Prediction of fatigue life in composite materials using thermoelastic stress analysis

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Abstract. Thermoelastic Stress Analysis (TSA) is developed to provide a prediction of fatigue life in glass reinforced polymers. A test specimens has been designed to promote cracking and a methodology is defined that allows the measurement of the strain in the damaged region. It is shown that a TSA approach can evaluate fibre breakage, matrix cracking and delamination damage. A strain based metric is established based on calibrated data obtained from the TSA, which can be used to assess the condition of a component throughout its fatigue life.

Introduction

The indiscriminate manner in which damage initiates and propagates in composite materials requires detection and analysis techniques that enable timely intervention and repair to prevent component failure. In metallic structures damage is often localised in the form of cracks, however in composite materials damage accumulates throughout the structure. The mechanisms by which damage initiates and propagates have been well documented and are dependent on the architecture of the laminate. The damage mechanisms are well known and include fibre breakage, matrix cracking, debonding, transverse-ply cracking and delamination. Whilst it is possible to obtain a stress analysis of a virgin laminate, the task of modelling the laminate in the presence of damage is difficult due to the complex manner in which the different damage types interact and propagate simultaneously. Therefore it would be beneficial to devise an experimental approach that provides data that relates to the stress field local to the damage. Although research into the cause and effect of damage has been ongoing, the provision of a quantitative non-contact, full-field technique to rapidly assess the stress redistribution due to damage has not been forthcoming. In this paper Thermoelastic Stress Analysis (TSA) [1] is used to obtain localised strain measurements from the neighbourhood of damage in a component subjected to fatigue loading to provide a means of predicting the remnant life.

TSA is a well established technique for the evaluation of stresses in isotropic engineering components, e.g. [1, 2]. In TSA an infra-red (IR) detector is used to measure the small reversible temperature change associated with the thermoelastic effect from a component subjected to cyclic load. The detector output signal is related to the changes in the sum of the principal stresses on the surface of the material. For orthotropic materials the small temperature change (ΔT) is related to the changes in the stresses in the principal material directions on the surface of the material ($\Delta\sigma_1$, $\Delta\sigma_2$) by the following expression [3]:

$$\Delta T = -\frac{T}{\rho C_p}(\alpha_1 \Delta\sigma_1 + \alpha_2 \Delta\sigma_2) \qquad (1)$$

where α_1 and α_2 are the coefficients of linear thermal expansion relative to the principal fibre axes, ρ is the density, C_p is the specific heat at constant pressure and T is the absolute temperature of the surface.

ΔT is obtained using a DeltaTherm 1400 system to record the thermoelastic data; the IR detector incorporated in the system is a 256 x 256 pixel staring array capable of resolving a temperature change of 2 mK. The thermoelastic signal is processed digitally and allows the data to be recorded in a matter of seconds. This rapid processing enables data collection in practically real-time, providing clear benefit in damage propagation studies, as the system allows the component to be inspected at regular intervals through its life, under the actual fatigue load. In [4] such a system has been used to obtain the stress intensity factors from a metallic subject to fatigue loading. In this paper a fatigue crack is grown in an orthotropic composite and the thermoelastic data is used to give a measure of damage severity.

To obtain the stress change from the IR detector output the usual approach is to calibrate in terms of stress. The complexities of measuring or calculating the direct surface stresses from a laminated orthotropic material has meant that a generalised calibration routine for composites has been elusive. Recent work by the authors [3] has developed a means of calibration that allows the evaluation of the strain change in a laminated composite structure. This means that during damage evolution a quantitative measure of the strain distribution can be made. Aside from the calibration, a further complication in the analysis of the thermoelastic data is that as damage evolves heat is generated local to the damage site. Inspection of Eq. (1) shows that changes in the absolute temperature will effect the magnitude of ΔT and in turn the thermoelastic signal; in [5] a means for correcting for an absolute temperature increase has been devised by the authors that decouples the absolute temperature response from the response associated with the stress change.

