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Internal Structural Failure Analysis of Anisotropic Composite Materials under External Solicitations Using "Smart" Materials

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Abstract

The internal plane, aircraft or space structures are often composed of element bars, light beams assembled to their nodes by welding, riveting or even by molding using simple or complex geometrical forms. They are often covered by lightweight materials such as aluminum, composite materials, shape memory alloys or adaptable materials. The assembly of all these elements and materials constitute a structure of high rigidity and of a high mechanical resistance quality and a very significant state of art in the field of aviation and aerospace. These structures are subject to considerable external stimuli such as: external forces, shock, vibration, or change of environment that can cause to them crack propagation or even their total destruction. This analysis can be done discreetly (element by element) using the method of bar elements, element beam or plate elements or continuous models representing the discrete structure and the material that covers it. For this, the definitions of the various materials adaptable composite materials and different coating materials, as well as the internal structures used in aircraft will be defined and presented. Their matrices of mass and stiffness will be determined analytically and by assembling arrays for mass and stiffness of the whole system. Examples of computing applications will be considered for different types of materials and internal structures of aircraft wings to static and dynamic.

Keywords: Composite materials, PZT, finite elements, cyclic loading, damage.

1. Introduction

Composite materials are very popular in modern industry. Among other things, the laminated composites with long fibers that play an integral role over. Originally used in secondary parts and little stressed, they now displace metal alloys, even in critical aircraft structures. The latest Boeing, the "Dreamliner", and has mass 50 percent composites, much of which laminates. The A350 Airbus exceeds this proportion. In the latter, the fuselage, part of the wings and rudders are made of composite laminates. Significant gains in terms of weight engendered by replacing metal alloys by composite materials make these devices more sober and therefore both more ecological and economical. The high temperature and constraints due to theft can cause deformation of creep and oxidation of the material. The succession of phases of flight can lead to fatigue in the presence of environmental variation. The development of structures in such conditions the problem of durability and damage tolerance. Composite materials are widely used in aviation. Due to major constraints experienced by the structure of an aircraft in flight, the appearance of small cracks is inevitable. Depending on the situation, these cracks are more or less dangerous. Some cracks do not propagate, others pose a significant risk. Therefore his risk once a crack has been detected, the question is whether it can be dangerous or not. The safety of people is obviously necessary to repair the structure [06] in the first case. But do not repair any cracks, as in the second case where the crack is not dangerous, the repair will be a significant cost to the airline, and will cause delays for passengers. We need to know precisely if the crack is dangerous or not. Apart from extreme cases (crack very small or very large), this diagnosis is not easy to ask, because even a small crack can propagate abruptly. It is clear that the sharpness of this diagnosis is an important issue. Ply delamination of composites is sensitive phenomenon which can occur at any time, but how detected? The search operation on piezoelectric materials into the broader materials "Intelligent" and multifunctional structures and focuses, more specifically, to composite structures consist of beams and plates multilayer whose behavior varied confer their applications in the field of advanced technologies (adaptive structures, forms control and vibration aerospace, medical engineering, etc.)..

2. Modeling of a smart structure

2.1. Laminates comprising a piezoelectric ply.

Bonding a piezoelectric ply above a composite laminate provides a laminate structure having a fold called piezoelectric Fig. 1.

2.2 Laminates having piezoelectric patches.

The laminated piezoelectric chips having folds consists of composite material, while the last fold of the pellets contains square sections glued directly above the latter fig. 2 and 3.

3. Modeling delamination of a composite beam

Delamination of a composite beam is decohesion between two ply, it is the rupture mode o f the interface (01) at the end of the beam Fig. 4, Fig.6 and Fig.7.

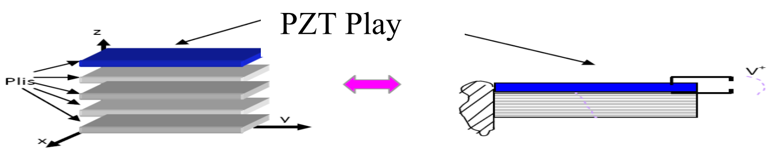

Fig.1. Laminates comprising a piezoelectric ply.

