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FEM MODAL ANALYSIS FOR DAMAGE DETECTION IN AIRPLANE STRUCTURES

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Abstract. *Technics for damage and fatigue detection has become an essential part of Aeronautic industry safety. The duration of service lives of such vehicles can be considerably extended by systematic monitoring of small, possibly undetected, damages to their structures. In this paper, we propose damage detection in airplane structures via modal analysis. By using Finite Element Method numerical models, developed in commercial software, as a first step, we make comparisons of the free vibration frequencies of undamaged models and models where we deliberately introduce damage. We observe the difference in frequency values. In future work we will propose numerical damage indicators based on the modal analysis.*

Keywords: *Aeronautical Structure, Damage Detection, Modal Analysis, FEM.*

1 INTRODUCTION

Structural Health Monitoring (SHM) has become an essential technique in Aerospace industry. The duration of service lives of aircrafts and space vehicles can be considerably extended by systematic monitoring of small, possibly undetected, damages to their structures. Also, this monitoring can prevent accidents and contribute to flying safety. A historical example of such undetected damage is the 1950's infamous Comet 1 British jet airliner.

In this paper, we propose ways of monitoring damage in airplanes via modal analysis by using Finite Element modeling, as a first step in this direction. The idea is to find a noninvasive a nondestructive method to evaluate possible damage present on aeronautical vehicles.

We make comparisons between airplane structures by adopting internet available real FE elements models of airplanes, and determining its free vibrations frequencies and modes for the undamaged situation. Next, we deliberately introduce damage (simulated by holes, changes in Young's module etc.) in the structures and new modal analyses are performed.

The afore mentioned comparative modal analysis shows that the free vibrations frequencies of a damaged aircraft parts are less than those of an undamaged one. We conclude that the damage, depending on its extents, results in changes in the free vibrations frequencies. In future work, we will test some numerical indicators, described in the literature, based on modal analysis, to discover damage. We also intend to investigate which of the obtained modes are useful for the analysis we are performing and which modes present inconsistent results.

2 THEORETICAL DEVELOPMENT

In this work, we are interested in the free undamped vibrations of aeronautical structures, to enable us to determinate the natural frequencies and vibration mode shapes. The equations of motion of free undamped vibrations are:

$$[M]\{U''\} + [K]\{U\} = \{0 \tag{1}$$

where,

- [M] - System Mass Matrix.
- {U´´} - Accelerations Vector.
- [K] - System Stiffness Matrix.
- [U] - Displacements Vector.

Differently from a system with only one degree of freedom, Eq. (1) is represented with matrices and vectors for we are dealing with several degrees of freedom. The order of the matrices will depend on the number of degrees of freedom adopted in the system. For instance, for a system with N degrees of freedom, the matrices will have the order of NxN and the vectors Nx1 (Juliani, 2014).

Considering the free vibration motion, in the same way as in a system with one degree of freedom, the dynamic response can be expressed as:

$$\{U(t)\} = \{\hat{U}\}\cos(\omega t - \Theta) \tag{2}$$

Deriving for the second time Eq. (2) and substituting in Eq. (1) we generate the eigenvalues and eigenvectors problem:

$$[[K] - \omega^2[M]\{\hat{U}\}] = \{0\} \tag{3}$$

where:

- ω^2 - Eigenvalues that represent the squares of the circular frequencies (rad/s);
- $\{\hat{U}\}$ - Eigenvectors that represent the corresponding dimensionless displacements (shape) of their respective modes (Clough and Penzien, 2003).

Equation (3) is a set of algebraic linear homogenous equations and the non-trivia solution of the equation is only possible if (Anderson and Naeim, 2012):

$$|[K] - \omega^2[M]| = 0 \tag{4}$$

The determinant in Eq. (4) results in a N-th order polynomial equation, having as unknown the parameter ω^2. The N roots represent the natural frequencies of the N modes of vibration adopted in the system (Clough and Penzien, 2003). The smallest frequency value calculated corresponds to the first vibration mode, the second to the second vibration mode and so on, successively.