In the work described in the present paper damage propagation in a glass reinforced epoxy test specimen is investigated. The specimen is subject to a fatigue loading to generate damage. It is intended that the TSA will provide data that allows a full-field visualisation of the stress redistribution in the specimen. The test programme covers a number of damage initiation mechanisms and will demonstrate that the full-field nature of TSA can be exploited to monitor distributed damage as it propagates throughout a structure. In the work in this paper, holes are introduced in test specimens as stress raisers. Although the stress concentration at cut-outs (such as holes) are known to generate a detrimental stress state and initiate damage, the complexity of this three dimensional problem has limited analytical treatment. Thereby the work in this paper will also demonstrate the usefulness of TSA as an experimental tool in damage and stress analysis of complex composite components.

Damage assessment methodology

To calibrate the thermoelastic data obtained from a laminated composite structure it is necessary to derive the stress change in the surface ply by obtaining a host of mechanical properties and applying classical laminate theory [3]. A generalised form of the thermoelastic equations can be presented in terms of the change in laminate strains [3], $\Delta\varepsilon_L$, $\Delta\varepsilon_T$ and $\Delta\gamma_{LT}$ which appears to be far more complicated. However, the laminate strains can be measured in a simple tensile test and related to the thermoelastic signal, so that it is possible to group the material constants into a 'calibration constant' [3]. It has been shown [3] that the thermoelastic response does not emanate from the orthotropic surface ply but from the isotropic resin-rich surface layer. The existence of the resin-rich layer considerably simplifies the analysis. The resin-rich layer renders the laminates 'mechanically orthotropic', but 'thermoelastically isotropic'. This enabled a new calibration constant to be devised, B^*, as follows:

$$\frac{\Delta\varepsilon_L(1-\nu_{LT})}{S} = \frac{A^*(1-\nu_R)}{\alpha_R E_R} = B^* \tag{2}$$

where ν_R is the major Poisson's ratio of the resin-rich layer, α_R is the coefficient of thermal expansion of the resin-rich layer, E_R is the Young's modulus of the resin-rich layer, ν_{LT} is the major Poisson's ratio of the laminate, A^* is a calibration constant [1] and S is the thermoelastic signal, i.e. the digitized output from the IR detector.

Fig. 1 shows the proposed methodology for analysing damage using TSA. The procedure is implemented using a MATLAB program that is applied to the data array obtained from the Deltatherm software and presents the output in a full-field manner. It incorporates both the temperature correction procedure [5] and the strain calibration procedure [3]. Firstly thermoelastic data, S_0, and absolute temperature, T_0, are obtained from the undamaged specimens. The S_0 data is used to obtain the calibration constant B^* and the T_0 data is used as the baseline for the temperature correction. A damaging load is applied to the specimen and thermoelastic data, S_m, and the temperature, T_m, are obtained at various stages throughout the component life. T_m is also used in the temperature correction in the form $\left(T/T_m\right)^{9.8}$ [5]. S_m is corrected using this quantity and a data set is obtained that is temperature corrected in a point-by-point fashion. The resulting output from this, S_c, is then calibrated as follows:

$$\Delta(\varepsilon_L + \varepsilon_T) = B^*(1-\nu_{LT})S_c \tag{3}$$

The output of the procedure is a metric that is related to the strain sum change in the damaged component that occurs purely as a result of the stress distribution in the component.

