MECSOL 2019 - Proceedings of the 7th International Symposium on Solid Mechanics,
V. Tita, A. Ferreira, M.L. Ribeiro, A. Vieira and A.T. Beck (Editors), ABCM, Sao Carlos, SP, Brazil, April 15th to 17th, 2019

Nonlinear Finite Element Aeroelastic Analysis of Semi-Infinite Composite Plates Considering Progressive Failure

Douglas Quintanilha Tsunematsu and Maurício Vicente Donadon

Divisão de Engenharia Aeronáutica, Instituto Tecnológico de Aeronáutica, São José dos Campos, SP, Brazil

Abstract: The present work considers a nonlinear finite element model for predicting the aeroelastic behavior of semi-infinite composite plates undergoing intralaminar and translaminar progressive damage in supersonic flow. The semi-infinite plate is modeled using the classical plate theory in conjunction with the von Kármán nonlinear strains, where a plane strain state is assumed. The linear piston theory is used to model the aerodynamic loads. Progressive failure is modeled by a smeared cracking formulation in which stress-based, continuum damage mechanics and fracture mechanics approaches are combined. The nonlinear equations are solved by employing the Newmark method without performing a modal reduction procedure. A simulation for an angle-ply laminate is presented, and the results show that damage considerably changes the aeroelastic behavior of the structure.

Keywords: Aeroelasticity, Panel Flutter, Composites, Progressive Failure, Finite Element Method

INTRODUCTION

Panels that constitute the external structure of aerospace vehicles which operate in supersonic and hypersonic speeds are prone to present an aeroelastic instability called panel flutter. This instability is characterized by a self-excited oscillation of the panel after the flow velocity has reached a critical level.

It is widely known that the occurrence of flutter may damage the panel, leading to catastrophic or fatigue failure. However, this matter has been scarcely investigated until now. Based on nonlinear analyses, Dowell (1970) and Xue and Mei (1993) provided some examples of fatigue life estimation and discussed their implications for the design of isotropic panels. Strganac and Kim (1996) studied the aeroelastic behavior of composite plates considering the natural progression of microcracks in the composite structure. The growth of the microcrack density was modeled using an internal state variable which is a function of the number of cycles. Kim et al. (1998) presented a study of bimodular flutter oscillations of microcrack damaged composite plates, and their results showed that the presence of microcracks reduces the aeroelastic stability of the panel. AsadiGorgi et al. (2015) investigated the effects of all-over part-through cracks on the nonlinear aeroelastic behavior of isotropic panels. The results showed that the amplitudes of the flutter motion can increase or decrease depending on the location of the cracks.

In order to give new insights concerning the flutter-induced damage in composite panels, this work presents a nonlinear finite element model for predicting the aeroelastic behavior of semi-infinite composite plates undergoing intralaminar and translaminar progressive damage in supersonic flow.

FINITE ELEMENT FORMULATION

Consider a semi-infinite laminated plate of length a and thickness h, subjected to a supersonic flow, and located in a right-handed Cartesian coordinate system defined by the axes x, y and z. The length a is aligned with the x direction and the airflow above the panel is in the positive x direction.

The von Kármán nonlinear strain-displacement relation expressed in terms of the classical plate theory displacement field (Reddy, 2004) for a plane strain state is

$$\varepsilon_x = \frac{\partial u}{\partial x} + \left(\frac{1}{2}\frac{\partial w}{\partial x}\right)^2 - z\frac{\partial^2 w}{\partial x^2} = \varepsilon_{x,L}^m + \varepsilon_{x,NL}^m - z\kappa_x \tag{1}$$

where u and w are the midplane displacements in the x and z directions, respectively.