To obtain the vibration modes we solve:

$$[E^n]\{\hat{U}_n\} = \{0\} \tag{5}$$

where

$$[E^n] = [K] - \omega_n^2[M] \tag{6}$$

Matrix $[E^n]$ varies with each frequency and so it is different for each mode. This way, for each ω_n, a vibration mode may be calculated with Eq. (5). But, in this system, one of the equations is redundant, so the vibration modes are found by arbitrarily giving a value to one of the displacements (usually unitary).

With the advances of computation and the use of programs, all the formulation presented here can be rapidly solved using iterative methods, whose description is out of the scope of this paper.

3 FEM MODELS AND NUMERICAL RESULTS

3.1 Tubular spar

First, we present, in Fig.1, a tubular carbon composite spar of an airplane model developed by students of the Federal University of ABC, Santo André, SP, Brazil. Its external diameter is 12 mm and the internal one is 10 mm.

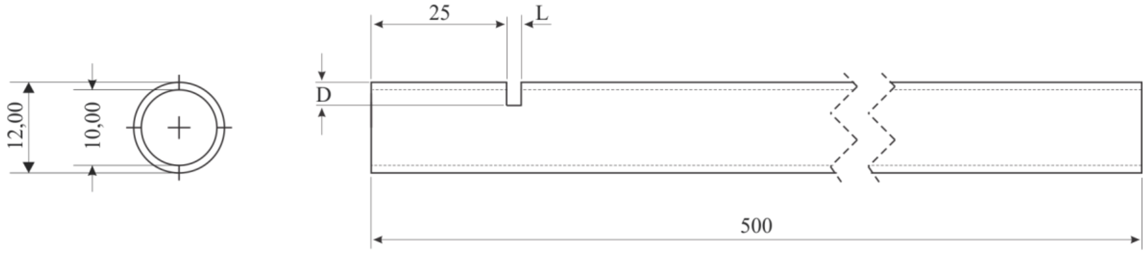

Figure 1: Composite tubular spar

Experimental tests carried out by the students, following D790-02 (Flexural Properties of Composites) Code, obtained Young's module to be 450MPa. Measured density is 2800 kg/m^3.

Figure1 shows a fictitious damage artificially introduced in the spar as a small cut, L =1 mm thick, on the upper surface of the tube. We call D the depth of the cuts and show, in Table 1, the variation of the frequencies of the first 6 modes as a function of this value. Frequencies were determined via ANSYS FEM commercial program and are displayed in Hz.

Table 1

	No damage	D=3mm	D=4.5mm	D=6mm
Mode 1	3,5014	3,0174	2,8471	2,4935
Mode 2	3,5014	3,1786	3,1423	3,0675
Mode 3	21,851	19,068	18,672	18,262
Mode 4	21,852	19,718	19,738	19,727
Mode 5	60,781	53,523	53,254	54,264
Mode 6	60,783	55,061	55,29	55,983

3.2 Wing model

Next, we present a full FEM model of a real experimental airplane 4 meters long wing represented in Fig. 2. The material is 7050 aluminum. The spar is a 120 mm diameter tubular section with 3.18 mm wall thickness.