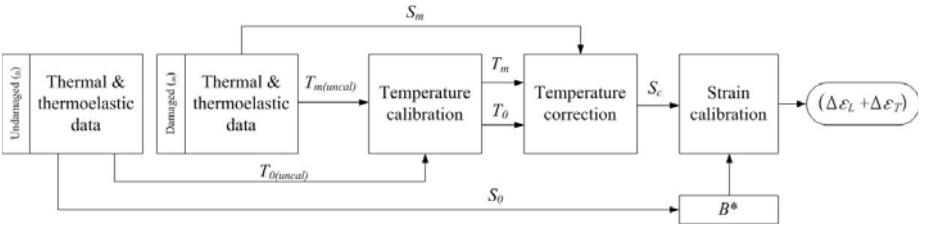

Fig. 1 Damage assessment methodology

Test specimen

A panel was manufactured from 13 layers of a unidirectional (UD) E-glass epoxy (SE84) pre-impregnated (pre-preg) material. A 'cross-ply' laminate was produced with the stacking sequence $[(0/90)_3, 0, (90/0)_3]$. The panel was consolidated under vacuum pressure for one hour and then cured for four hours at a temperature of 80 ºC. After curing, end tab strips were bonded to both sides of the panel using an adhesive film. Specimens were cut from the panel that were 40 mm wide and had an approximate length of 180 mm and thickness of 3.5 mm. An 8 mm hole was introduced in the centre of the specimens using a modified drill bit that minimised tearing of the surface plies. In a cross-ply laminate the dominant damage mechanism is matrix cracking caused by the large mismatch of mechanical properties between the layers [6]; the damage takes the form of small

longitudinal cracks in the transverse ply and splits in the longitudinal ply between the fibre and the matrix. Delamination initiates where these two mechanisms intersect in a laminate stack. In the laminate it is expected that the damage will accumulate and cause stress transfer to the remaining intact plies until a stress state is generated that causes gross failure of the laminate through the failure of the fibres or matrix across the width of the test specimen.

It is intended to load the specimens in uniaxial tension-tension fatigue resulting in traction free edges along all four vertical sides. The stiffness discontinuities between the plies that generate transverse stresses promote a complex stress state at the free-edges. In cross-ply laminates the finite in-plane transverse stress reduces to zero at the free edge causing a finite through-thickness direct stress within the laminate. Also as a consequence of this a finite through-thickness shear stress is introduced in the proximity of the edge which also must reduce to zero at the free edge. This stress system acts within one laminate thickness, t, of the free edge and results in undesirable large stress gradients. The large stress gradient at the edges will tend to *peel* successive plies and result in delamination at the ply interfaces. The ability to withstand this is matrix related and thus damage will be initiated below the expected failure strength of the fibre reinforcement. Therefore edge stresses will initiate localised matrix cracking damage and will tend to propagate into the laminate under the fatigue loading. The magnitudes of the stresses generated at the free edges are a function of the in-plane stresses and therefore will be accentuated in areas of higher in-plane stress. As the specimen contains a hole, it is expected that damage will initiate from the free edge at the hole boundary as a result of the stress concentration, which has been reported to be approximately 5 for a boron reinforced epoxy cross ply laminate [7].

Test procedure

The requirement for cyclically loading specimens during TSA is a major consideration when applying the technique to components subject to damage. As the damage accumulates the elastic properties of the material will change, resulting in changes in the laminate strains for the same given load. This in turn causes a modification to the surface ply stresses and hence a change in the thermoelastic response. To ensure that the readings obtained from TSA are as a result of the stress/strain redistribution due to the damage, a constant *displacement* load of 0.167 mm at a frequency of 10 Hz was applied to the test specimens to maintain a uniform level of longitudinal strain in the laminate. This means that as the material elastic properties deteriorate (particularly the longitudinal modulus) the load applied reduces in proportion to the reduction in the stiffness. The thermal, T, and thermoelastic, S, data were recorded using a DeltaTherm system with a 25 mm lens that meant the detector was positioned at a stand-off distance of 500 mm from the specimen surface to obtain a full-field of view the specimen. The S and T data are the inputs for the thermoelastic procedure illustrated in Fig. 1. The specimen surface was not painted as the epoxy surface provided a sufficiently high and relatively uniform emissivity for the thermoelastic work.

