelastic evolution are analyzed using ZAERO-III, an aeoroelastic dedicated commercial software. The wing is discretized in aerodynamic panels. The aerodynamic enviroment is subsonic with constant air density $1.225 kg/m^3$. The wind speed analysed are expressed in Tab. 3 .

Tabela 3: Wing speed used in the aeroelastic analyses.

V_∞ [m/s]	0.5	1.0	2.0	3.0	4.0	5.0	6.0	8.0
	10.0	12.0	14.0	16.0	18.0	20.0	22.0	24.0

The reduced frequencies analyzed are adjusted to cover all natural frequencies of modes and wind speed considered according Eq. 8. The greatest natural frequency of interest identified is below $500 Hz$. The reduced frequencies considered to cover all natural frequencies are expressed in the Tab. 4 . The aeroelastic evolution for each analyzed using mode shapes extracted from different FEM approach are compared.

Tabela 4: Reduced frequencies used in the aeroelastic analyses.

k [1]	0.05	0.10	0.20	0.30	0.40	0.50	0.60
	0.70	1.00	2.00	3.00	4.00	5.00	6.00

4. DISCUSSION OF RESULTS

The natural frequencies obtained using several finite element solvers are summarized in Tab. 5. These frequencies are according with same results obtained experimetally captured with stable poles as presented by the circle marker in the Fig. 2 . The normal modes obtained by all FEM codes represent perfectly the modal shapes as can be seen in the Fig. 3 . The normal modes could be used as ZAERO-III input and evaluate the aeoroelastic evolution for each case analyzed.

Comparing the natural frequencies according their formulation approaches, Abaqus-Calculix and NX NASTRANTM-NAS95, it is observed that the results obtained in the last one are closer, indicating a good correlation between these codes. Nevertheless, the aeroelastic evolution obtained using their mode shapes could not capture the instabilities, neither the coalescence tendency among the modes as can be observed in the Fig. 6 and Fig. 7.

The expected aeoroelastic behaviour is observed when it used the mode shapes from Abaqus and Calculix as can be observed in the Fig. 4 and Fig. 6. In this figures, for each case, flutter occurs with expected wind speed and the coalescence tendency between the second mode and third mode. Regarding the slopes for each ballast case it can be observed the intensity of flutter for each cases.

Tabela 5: Comparison of Natural frequencies of the wing obtained experimentally and with different Finite Element solver.

Position	Mode	Experimental	Abaqus TM	Calculix	NX-NASTRAN TM	NASTRAN
d ₀	1	2.66	2.21	2.67	2.59	2.59
	2	27.31	26.11	30.44	22.64	22.70
	3	29.42	28.86	33.22	29.47	29.51
	4	-	84.01	97.83	93.02	93.30
d ₁	1	2.62	2.21	2.67	2.59	2.59
	2	26.89	25.97	30.24	22.47	22.53
	3	29.84	28.97	33.38	29.55	29.59
	4	-	84.05	97.88	93.06	93.34
d ₂	1	2.65	2.21	2.67	2.59	2.59
	2	26.07	25.61	29.76	21.99	22.04
	3	30.54	29.26	33.78	29.76	29.80
	4	-	84.17	98.03	93.17	93.45
d ₃	1	2.59	2.21	2.67	2.59	2.59
	2	25.39	25.12	29.12	21.28	21.33
	3	31.14	29.63	34.29	30.04	30.08
	4	-	84.36	98.28	93.35	93.62