The laminate constitutive relations (Reddy, 2004) are given by

$$\begin{Bmatrix} N_x \\ M_x \end{Bmatrix} = \begin{bmatrix} A_{11} & B_{11} \\ B_{11} & D_{11} \end{bmatrix} \left(\begin{Bmatrix} \varepsilon^m_{x,L} \\ \kappa_x \end{Bmatrix} + \begin{Bmatrix} \varepsilon^m_{x,NL} \\ 0 \end{Bmatrix} \right) = \mathbf{C}\boldsymbol{\varepsilon} \tag{2}$$

where N_x and M_x are the in-plane force and moment resultants, and A_{11}, B_{11} and D_{11}, that form the laminate constitutive matrix $\mathbf{C}$, represent the extensional, bending-extensional coupling and bending stiffnesses, respectively. These quantities are expressed in terms of the lamina constitutive matrix in the principal material coordinate system $\mathbf{Q}$ and the fiber orientation angle θ. The terms of the matrix $\mathbf{Q}$ are calculated as a function of E_1, E_2, G_{12}, ν_{12} and ν_{21}, which are the lamina elastic moduli, lamina shear modulus and Poisson's ratios, respectively.

The aerodynamic load is described by the quasi-steady first order piston theory (Guo and Mei, 2003), and is expressed as

$$p_a = -\frac{2q}{\beta}\left(\frac{\partial w}{\partial x} + \frac{M^2-2}{M^2-1}\frac{1}{V}\frac{\partial w}{\partial t}\right) = -\left(\lambda \frac{D^0_{11}}{a^3}\frac{\partial w}{\partial x} + g_a \frac{D^0_{11}}{\omega_0 a^4}\frac{\partial w}{\partial t}\right) \tag{3}$$

with

$$\lambda = \frac{2qa^3}{\beta D^0_{11}}, \quad g_a = \frac{\rho_a V(M^2-2)}{\rho_m h \omega_0 \beta^3} = \sqrt{\lambda c_a}, \quad c_a = \left(\frac{M^2-2}{M^2-1}\right)\frac{\mu}{\beta}, \quad \mu = \frac{\rho_a a}{\rho_m h}$$
$$\omega_0 = \sqrt{\frac{D^0_{11}}{\rho_m h a^4}}, \quad D^0_{11} = \frac{E_1 h^3}{12(1-\nu_{12}\nu_{21})}, \quad q = \frac{\rho_a V^2}{2}, \quad \beta = \sqrt{M^2-1}$$

where λ is the nondimensional aerodynamic pressure, g_a the nondimensional aerodynamic damping, c_a the aerodynamic damping coefficient, μ the mass ratio, ω_0 the reference frequency, D^0_{11} the reference stiffness, q the dynamic pressure, V the airflow speed, M the Mach number, ρ_a the air density and ρ_m the material density.

Applying the Hamilton's principle, we can write

$$\int_{t_1}^{t_2}\int_0^{l_x}\left[\delta u \rho_m h \ddot{u} + \delta w \rho_m h \ddot{w} + \delta w_{,x}\frac{\rho_m h^3}{12}\ddot{w}_{,x} + \delta\boldsymbol{\varepsilon}^T\mathbf{C}\boldsymbol{\varepsilon} - \delta w p_a\right]dxdt = 0 \tag{4}$$

where l_x is the element size, t_1 the initial time and t_2 the final time.

An element based on the Rodrigues' form of Legendre orthogonal polynomials (Bardell, 1989) is employed for the finite element discretization, where fifth and third order shape functions are used for the bending and membrane degrees of freedom, respectively. With this, from Eq. (4) and applying the boundary conditions, the system equations of motion can be assembled, and are given by

$$\mathbf{M}\ddot{\mathbf{w}} + g_a\mathbf{C}_a\dot{\mathbf{w}} + (\mathbf{K}_{NL} + \lambda\mathbf{K}_a)\mathbf{w} = \mathbf{0} \tag{5}$$

where $\mathbf{M}$ is the global stiffness matrix, $\mathbf{C}_a$ the global aerodynamic damping matrix, $\mathbf{K}_{NL}$ the global nonlinear stiffness matrix, $\mathbf{K}_a$ the global aerodynamic stiffness matrix and $\mathbf{w}$ the global nodal displacement vector.

No modal reduction is performed, and applying a small initial perturbation to the panel, the system equations of motion are fully solved in the time domain by employing the Newmark method (Bathe, 2016).