The fatigue damage was introduced independently from the load cycling required for the TSA. *Fatigue steps* that had a constant *load* range of 14 ± 12 kN and frequency of 2 Hz were applied in packets of 3000 cycles. This was because it was not possible to record the TSA data at the amplitude of the fatigue load, as the servo-hydraulic test machine could not achieve the required displacements at the frequency required for TSA. Initially the elastic properties of the undamaged specimen were obtained along with the thermal and thermoelastic data. This was followed by the application of a fatigue load step, after which the cyclic displacement was applied and the TSA data collected. The procedure was repeated 17 times until gross failure had occurred. Glass/epoxy is transparent, so a visual inspection of the specimen provided an indication of the damage in the specimen. Fig. 2 shows an image of the specimen after 17 fatigue steps taken in a macroscope by illuminating from the underside. The fatigue loading has initiated localised damage around the hole. The mismatch in the Poisson's ratio between the 0˚ and 90˚ plies produces an interlaminar shear sufficient to cause cracking of the epoxy matrix. There is matrix cracking in the transverse plies

where the matrix cracks (the short dark horizontal lines in the image) appear to be restricted to the areas of the specimen subject to a tensile strain. Longitudinal splits have occurred in the 0˚ plies, running vertically and parallel with the 0° fibres; these are most severe at the edge of the hole. The dark areas with diffuse edges between the longitudinal splits indicate delaminations. As the concentration of matrix cracking increases to a saturation point, the stress is redistributed at the local level into the unbroken constituents in the 0˚ ply. As there are large stress concentrations at the hole, fibre breakage was initiated at the edge of the hole. Although not visible in the image shown in Fig. 2, this occurred as cracks extending from the hole shown by the dotted lines in Fig. 2.

Thermal data were recorded simultaneously with the thermoelastic data and showed significant variations as the damage was introduced. An increase of 12 K was recorded local to the damage site, with an increase of 5 K away from the damage. From the relationship presented in [5] this would give a 30 % increase in the thermoelastic signal and hence justifies the inclusion of the point-by-point temperature correction procedure in the damage assessment methodology.

At the start of each step the longitudinal and transverse stains were recorded in a quasi-static tension test over a 0 to 5 kN range with a ramp-rate of 1 kN/min. The load applied during the quasi-static test was used to obtain that the global secant Young's modulus, E_L, for the specimens. The purpose of calculating the modulus was to provide a metric with which to compare the TSA data by establishing the residual stiffness of the specimens after N cycles.

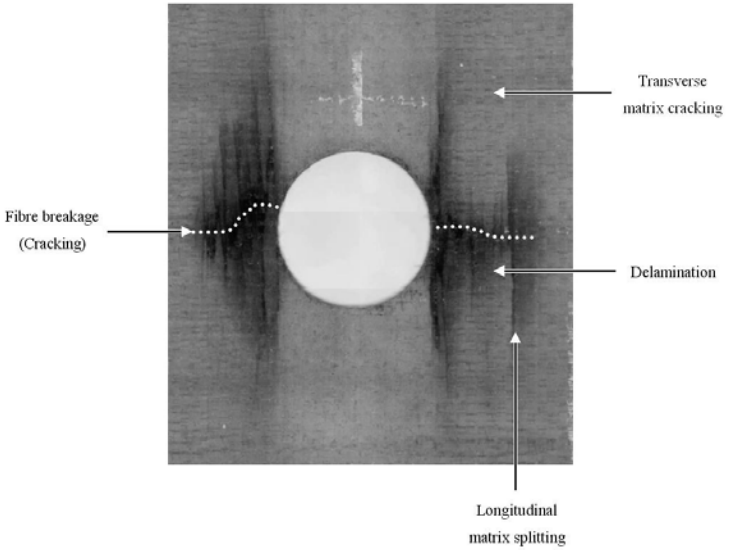

Fig. 2 Macroscope image of damage in crossply

Results and discussion

The strain sum distribution recorded by the TSA at the start of the test is shown in Fig. 3a, at fatigue step 14 in Fig. 3b and from fatigue step 17 in Fig. 3c. The data shown as red around the hole in the image given in Fig 3b and 3c occurs as a consequence of the test specimen motion. This is a well-known phenomenon in TSA and is most pronounced at edges. The effect of motion causes the thermoelastic signal to 'blur'. Observation of the affected area through the fatigue history shows motion has a more severe effect on the data as the stiffness reduces local to the hole. A robust method of compensating thermoelastic data was not available with the DeltaTherm system and therefore in this work it is ignored as the only significant effects were restricted to the vicinity of the hole edge.