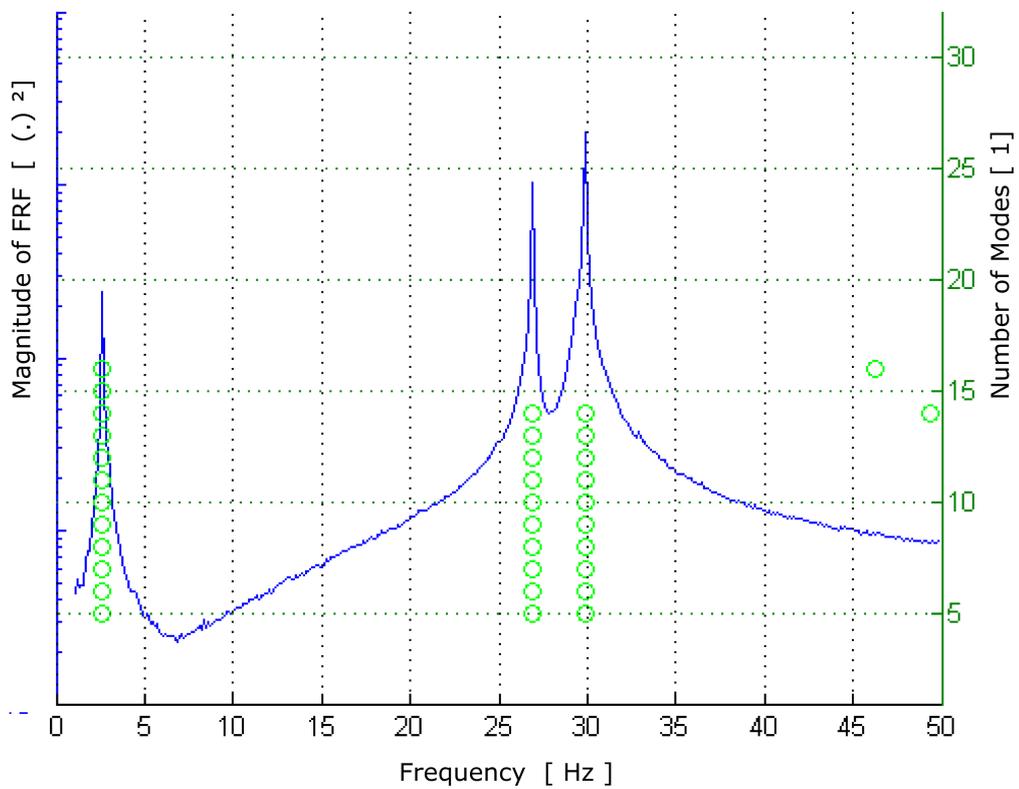


Figura 2: Stabilization diagram.

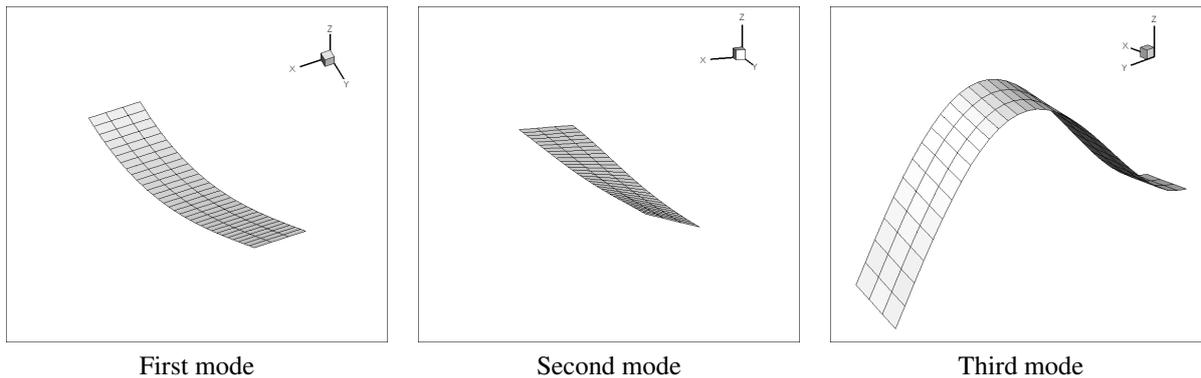


Figura 3: The first three modes of the rectangular wing.