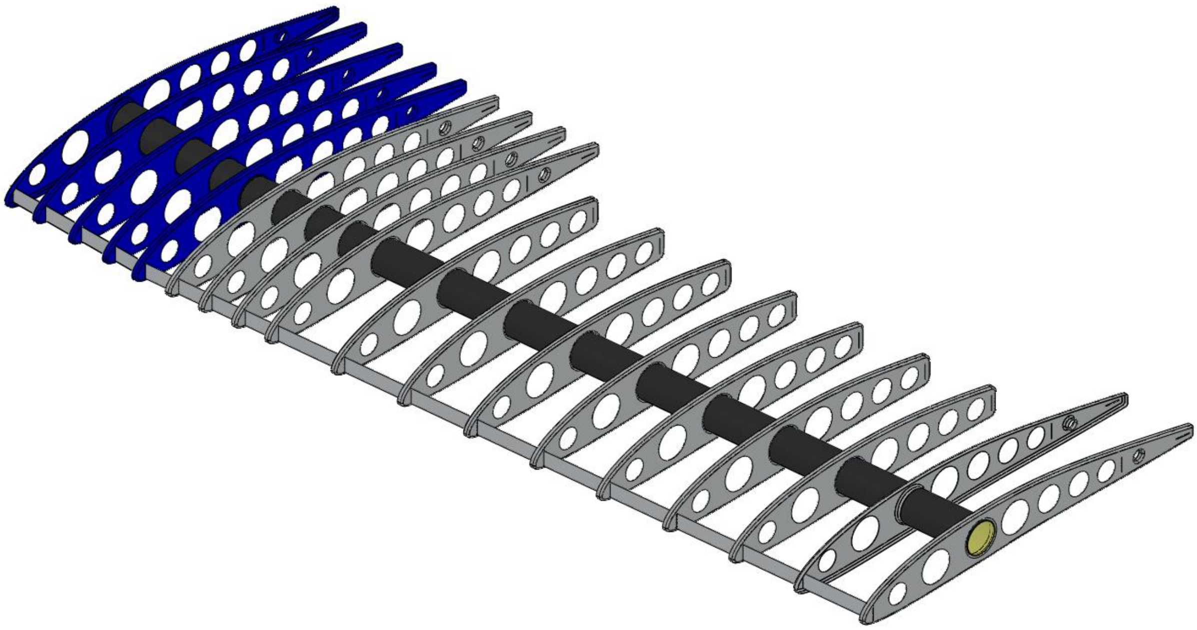

Figure 2: The wing model

The considered fictitious damage is set to be a small variation in the aluminum's Young's module near the root or the spar, the area indicated by dark ribs in Fig.2. Table 2 presents the variation of the first and second natural frequencies with some percentage negative variation of the metal module. Frequencies were determined via ANSYS FEM commercial program.

Table 2 - Frequency per damage percentage (Hz)

Damage	**0%**	**5%**	**10%**	**15%**	**20%**
Mode 1	5,44	5,38	5,31	5,24	5,16
Mode 2	5,64	5,57	5,50	5,42	5,34

Figure 3 shows graphically this variation of the frequencies with some percentage loss of the Young's module of the material.