A comparison of the line of the cracks in the TSA data with the macroscopic image, shows that the strain has redistributed as a consequence of the cracks and the strain 'concentration' has moved to the tip of the crack. To make a concise global comparison of the all of the collected thermoelastic data an analysis routine was developed so that strain sum was analysed at each stage of the fatigue loading. Three metrics were established: the percentage of the image area that gave a strain sum of greater than 0.001, the percentage of the image area that gave a strain sum of less than 0.001 and the maximum strain sum. The lower image area metric provides an indication of the reduction in strain in certain areas as the load carrying capacity reduces; the upper limit provides an indication of the strain redistribution as a result of the damage. The expectation is that these two metrics will change at the same rate. This data is plotted in Fig. 4 along with the percentage decrease in the measured Young's modulus of the specimen.

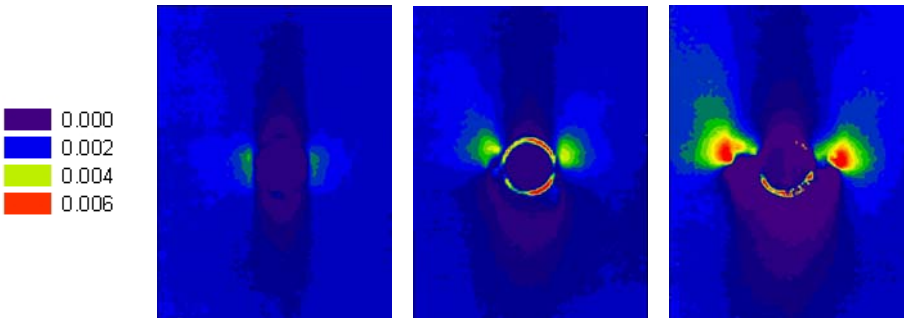

Fig. 3 Strain sum from TSA data

Fig. 4 shows that in the early stages of the fatigue loading (up to step 8) the decrease in Young's modulus is more rapid than the strain redistribution indicated by the TSA data. In fact there is no change in the maximum strain until fatigue step 8. This is because transverse matrix cracking is occurring in the 90˚ plies only during these fatigue steps. As little of the stress is carried by these plies it has a small effect on the global strain and has a less pronounced effect on the strain sum data collected by the TSA. A close examination of the TSA data in the undamaged state and the macroscope image shown in Figure 5 reveals that the transverse matrix cracks are restricted to the areas of tensile strain observed in the undamaged TSA image. Between fatigue steps nine and ten there is a large decrease in Young's modulus of 5 %. Inspection of the specimen revealed the initiation of breakage of the 0˚ fibres at the hole and explains the step change in stiffness at this stage. At fatigue step 9 there is a change in all three TSA strain data sets. At 9 there is an increase in the maximum strain and at 10 there is a decrease. There was a large strain sum concentration at the hole edge at step 9 which was not evident at step 10 clearly indicating that TSA is able to identify the imminent failure seen in the next fatigue step when the crack occurred. At fatigue steps 10 and 11 there is a reduction in the maximum strain followed by an increase at step 12. At step 11 the area metrics also start to increase/decrease more rapidly. At this stage more fibre breakage occurred and the crack in the specimen started to grow progressively. The large changes noted in the TSA data are not present in the modulus data, which simply shows a steady decrease throughout the fatigue life. The TSA data is indicating that significant damage is present at step 9 and at step 12 failure is imminent. The work in this section clearly shows that TSA data can be used as a damage assessment tool. In the future different stacking sequences will be studied that will evaluate the use of the technique on differing damage mechanisms.

Conclusions

It has been shown that TSA can be used in a quantitative manner to obtain the strain distribution in the neighbourhood of damage in laminated glass reinforced fibre composites.