PROGRESSIVE DAMAGE MODEL

The progressive damage model employed in this work is based on the efforts of Yokoyama et al. (2010). The formulation combines stress-based, continuum mechanics and fracture mechanics approaches. Initially, the damage onset is detected by a set of stress based criteria for the different failure modes of a unidirectional ply under plane stress state. After that, irreversible damage variables which are based on a smeared cracking formulation are calculated. These damage variables gradually degrade the stiffness in the lamina level of each element of the structural model. The intralaminar and translaminar fracture toughnesses are considered in the damage variables formulation, enabling the control of the energy dissipation associated with each failure mode.

Damage onset detection

The damage onset is detected by maximum stress criteria, which are given by

$$\frac{\sigma_1}{X_t}=1,\quad \frac{|\sigma_1|}{X_c}=1,\quad \frac{\sigma_2}{Y_t}=1,\quad \frac{|\sigma_2|}{Y_c}=1,\quad \frac{|\tau_{12}|}{S}=1 \tag{6}$$

where σ_1, σ_2 and τ_{12} represent the state of stress in the principal material coordinate system, X_t the tensile strength in the fiber direction, X_c the compressive strength in the fiber direction, Y_t the tensile strength in the matrix direction, Y_c the compressive strength in the matrix direction and S the in-plane shear strength.

Damage evolution laws

When one of the first two failure criteria given in Eq. (6) is met, fiber damage commences and grows according to the following evolution law:

$$d_f = 1-(1-d_f^t)(1-d_f^c) \tag{7}$$

with

$$d_f^t = \frac{\varepsilon_{1,u}^t}{\varepsilon_{1,u}^t-\varepsilon_{1,0}^t}\left(1-\frac{\varepsilon_{1,0}^t}{\varepsilon_1}\right),\quad d_f^c = \frac{\varepsilon_{1,u}^c}{\varepsilon_{1,u}^c-\varepsilon_{1,0}^c}\left(1-\frac{\varepsilon_{1,0}^c}{|\varepsilon_1|}\right)$$

$$\varepsilon_{1,0}^t=\frac{X_t}{E_1},\quad \varepsilon_{1,0}^c=\frac{X_c}{E_1},\quad \varepsilon_{1,u}^t=\frac{2G_{fc}^t}{X_t l^*},\quad \varepsilon_{1,u}^c=\frac{2G_{fc}^c}{X_c l^*},\quad l^* = l_x\sqrt{2}(45°-|\theta|)$$

where ε_1 is the longitudinal strain in the principal material coordinate system and G_{fc}^t and G_{fc}^c are the translaminar fracture toughnesses associated with longitudinal tension and compression, respectively. The quantity l^* represents the characteristic length (Bažant, 1985).

For matrix damage, once any of the failure criteria given by Eqs. (6) are met, damage commences and grows according to the following evolution law:

$$d_m = 1-(1-d_m^t)(1-d_m^c)(1-d_m^s) \tag{8}$$

with

$$d_m^t = \frac{\varepsilon_{2,u}^t}{\varepsilon_{2,u}^t-\varepsilon_{2,0}^t}\left(1-\frac{\varepsilon_{2,0}^t}{\varepsilon_2}\right),\quad d_m^c = \frac{\varepsilon_{2,u}^c}{\varepsilon_{2,u}^c-\varepsilon_{2,0}^c}\left(1-\frac{\varepsilon_{2,0}^c}{|\varepsilon_2|}\right),\quad d_m^s = \frac{\gamma_{12,u}}{\gamma_{12,u}-\gamma_{12,0}}\left(1-\frac{\gamma_{12,0}}{|\gamma_{12}|}\right)$$

$$\varepsilon_{2,0}^t=\frac{Y_t}{E_2},\quad \varepsilon_{2,0}^c=\frac{Y_c}{E_2},\quad \gamma_{12,0}=\frac{S}{G_{12}},\quad \varepsilon_{2,u}^t=\frac{2G_{mc}^t}{Y_t l^*},\quad \varepsilon_{2,u}^c=\frac{2G_{mc}^c}{Y_c l^*},\quad \gamma_{12,u}=\frac{2G_{mc}^s}{S l^*}$$

where ε_2 and γ_{12} are transverse and in-plane shear strains in the principal material coordinate system and G_{mc}^t, G_{mc}^c and G_{mc}^s are the intralaminar fracture toughnesses associated with transverse tension, transverse compression and in-plane shear, respectively.