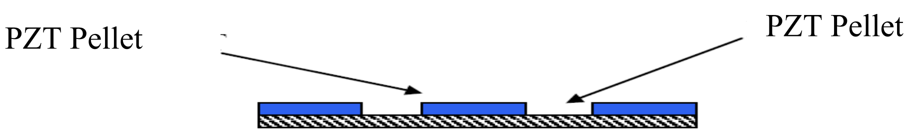

Fig.2. ply comprising a piezoelectric pellet.

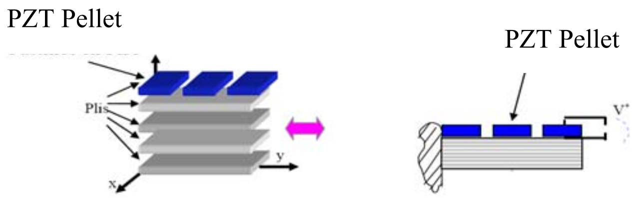

Fig.3. Laminates having piezoelectric patches.

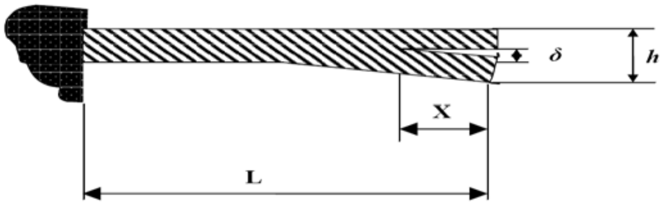

Fig.4. delamination leading.

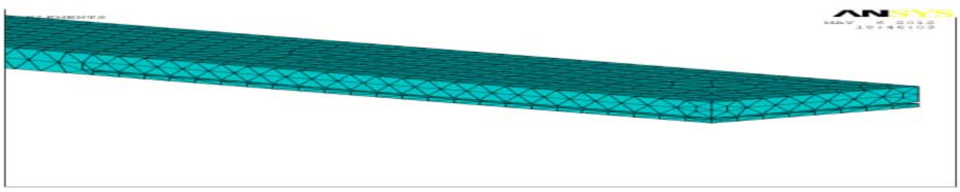

Fig.5. Mesh of a delaminated beam by ANSYS.

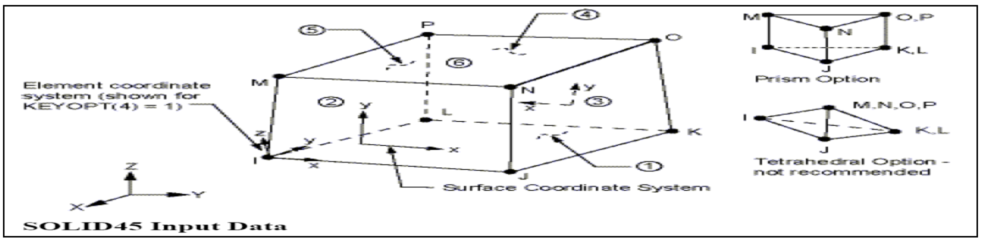

Fig.6. SOLID 45 [Help ANSYS 11]

4. Equations of motion

To determine the dynamic response of a piezoelectric laminate beam, the following equation must be solved:

$$\begin{bmatrix} M & 0 \\ 0 & 0 \end{bmatrix} \begin{Bmatrix} \ddot{u}(t) \\ \ddot{\tilde{V}}(t) \end{Bmatrix} + \begin{bmatrix} K_{uu}^{*} & K_{u\tilde{V}} \\ K_{\tilde{V}u} & K_{\tilde{V}\tilde{V}} \end{bmatrix} \begin{Bmatrix} u^{*}(t) \\ \tilde{V}(t) \end{Bmatrix} = \begin{Bmatrix} F^{*}(t) \\ Q(t) \end{Bmatrix}$$

For modal analysis the stiffness matrices [K] and mass [M] and the force vectors {F (t)} and {Q (t)} were assembled without considering the boundary conditions. The piezoelectric elements can be used either as a sensor via the direct effect of piezoelectricity, or as an actuator or via the reverse effect of piezoelectricity. Depending on their use, the electrical boundary conditions imposed vary from piezoelectric component to another. The equation of motion of the finite element system before the imposition of essential boundary conditions is:

$$\left(\begin{bmatrix} M^{ff} & M^{fl} & 0 & 0 \\ M^{lf} & M^{ll} & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \end{bmatrix}\begin{Bmatrix} \{\ddot{\Phi}^{f*(\Gamma)}\} \\ \{\ddot{\Phi}^{l*(\Gamma)}\} \\ \{\ddot{\tilde{V}}^{f*(\Gamma)}\} \\ \{\ddot{\tilde{V}}^{l*(\Gamma)}\} \end{Bmatrix} + \begin{bmatrix} K_{uu}^{ff*} & K_{uu}^{fl*} & K_{u\tilde{V}}^{fl} & K_{u\tilde{V}}^{ff} \\ K_{uu}^{lf*} & K_{uu}^{ll*} & K_{\tilde{V}u}^{ll} & K_{\tilde{V}u}^{lf} \\ K_{\tilde{V}u}^{lf} & K_{\tilde{V}u}^{ll} & K_{\tilde{V}\tilde{V}}^{ll} & K_{\tilde{V}\tilde{V}}^{lf} \\ K_{\tilde{V}u}^{ff} & K_{\tilde{V}u}^{lf} & K_{\tilde{V}\tilde{V}}^{lf} & K_{\tilde{V}\tilde{V}}^{ff} \end{bmatrix}\begin{Bmatrix} \{\Phi^{f*(\Gamma)}\} \\ \{\Phi^{l*(\Gamma)}\} \\ \{\hat{\tilde{V}}^{f*(\Gamma)}\} \\ \{\hat{\tilde{V}}^{l*(\Gamma)}\} \end{Bmatrix}\right) = \begin{Bmatrix} F^{f*}(t) \\ F^{l*}(t) \\ Q^{f*}(t) \\ Q^{l*}(t) \end{Bmatrix}$$

Solving for the free DDLs $\o^{l}$ and $\tilde{V}$ we obtain:

$$\left(\begin{bmatrix} M^{lf} & M^{ll} & 0 & 0 \\ 0 & 0 & 0 & 0 \end{bmatrix}\begin{Bmatrix} \{\ddot{\Phi}^{f*(\Gamma)}\} \\ \{\ddot{\Phi}^{l*(\Gamma)}\} \\ \{\ddot{\tilde{V}}^{f*(\Gamma)}\} \\ \{\ddot{\tilde{V}}^{l*(\Gamma)}\} \end{Bmatrix} + \begin{bmatrix} K_{uu}^{fl*} & K_{uu}^{ll*} & K_{\tilde{V}u}^{ll} & K_{\tilde{V}u}^{lf} \\ K_{\tilde{V}u}^{lf} & K_{\tilde{V}u}^{ll} & K_{\tilde{V}\tilde{V}}^{ll} & K_{\tilde{V}\tilde{V}}^{lf} \end{bmatrix}\begin{Bmatrix} \{\Phi^{f*(\Gamma)}\} \\ \{\Phi^{l*(\Gamma)}\} \\ \{\hat{\tilde{V}}^{f*(\Gamma)}\} \\ \{\hat{\tilde{V}}^{l*(\Gamma)}\} \end{Bmatrix}\right) = \begin{Bmatrix} F^{l*}(t) \\ Q^{l*}(t) \end{Bmatrix}$$

Note that the imposition of the value of certain DDLs amounts to changing the vector of external forces. In fact, the new force vector is obtained which can distort the structure in order to obtain the values of the DDLs imposed. Note also that the vector of electrical potentials imposed V^{f} represents the voltages applied to the terminals of the piezoelectric actuators. As for piezoelectric sensors, their voltages are obtained by solving the vector of voltages Free V^{l}.

5. Validation of the finite element

From these notions presented a Fortran program (DELAMFORT) has been developed to validate the accuracy of the formulation. This section discusses the validation of the finite element laminated. The purpose of this section is to demonstrate that the developed finite element behaves correctly. The validation procedure is divided into two steps: validation of the elastic component and the piezoelectric component. For each step, a comparison will be made between the results obtained by ANSYS and our finite element model (DELAMFORT). This will be achieved as a result of modal analysis for the different cases of delamination.