Three types of damage have been studied:

1. Fibre breakage
2. Delamination
3. Matrix cracking

Damage metrics have been developed based on the thermoelastic response throughout the fatigue life. The experimental work described in the paper has shown that these can be used as a damage indicator that is directly related to the level of fatigue damage that the specimen has been exposed to.

The work represents an important initial step in which a methodology for damage assessment has been established. The methodology using TSA accounts for changes in surface temperature due to damage evolution and incorporates a calibration procedure so that the data is presented in terms of strain.

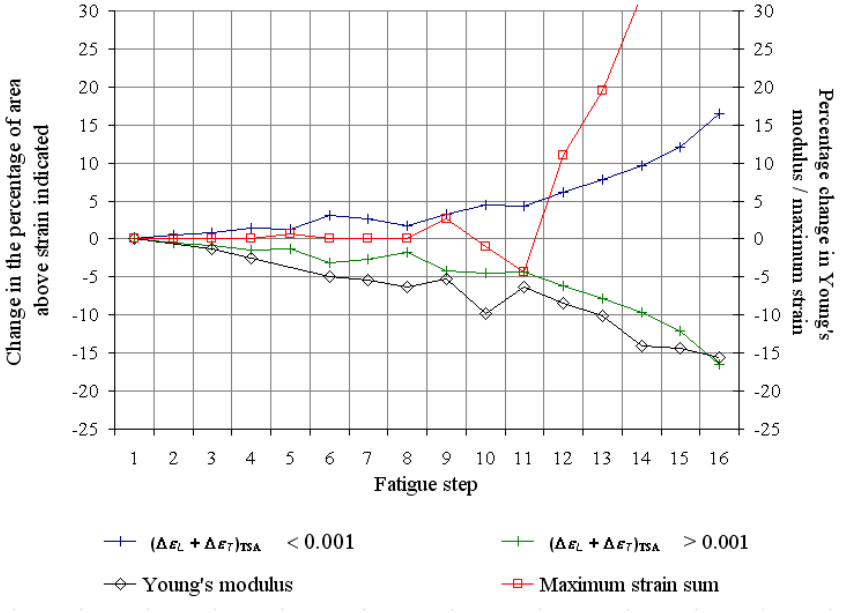

Fig. 4. Strain metrics and Young's modulus decrease

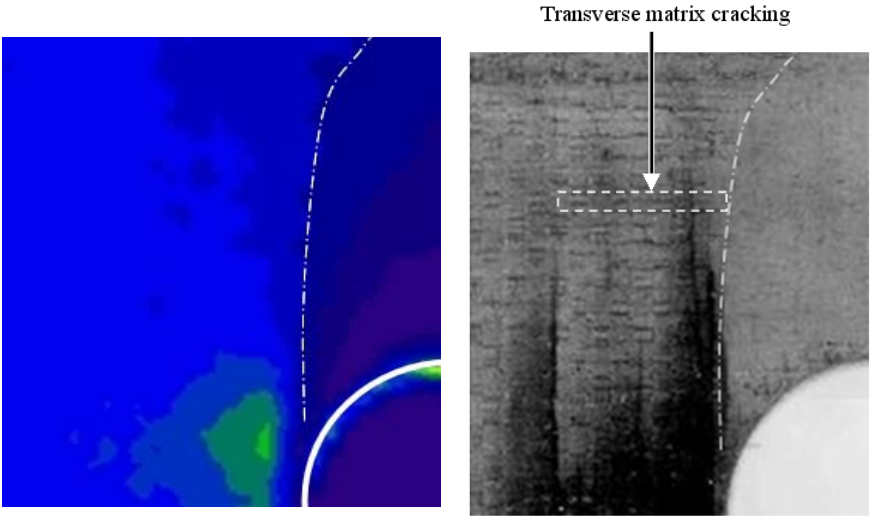

Fig. 5 Comparison of damaged region with initial strain distribution from TSA

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