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Quantifying matrix cracking in composites by a thermoelastic method

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Abstract

A new experimental and theoretical methodology has been developed to quantify transverse matrix cracking in composite laminates. A model was developed to relate changes in laminate longitudinal stiffness and Poisson's ratio to cyclically-induced temperature changes arising from the thermoelastic effect. Experiments have been performed to measure the above parameters in $(0_3/90_3)_s$ Gl/Ep laminates, and very good correlations were observed between the results and theoretical predictions.

Nomenclature

B	Stefan-Boltzmann constant
C_ε	specific heat
D	damage parameter
e	emissivity
E_{ii}	lamina stiffness in *i* direction
E_{ii}^{o}	original lamina stiffness in *i* direction
E_{xx}	laminate longitudinal stiffness
$\bar{E}_{xx}$	normalized laminate longitudinal stiffness
R	load ratio
t	lamina thickness
S	TSA signal
$\bar{S}$	normalized TSA signal
T_o	average lamina temperature
α_i	lamina CTE in *i* direction
α_i^{o}	original lamina CTE in *i* direction
$\Delta\sigma_i$	lamina stress in *i* direction
$\Delta\sigma_x$	non destructive cyclic stress range
ΔT	thermoelastically-induced temperature change
K	laminate thermal diffusivity
ν_{12}	lamina major Poisson's ratio
ν_{21}	lamina minor Poisson's ratio
ν_{21}^{o}	original lamina minor Poisson's ratio
ν_{xy}	laminate Poisson's ratio
$\bar{\nu}_{xy}$	normalized laminate Poisson's ratio
ρ	density
σ_d	damaging stress

1. Introduction

The initial damage mechanism for composite laminates undergoing tension-tension cyclic loading is transverse intralaminar matrix cracking [1, 2]. These cracks form in the off-axis plies

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and propagate quickly across the width of the laminate. Formation of the cracks does not constitute failure of the specimen, but instead results in a transfer of load to the principal load-carrying plies. It is the failure of these plies which eventually results in the termination of specimen life [3].

It is well-established that longitudinal stiffness decreases with the development of transverse matrix cracks in cross-ply laminates [4]. Researchers have also shown that stiffness changes are directly correlated to a decrease in life for the damaged laminate. Stiffness change has thus proven to be a simple and powerful measure of damage, and most theoretical and experimental work in the area is focused on this. Yet it remains unclear how local stiffness changes are measured in specimens other than simple laboratory laminates. Other damage measures such as changes in Poisson's ratio [5] and coefficients of thermal expansion [6] are also difficult to measure for composite structures.

A new experimental and theoretical methodology has been developed to relate the evolution of matrix cracks in composite laminates to the thermoelastically-induced temperature change in cross-ply laminates using thermoelastic stress analysis (TSA). Changes in the TSA signal were also related to changes in laminate longitudinal stiffness (E_{xx}) and Poisson's ratio (ν_{xy}) in
2 $(0_3/90_3)_s$ Gl/Ep laminates. This work provides an improved theoretical foundation for the behaviour first noted by ZHANG and SANDOR [7].

2. Theoretical developments

The TSA signal S is measured using a sensitive detecting system employing a liquid nitrogen-cooled HgCdTe thermal detector. S corresponds to changes in the infrared emission that result from the thermoelastically-induced temperature change ΔT, and can be expressed as:

$$S = e\,B\,f(T)\,\frac{\Delta T}{T_o} \tag{1}$$

where e is the surface emissivity, B is the Stefan-Boltzmann constant, T_o is the average laminate temperature and $f(T)$ depends upon the thermal detector used in the system. POTTER and GREAVES [8] used the first and second laws of thermodynamics and the Helmholtz free energy to develop the general thermoelastic equation relating the deformation of an anisotropic material to ΔT:

$$\frac{\Delta T}{T_o} = \frac{-1}{\rho C_\varepsilon}\,\{\alpha_1\,\Delta\sigma_1 + \alpha_2\,\Delta\sigma_2\} \tag{2}$$

where ρ is the density, C_ε is the specific heat, and α_i, $\Delta\sigma_i$ are the coefficient of thermal expansion and cyclic stress for the orthotropic ply in the i direction, respectively. Equation (2) assumes that adiabatic conditions are present and that second order effects can be ignored. Equation (1) can now be rewritten as:

$$S = \frac{e\,B\,f(T)}{\rho C_\varepsilon}\,\{\alpha_1\,\Delta\sigma_1 + \alpha_2\,\Delta\sigma_2\} \tag{3}$$