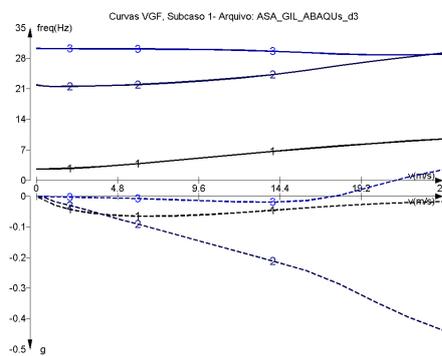
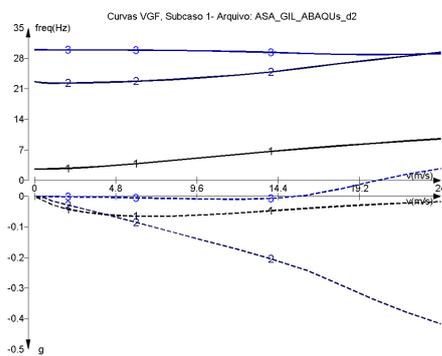
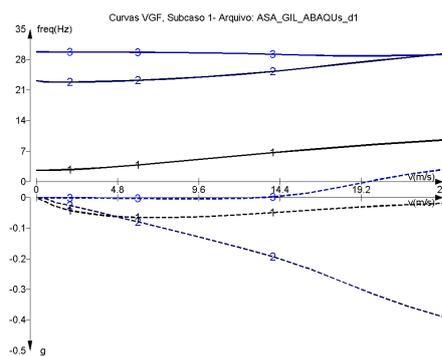
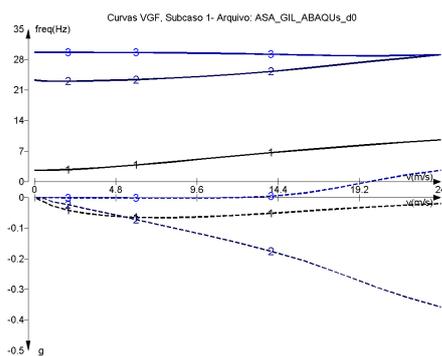
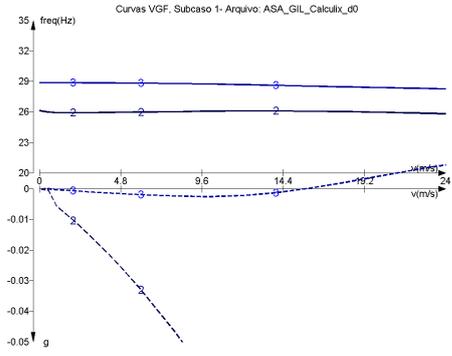
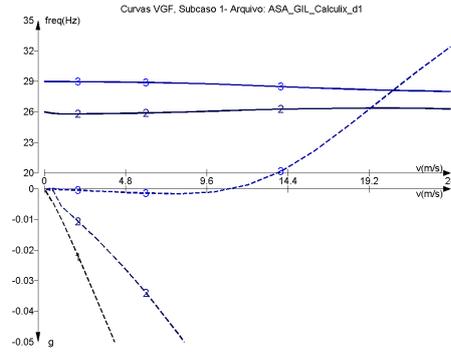


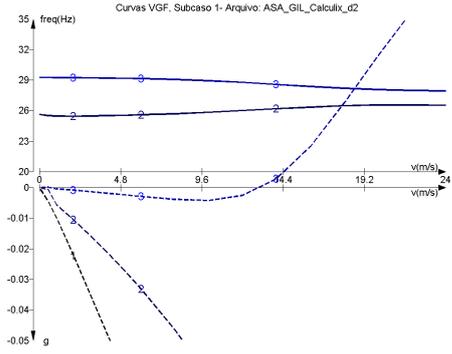
Figura 4: Aeroelastic evolution using mode shapes from AbaqusTM.



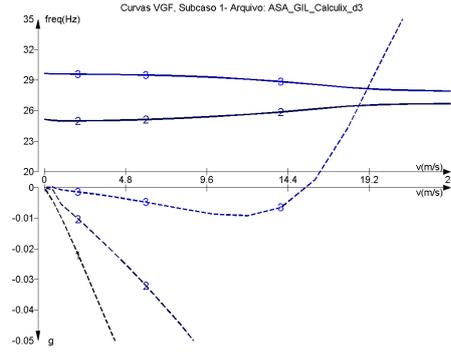
(d₀)



(d₁)

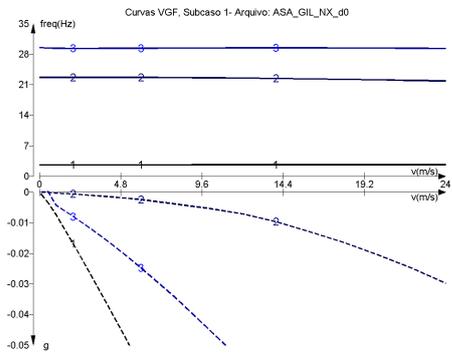


(d₃)

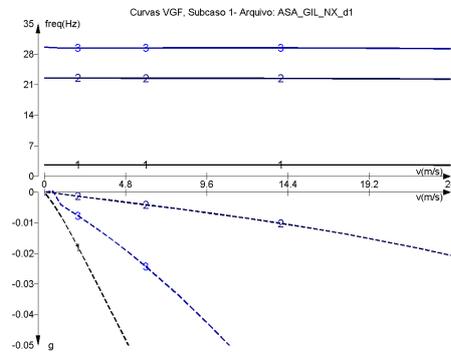


(d₄)

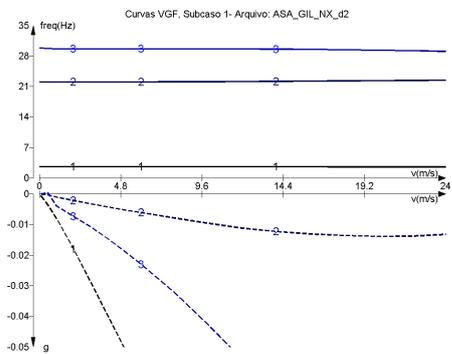
Figura 5: Aeroelastic evolution using mode shapes from Calculix.