Constitutive matrix degradation

The lamina constitutive matrix in the principal material coordinate system is degraded in each layer of each element using the degradation approach proposed by Van Paepegem and Degrieck (2003). The damaged lamina constitutive matrix in the principal material coordinate system is expressed as

$$\mathbf{Q}_d = \begin{bmatrix} (1-d_f)Q_{11} & \sqrt{(1-d_f)}\sqrt{(1-d_f)(1-d_m)}Q_{12} & 0 \\ \sqrt{(1-d_f)}\sqrt{(1-d_f)(1-d_m)}Q_{12} & (1-d_f)(1-d_m)Q_{22} & 0 \\ 0 & 0 & (1-d_f)(1-d_m)Q_{66} \end{bmatrix} \tag{9}$$

RESULTS AND DISCUSSION

In order to illustrate the capabilities of the model, the results of a simulation for a simply supported symmetric angle-ply AS4/PEEK panel of fourteen layers $[+60^o -60^o +60^o -60^o +60^o -60^o +60^o]_S$ for $\lambda = 500$ and $c_a = 0.01$ are presented. The panel dimensions are $a = 200$ mm and $h = 1.78$ mm. The material properties are $E_1 = 127.6$ GPa, $E_2 = 10.3$ GPa, $G_{12} = 6$ GPa, $\nu_{12} = 0.32$, $\rho_m = 1600$ kg/m³, $X_t = 2023$ MPa, $X_c = 1234$ MPa, $Y_t = 80$ MPa, $Y_c = 160$ MPa, $S = 82.6$ MPa, $G_{fc}^t = 128$ kJ/m², $G_{fc}^c = 128$ kJ/m², $G_{mc}^t = 5.6$ kJ/m², $G_{mc}^c = 9.31$ kJ/m² and $G_{mc}^s = 4.93$ kJ/m² (Chen et al., 2014, Carlile et al., 1989, Pinho et al., 2004, Zhang and Hartwig, 1998). For the discretization, 100 elements are used.

It can be seen in Figs. (1) and (2), where $W = w/h$ and $\dot{W} = \dot{w}/h$ at $x/a = 0.75$, that matrix damage leads to a more unstable aeroelastic behavior, with higher amplitudes and velocities. Additionally, new orbits are created in the phase plane plot, giving rise to a nonperiodic oscillation. Figure (3) shows the matrix damage distribution in the top layer ($\xi = x/a$), which is, in conjunction with the bottom layer, the layer where the damage is more critical. For this scenario, the matrix damage distribution for the top and bottom layers is very similar. No fiber damage is detected in this simulation.