The development of a uniform distribution of transverse intralaminar matrix cracks will result in significant changes in only two of the terms in equation (3). Numerous authors have shown that $\Delta\sigma_i$ change due to the development of damage in composite laminates, and others have shown that the laminate coefficients of thermal expansion also change. Damage-induced changes in S can now be expressed as:

$$\bar{S} = \frac{S}{S^o} = \frac{\{\alpha_1\,\Delta\sigma_1 + \alpha_2\,\Delta\sigma_2\}}{\{\alpha_1^o\,\Delta\sigma_1^o + \alpha_2^o\,\Delta\sigma_2^o\}} \tag{4}$$

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where $\bar{S}$ is the normalized TSA signal, S and S^o are the current and original average TSA signal for the laminate, and α_i, $\Delta\sigma_i$, α_i^o *and* $\Delta\sigma_i^o$ are the current and original ply coefficient of thermal expansion and cyclic stress in the i^{th} direction, respectively.

A new theoretical model was developed to relate the damage-induced changes in E_{xx} and ν_{xy} to $\bar{S}$ for laminates undergoing distributed transverse matrix cracking. Damage in cross-ply laminates was assumed to occur in the 90° plies and consist only of a uniform distribution of transverse matrix cracks. The development of transverse matrix cracks in a ply is assumed to reduce the transverse lamina stiffness E_{22} and transverse lamina coefficient of thermal expansion α_2, and to have no effect on E_{11}, ν_{12} and α_1. E_{22} and α_2 can now be expressed as:

$$E_{22} = E_{22}^o (1 - D) \tag{5}$$

$$\alpha_2 = \alpha_2^o (1 - D) \tag{6}$$

where D is a damage parameter that varies between 0 and 1, and E_{22}^o and α_2^o are the undamaged lamina transverse stiffness and coefficient of thermal expansion, respectively. The minor Poisson's ratio ν_{21} of the damaged lamina can be written as:

$$\nu_{21} = \frac{E_{22}}{E_{11}} \nu_{12} = \nu_{21}^o (1 - D) \tag{7}$$

Classical lamination theory (CLT) was now employed to determine the dependence of the ply stresses on the damage parameter D. The thermoelastically-induced temperature change corresponding to a given value of D for each ply of a laminate was found using the ply stresses and the results of equation (6) in equation (2). This approach was used to find ΔT for each ply in the laminate.

Equations (2) to (4) were developed for plies under adiabatic conditions. However, WONG [9] has shown that laminates can exhibit substantial conduction under typical loading frequencies. DUNN [10] has also shown that the thin coating of pure epoxy present on most composite laminates can significantly influence conduction in the laminate and must be considered in TSA of composites. Thus the thermoelastically-induced temperature changes corresponding to a given value of D were used in Dunn's conduction model [10] to obtain the TSA signal on the surface of the laminate. The normalized TSA signal $\bar{S}$ was then based on these results; however, little conduction was predicted to occur for the laminates and loading conditions under consideration.

E_{xx} and ν_{xy} for the laminate were also formulated using CLT, and each was linearly related to the damage parameter D. The above relationships were numerically evaluated for values of D between 0 and 1 to obtain plots of normalized laminate stiffness ($\bar{E}_{xx}$) and normalized Poisson's ratio ($\bar{\nu}_{xy}$) versus $\bar{S}$, where E_{xx} and ν_{xy} are normalized by the original laminate stiffness and Poisson's ratio, respectively.

3. Experimental results

$(0_3/90_3)_s$ S-2 Gl/Ep laminates were used to verify the theoretical developments of this work. Lamina longitudinal and transverse stiffness E_{11} and E_{22}, Poisson's ratio ν_{12} and coefficients of thermal expansion α_1 and α_2 were measured, and the values are given in *table I*. Other relevant material properties were estimated using results for similar materials, and these properties are also given in *table I*.