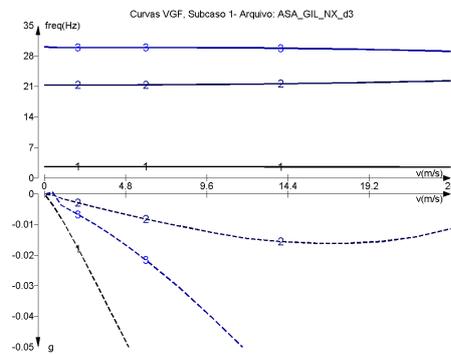
(d₀)



(d₁)



(d₃)



(d₄)

Figura 6: Aeroelastic evolution using mode shapes from NX NASTRANTM.

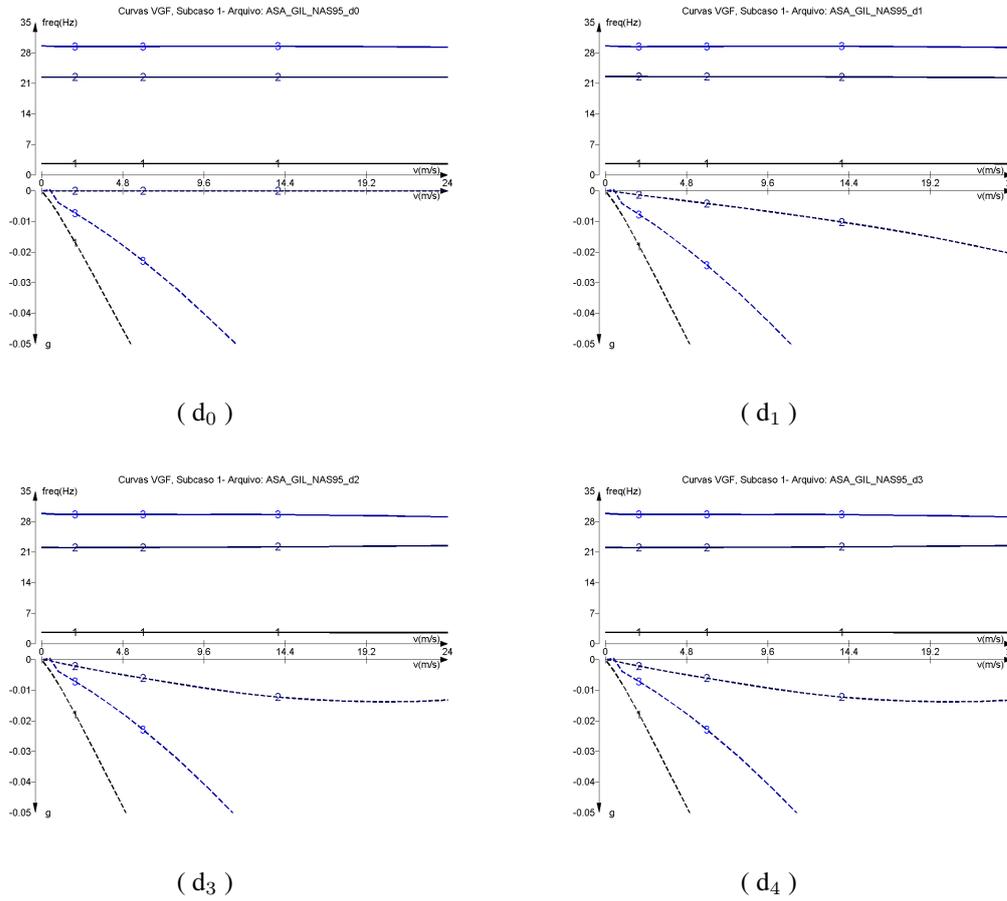


Figura 7: Aeroelastic evolution using mode shapes from NAS95.

5. CONCLUDING REMARKS

The structural model to be used in the aeroelastic analyses can be idealized with appropriated dynamic representative characteristics, such natural frequencies and respective mode shapes, using the finite element approach with open source codes.

The aeroelastic evolution has more sensibility to the modal shapes than the natural frequencies. In this way, in order to obtain more realistic results, the extracted modal shapes need to be normalized according the aeroelastic formulation. For example, with ZAERO-IIITM the normalization is Mass Normalization. In order to employ these codes for aeroelastic analyses, it is necessary to verify how the FE element codes normalize the modal shapes.

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