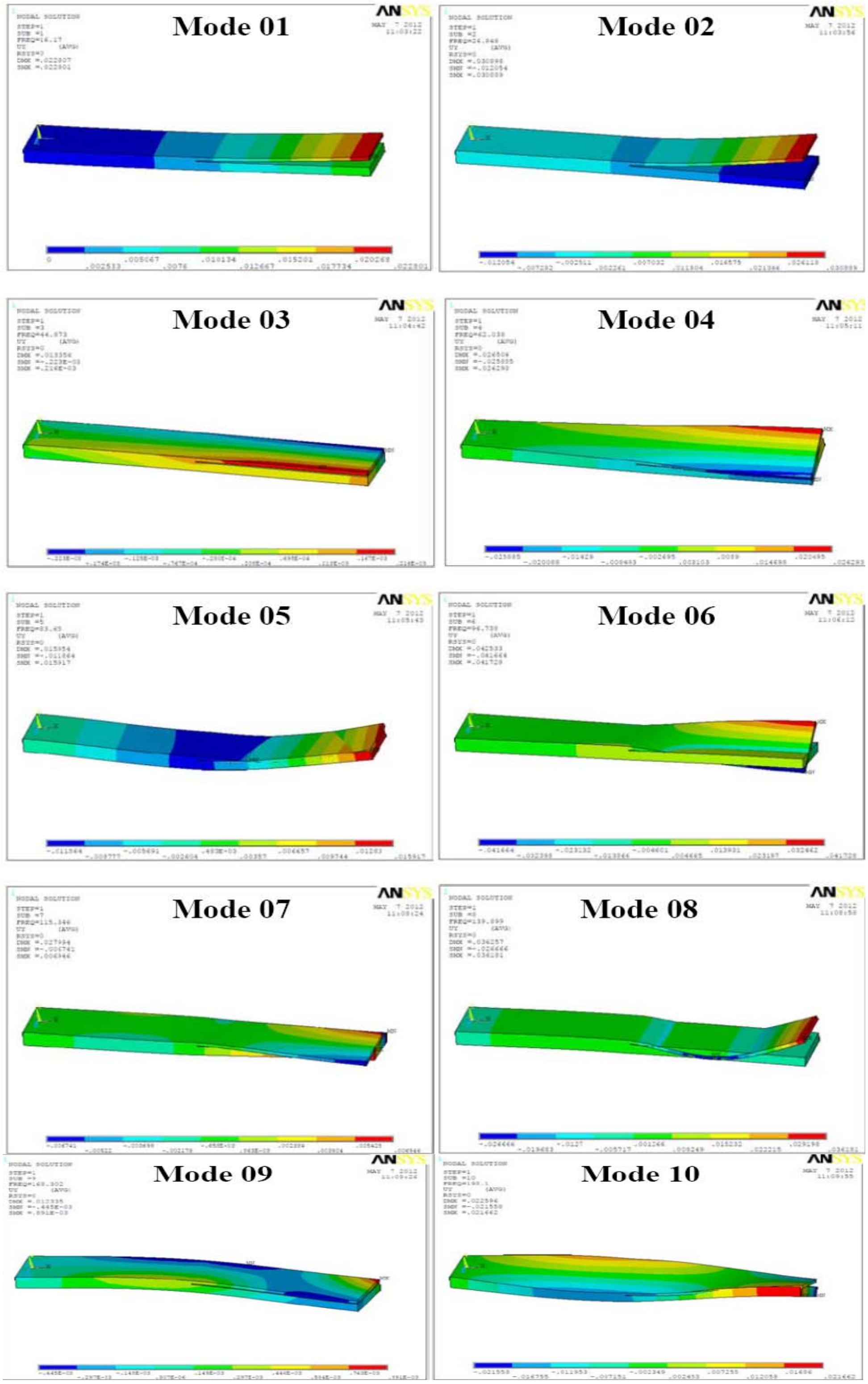

Fig.7. Vibration modes by ANSYS

Mode	ANSYS	Simulation	Error
1	14.507	14.6	-0.093
2	20.624	20.15	0.474
3	44.884	42.884	2
4	59.687	60.62	-0.933
5	79.906	79.9	0.006
6	81.963	84.1024	-2.1394
7	104.787	104.5	0.287
8	116.652	118.67	-2.018
9	168.391	168.63	-0.239
10	190.322	190	0.322

Table1. Frequencies depending on the modes of vibration ANSYS / Simulation delamination between 6em and 7em fold.