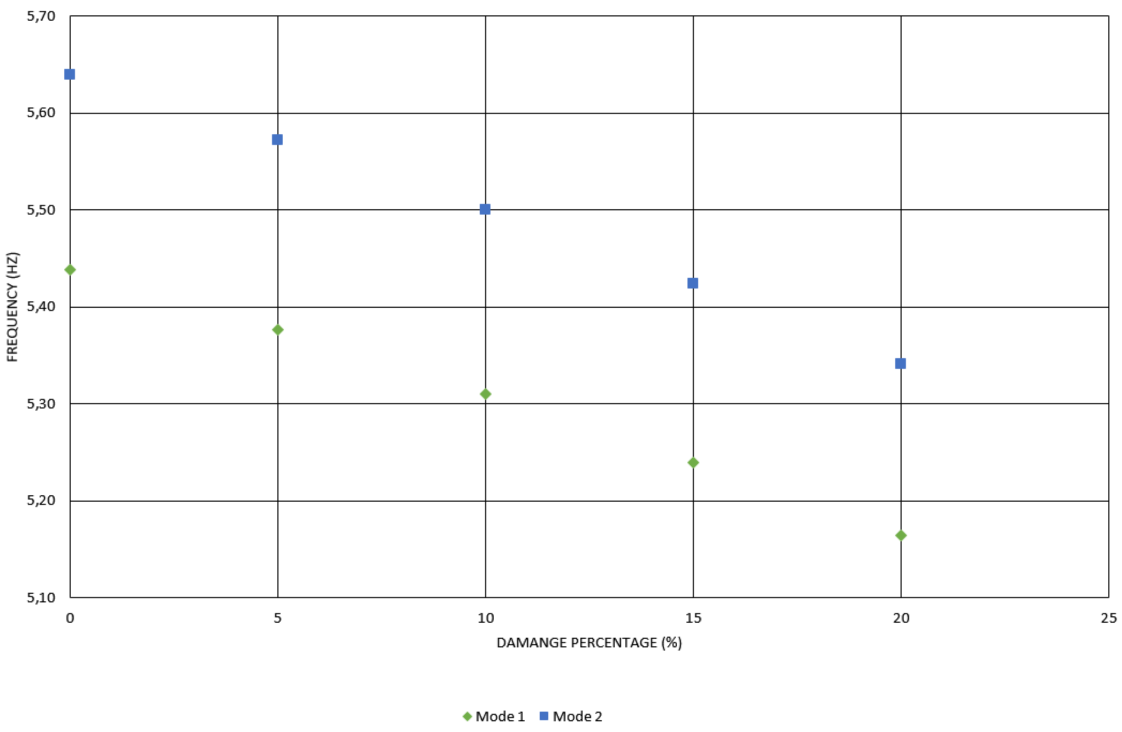

Figure 3: Frequency variation with Youing'1s module percent losses

3.3 Full wing and aircraft models

Finally, we present results of two FEM studies of large airliners. One is a full wing and another full aircraft model.

Figure 4 presents the first mode of free undamped vibrations of a full aluminum wing, in its original non damaged configuration. As we introduce artificial damage in the model, its first 5 frequencies vary, as shown in Table 3. These introduced damages are located in three positions along the wing: near its biding to the fuselage, in the center and at the tip of the wing.

Figure 5 displays the first mode of free undamped vibrations of a full aircraft model, in its original non damaged configuration. As we introduce artificial damage in the model, its 10 first frequencies vary, as shown in Table 3. The first 6 modes are, of course, the rigid body modes, to be neglect in our study. These introduced damages are located in two positions: at the wing, near its biding to the fuselage, and in the aircraft tail.