The general experimental procedure for monitoring E_{xx}, ν_{xy}, and TSA signal S was as follows:

initial values for S, E_{xx} and ν_{xy} were measured;
the specimen was step-loaded under uniaxial tension to induce damage;
after each step-loading values for S, E_{xx} and ν_{xy} were determined;

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the second and third steps were repeated until multiple delaminations formed, at which time the test was concluded.

3.1. TSA signal measurements

The $(0_3/90_3)_s$ Gl/Ep laminates were 250 mm long, 19 mm wide and 1.9 mm thick, and each had adhesively bonded and beveled tabs for uniform load introduction. The front face of each specimen was lightly sanded with a fine grit paper prior to testing to ensure a uniform laminate emissivity. No special coating was applied to enhance emissivity.

The TSA signal was measured using a scanning infrared camera and a servohydraulic closed-loop testing machine. A non destructive cyclic stress range $\Delta\sigma_x = 26$ MPa with $R = 0.1$ and 21 Hz loading frequency was used during scanning. The largest possible region of each specimen was scanned while avoiding the end tabs and the rubber bands used to hold the extensometer. The laminate TSA signal was obtained after each damaging stress σ_d was applied by averaging the TSA signals corresponding to many points on the laminate. *Figure 1* shows $\bar{S}$ versus σ_d for the $(0_3/90_3)_s$ laminates, which exhibit a reasonable amount of scatter. The TSA signal is seen to increase with damaging load levels, as previously predicted.

3.2. Stiffness and Poisson's ratio measurements

The longitudinal stiffness and Poisson's ratio for each laminate were determined after each damaging load was applied. Extensometers were used to record the laminate strains, and a workstation and pen plotter obtained and analyzed the resulting load and strain data. Epoxy tabs were used as a base for the knife edges of the 25.4 mm longitudinal extensometer, and internal springs held the diametral extensometer to the specimen. *Figures 2* and *3* show $\bar{E}_{xx}$ and $\bar{\nu}_{xy}$ versus σ_d for the laminates, and both measures are seen to decrease with increasing levels of damage.

4. Discussion and comparison of theoretical and experimental results

Figures 4 and *5* show the theoretical predictions and experimental results for $\bar{E}_{xx}$ and $\bar{\nu}_{xy}$ versus $\bar{S}$ for the $(0_3/90_3)_s$ laminates. There is very good correlation between the damage model and the experimental results, and the extent of scatter in the data is reasonable. Another useful check of the theoretical approach is seen in *figure 6*, which presents the theoretical predictions and experimental results for $\bar{E}_{xy}$ versus $\bar{\nu}_{xy}$. Again there is a very good correlation between the theoretical predictions and experimental results.

The theoretical developments in this work are directed toward relating TSA signal changes to other measures of damage for laminated composite specimens subjected to transverse matrix cracking. Additional theoretical work will be necessary to model other forms of damage such as delaminations and fiber fractures, and other types of composites such as random-fiber composites. However, the theoretical developments can be used to quantify damage in any layup that undergoes transverse matrix cracking, regardless of the material properties.

Additional work relating $\bar{S}$ to the density of the transverse microcracks has yielded very good results, and other materials and layups have also been modeled successfully. This approach is now being extended to laminates under random loading.

5. Conclusions

It has been shown that normalized TSA signal changes can be used to quantify the level of transverse matrix cracking in composite laminates. This new approach can also be related to more traditional, but less convenient measures of damage such as changes in stiffness and Poisson's ratio. Additionally, this approach is valid for both laboratory specimens and full-scale composite structures, thereby providing a new theoretical and experimental methodology for quantifying damage in composites.