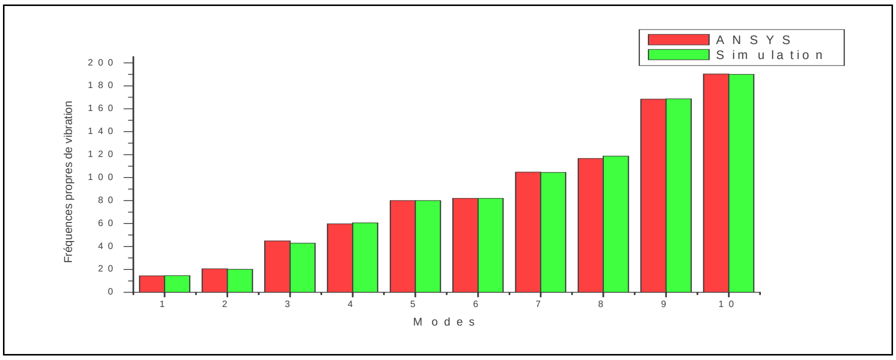

Fig.8. Frequencies depending on the modes of vibration ANSYS / Simulation delamination between 6em and 7em fold.

6. The static responses of piezoelectric transducer

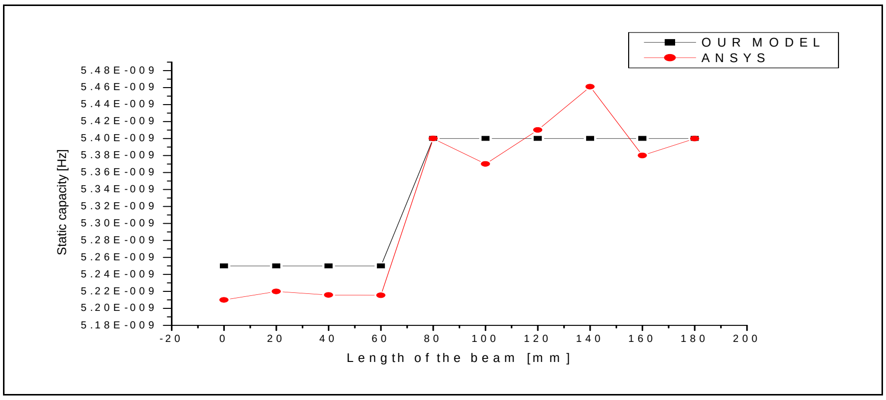

Fig.9. Static capacity transducers for delamination of 40 mm.

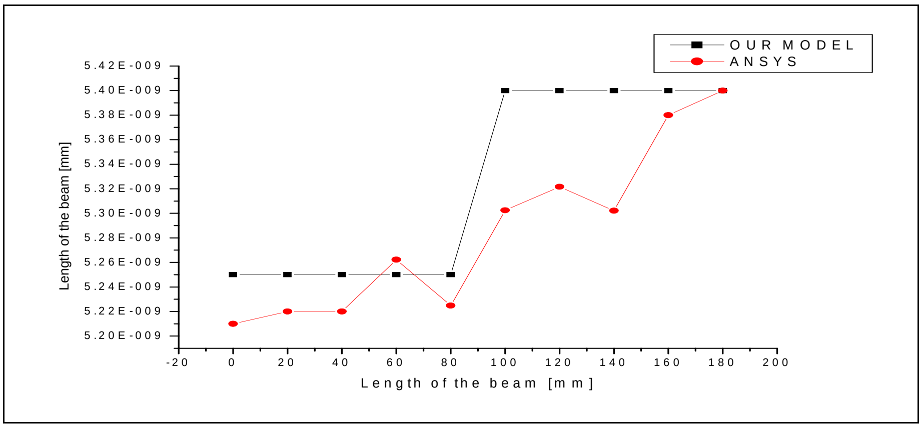

Fig.10. Static capacity transducers for delamination of 60 mm.

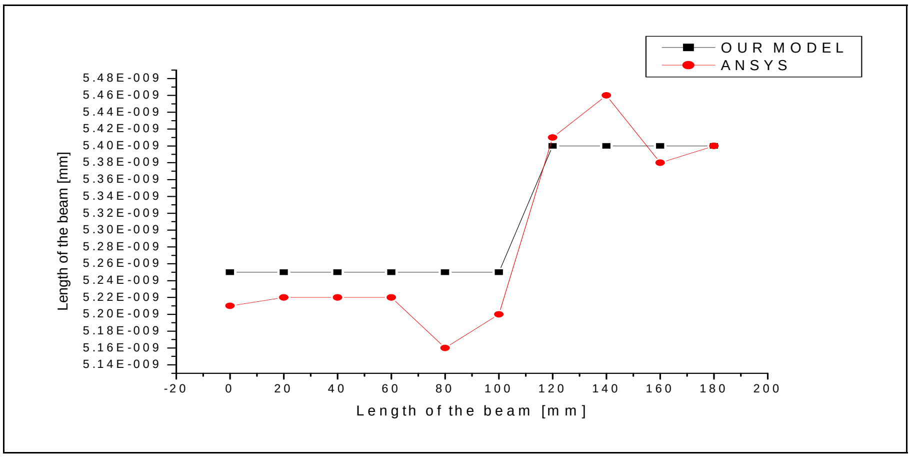

Fig.11. Static capacity transducers for delamination of 80 mm.