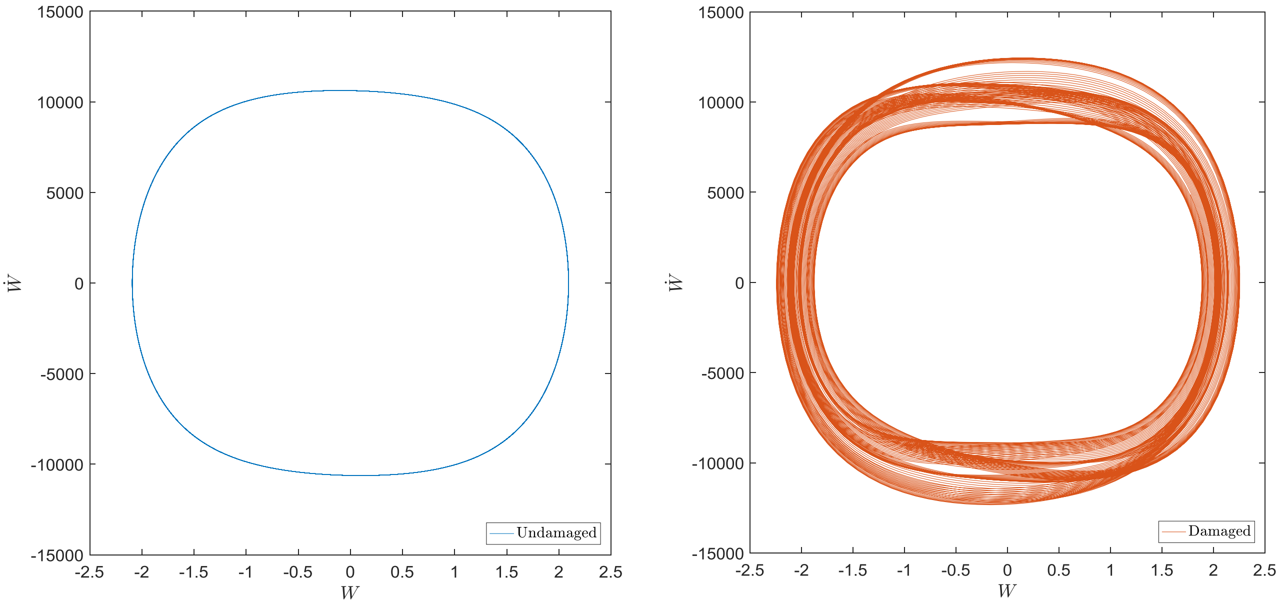

Figure 1 – Phase Plane Plots Comparison

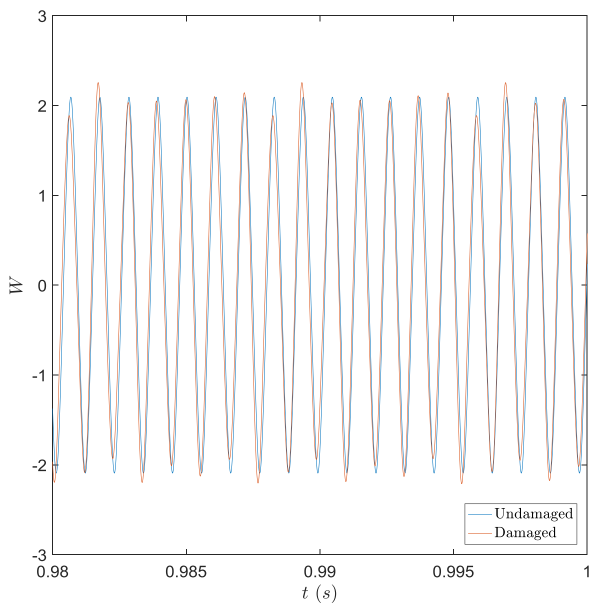

Figure 2 – Time Histories Comparison

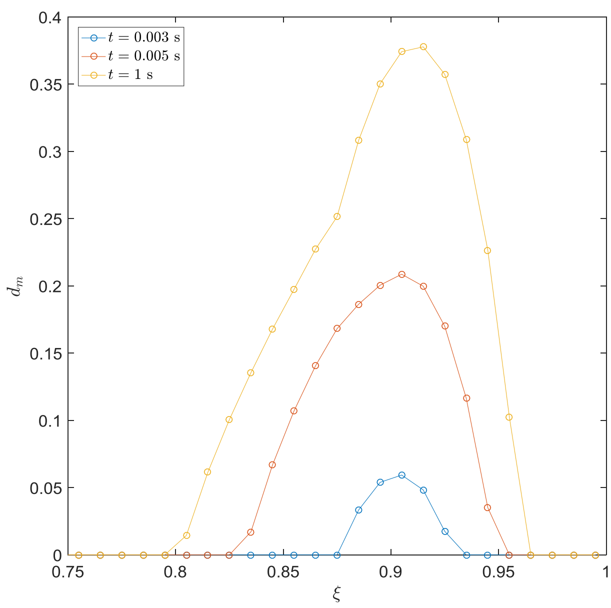

Figure 3 – Top Layer Matrix Damage

CONCLUSIONS

A nonlinear finite element model for predicting the aeroelastic behavior of semi-infinite composite plates undergoing intralaminar and translaminar progressive damage in supersonic flow has been presented. Results show that damage considerably changes the aeroelastic behavior of the structure, where higher amplitudes and velocities are present, and new orbits in the phase plane plot are created.