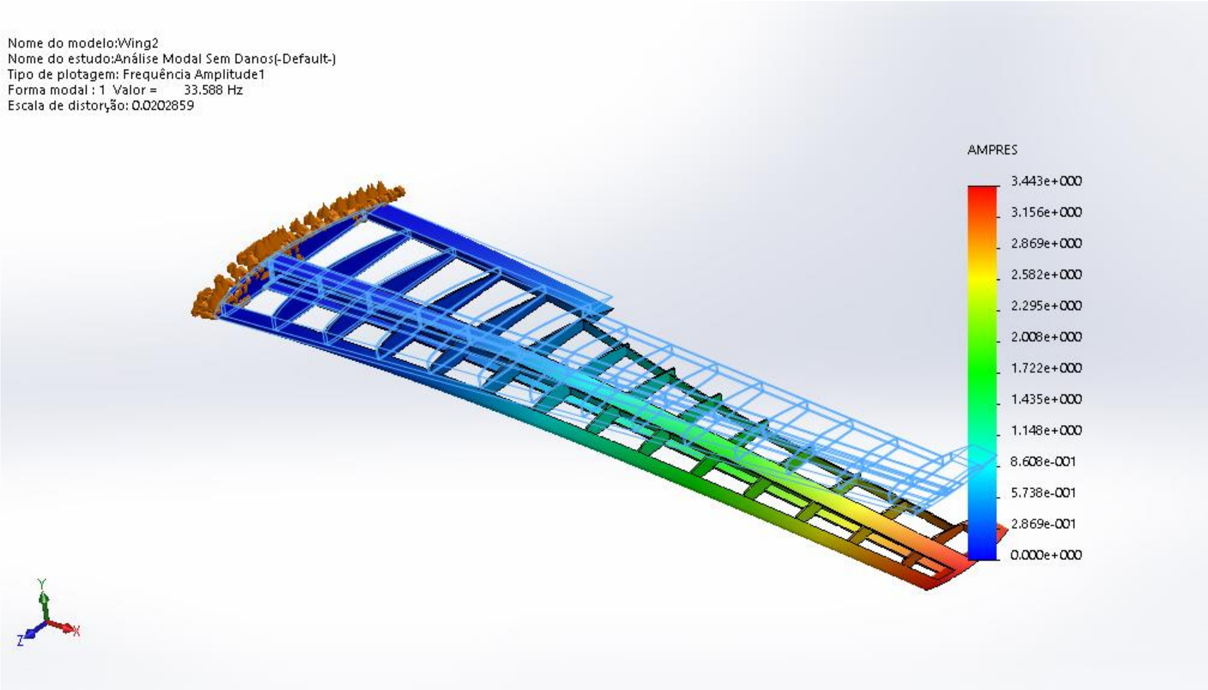

Figure 4: First vibration mode of an undamaged aluminum wing model

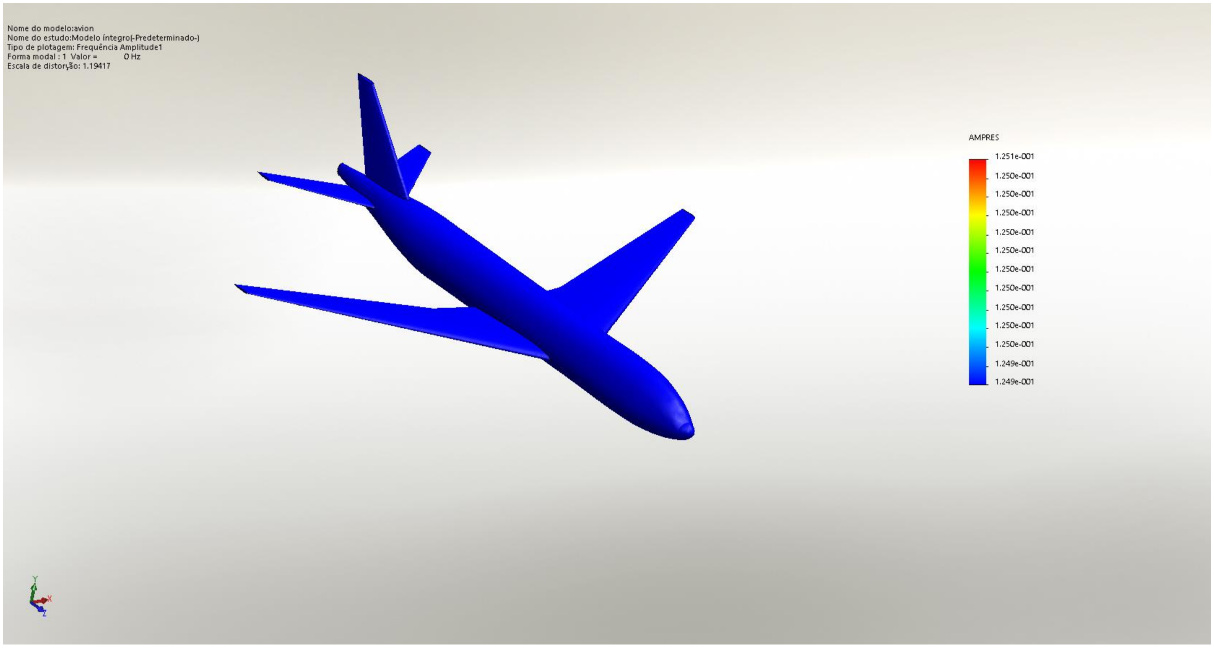

Figure 5: First vibration mode of an undamaged full aircraft model

Table 3

Modes	Natural Frequencies (Hz) for 7050 - t7651			
	Non-Damaged Wing	Damage Near the Binding Spot	Damage on the Center	Damage on the Free Side
1	33.588	33.409	33.528	33.615
2	47.602	47.598	47.595	47.599
3	79.055	79.113	78.474	79.11
4	93.063	92.973	92.77	93.088
5	131.04	131.02	131.04	131.03

Modes	Natural Frequencies (Hz) for 7050 - t7651		
	Non-Damaged Model	Wing damaged Model	Tail Damaged Model
1	0	0	0
2	0	0	0
3	0	0	0
4	0	0	0
5	0	0	0
6	0,00038085	0,001112	0,00078576
7	51,241	51,206	51,251
8	72,805	72,732	72,818
9	144,33	144,52	143,73
10	145,43	145,62	144,79

4 CONCLUSIONS

As expected, the natural frequencies of FEM models of damaged aeronautical structures have smaller values them those of the non-damaged models. As discussed before, this happens because the dynamic properties of the structure are altered when it is damaged, and the stiffness is lowered. For further analysis, we pretend to use some analysis criterions described in the literature to see the accuracy of such information and later develop a numerical analysis tool for damage detection.

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