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Table I
Material properties for Gl/Ep and epoxy coating

Property		Gl/Ep	Epoxy
E_{11}	(GPa)	40.3	4.8
E_{22}	(GPa)	13.0	4.8
ν_{12}		0.27	0.35
α_1	($\mu\varepsilon^{\circ}\cdot K^{-1}$)	5.7	72.0 [10]
α_2	($\mu\varepsilon^{\circ}\cdot K^{-1}$)	25.5	72.0 [10]
t	(mm)	0.18	0.01 [10]
C_ε	($J\cdot kg^{-1}\cdot K^{-1}$)	879.0 [11]	1 047.0 [10]
ρ	($kg\cdot m^{-3}$)	1 850.0 [11]	1 246.0 [10]
K	($m^2\cdot s^{-1}$)	2.1×10^{-7} [11]	1×10^{-7} [10]

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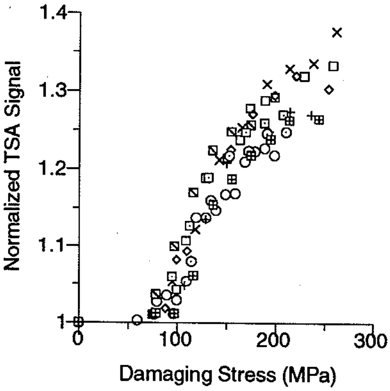

Figure 1. - Normalized TSA signal versus damaging stress, $(0_3/90_3)_s$ Gl/Ep

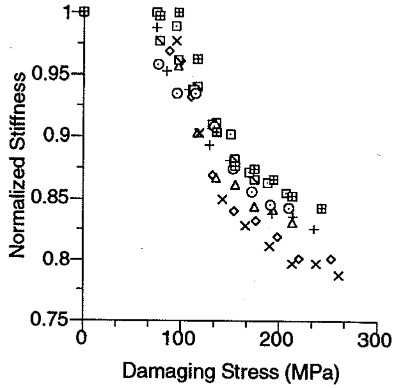

Figure 2. - Normalized stiffness versus damaging stress, $(0_3/90_3)_s$ Gl/Ep

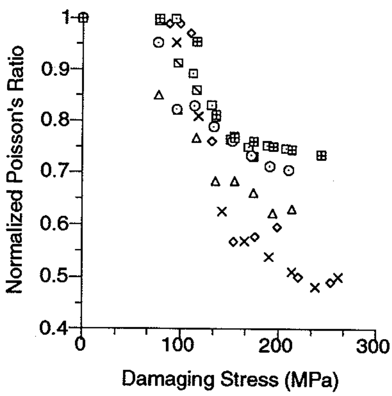

Figure 3. - Normalized Poisson's ratio versus damaging stress, $(0_3/90_3)_s$ Gl/Ep

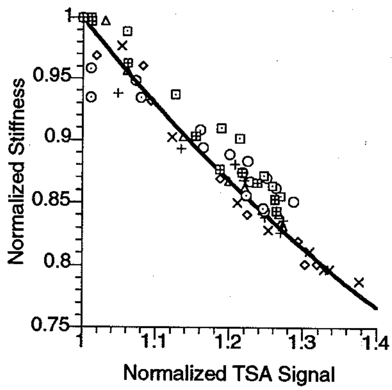

Figure 4. - Normalized stiffness versus normalized TSA Signal, $(0_3/90_3)_s$ Gl/Ep

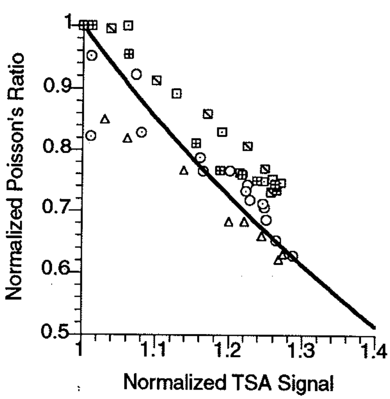

Figure 5. - Normalized Poisson's ratio versus normalized TSA signal, $(0_3/90_3)_s$ Gl/Ep

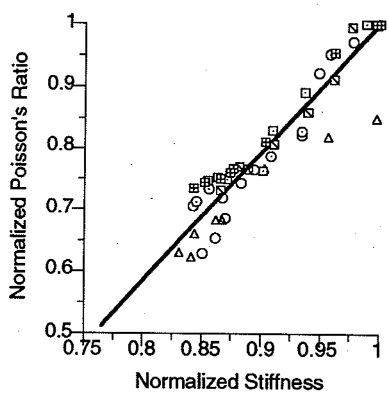

Figure 6. - Normalized Poisson's ratio versus normalized stiffness, $(0_3/90_3)_s$ Gl/Ep