ACKNOWLEDGMENTS

This work was supported by the Coordination for the Improvement of Higher Education Personnel (CAPES), grant number 88882.180833/2018-01, the Funding Authority for Studies and Projects (FINEP), grant number 0114018300 and the National Council for Scientific and Technological Development (CNPq), grant number 301053/2016-2.

REFERENCES

AsadiGorgi, H., Dardel, M., Pashaei, M.H., 2015, "Effects of All-Over Part-Through Cracks on the Aeroelastic Characteristics of Rectangular Panels", Applied Mathematical Modelling, Vol.39, No. 23, pp. 7513-7536.

Bardell, N.S., 1989, "The Application of Symbolic Computing to the Hierarchical Finite Element Method", International Journal for Numerical Methods in Engineering, Vol.28, pp. 1181-1204.

Bathe, K.J., 2016, "Finite Element Procedures", Klaus-Jürgen Bathe, Watertown, United States of America, 1043 p.

Bažant, Z.P., 1985, "Mechanics of Fracture and Progressive Cracking in Concrete Structures", Fracture Mechanics of Concrete: Structural Application and Numerical Calculation, Springer, Dordrecht, The Netherlands, pp. 1-94.

Carlile, D.R., Leach, D.C., Moore, D.R., Zahlan, N., 1989, "Mechanical Properties of the Carbon Fiber/PEEK Composite APC-2/AS-4 for Structural Applications", Advances in Thermoplastic Matrix Composite Materials, ASTM, Philadelphia, United States of America, pp. 199-212.

Chen, J.F., Morozov, E.V., Shankar, K., 2012, "A Combined Elastoplastic Damage Model for Progressive Failure Analysis of Composite Materials and Structures", Composite Structures, Vol.94, No. 12, pp. 3478-3489.

Dowell, E.H., 1970, "Fatigue Life Estimation of Fluttering Panels", AIAA Journal, Vol.8, No. 10, pp. 1879-1881.

Guo, X. and Mei, C., 2003, "Using Aeroelastic Modes for Nonlinear Panel Flutter at Arbitrary Supersonic Yawed Angle", AIAA Journal, Vol.41, No. 2, pp. 272-279.

Kim, T., Atluri, S.N. and Loewy, R.G., 1998, "Modeling of Microcrack Damaged Composite Plates Undergoing Nonlinear Bimodular Flutter Oscillatons", AIAA Journal, Vol.36, No. 4, pp. 598-606.

Pinho, S.T., Camanho, P.P., de Moura, M.F., 2004, "Numerical Simulations of the Crushing Process of Composite Materials", International Journal of Crashworthiness, Vol.9, No. 3, pp. 263-276.

Reddy, J.N., 2004, "Mechanics of Laminated Plates and Shells: Theory and Analysis", CRC Press, Boca Raton, United States of America, 831 p.

Strganac, T.W. and Kim, Y.I., 1996, "Aeroelastic Behavior of Composite Plates Subject to Damage Growth", Journal of Aircraft, Vol.33, No. 1, pp. 68-73.

Van Paepegem, W. and Degrieck, J., 2003, "Modelling Damage and Permanent Strain in Fibre-Reinforced Composites under In-Plane Fatigue Loading", Composites Science and Technology, Vol.63, No. 5, pp. 677-694.

Xue, D.Y. and Mei, C., 1993, "Finite Element Nonlinear Flutter and Fatigue Life of Two-Dimensional Panels with Temperature Effects", Journal of Aircraft, Vol.30, No. 6, pp. 993-1000.

Yokoyama, N.O., Donadon, M.V. and de Almeida, S.F.M., 2010, "A Numerical Study on the Impact Resistance of Composite Shells Using an Energy Based Failure Model", Composite Structures, Vol.93, No. 1, pp. 142-152.

Zhang, Z., Hartwig, G., 1998, "Low-Temperature Viscoelastic Behavior of Unidirectional Carbon Composites", Cryogenics, Vol.38, No. 4, pp. 401-405.

RESPONSIBILITY NOTICE

The authors are the only responsible for the printed material included in this paper.