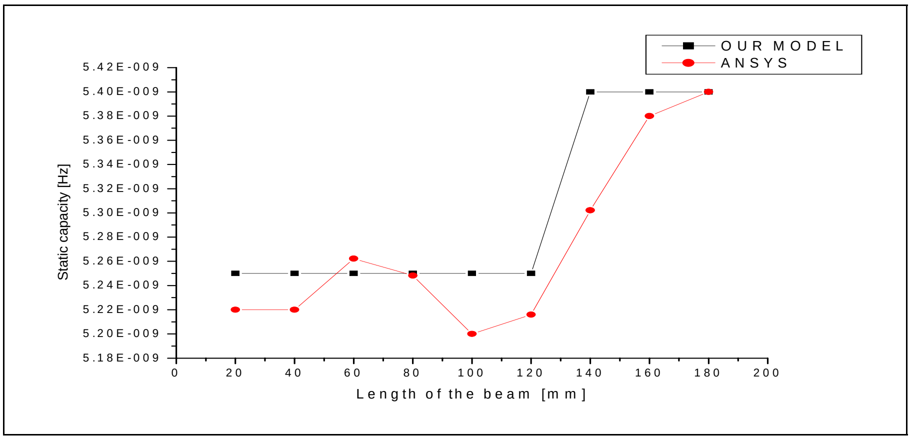

Fig.12. Static capacity transducers for delamination of 100 mm.

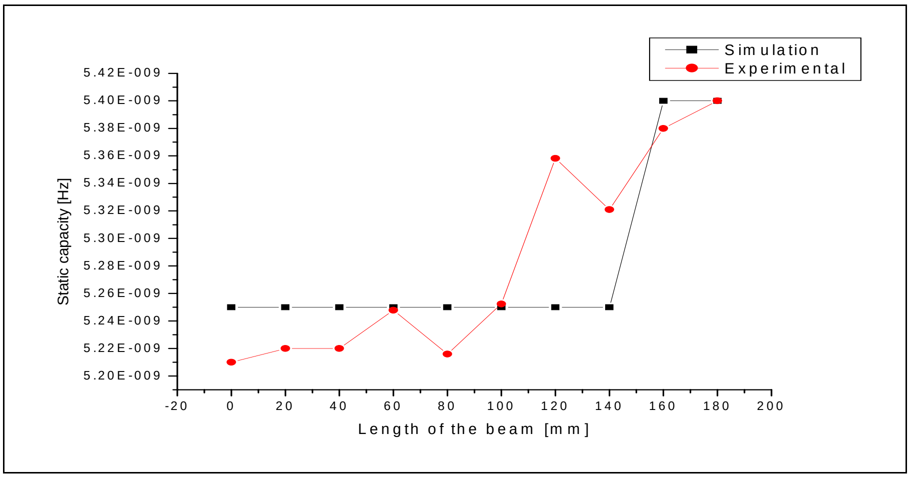

Fig.13. Static capacity transducers for delamination of 120 mm.

7. Conclusions

We have shown in this work the need to involve damage to the control of predictive models of their evolution. In laminated composites, damage mechanisms are complex, particularly for structures made of unidirectional plies. Modeling of crack propagation and delamination still requires significant development, especially under fatigue loading. In the case of laminates made of woven ply, modeling damage is simpler, at least for static loads. The skin of a helicopter blade, which is made of woven ply at 45 °, is a possible candidate for the installation of a monitoring system such as we have defined it. In this work, we have shown that the piezoelectric transducers allowed exploring the fissures monitoring critical areas. In addition, due to their thinness, they can be inserted between two layers of a composite laminate. An important problem remains: these transducers do not support large deformations, or certain measures need a strong coupling and especially do not vary with time, especially when fatigue loads.

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