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Full Field Strain Analysis of Aircraft Sandwich Structures

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Abstract:
A study of the application of two full field optical strain analysis techniques to aircraft standard carbon fibre composites is carried out. The goal of the work is to assess the feasibility of using the techniques on full scale tests of secondary sandwich structure aircraft panels. The work concentrates on two techniques: Digital Image Correlation (DIC) and Thermoelastic Stress Analysis (TSA). The DIC work focuses on the accuracy of the strain measured using different correlation patterns and parameters and the TSA concentrates on the interpretation of the data from different surface ply configurations. The work shows that it is possible to obtain meaningful data from the two techniques and acts as the preliminary confirmation exercise prior to full scale testing.

1. Introduction

Composite honeycomb sandwich panels are widely used in the aircraft industry as means to reduce weight. Composite panel manufacture is an expensive process, the reasons for which have been widely discussed in the literature [1-3]. The two main reasons are the labour intensive nature of the process, and to achieve high quality composites a requirement of autoclave cure. The autoclave is expensive to run, has high capital cost and forms a bottleneck in the process [4]. This study is interested in integrity assessment of composite panels produced by an out-of-autoclave process known as Resin Film Infusion (RFI). RFI uses layers of dry carbon fibre together with layers of resin film that are laid up by hand on to a tool. The whole 'stack' is cured in a conventional oven under vacuum pressure. This has the advantage of removing the expensive autoclave process and by using dry fabric that incorporates many layers in the stack, it is possible to reduce the labour hours by removing time consuming debulking.

The overall objective of the research project is to reduce the cost of composite sandwich panels for use on secondary wing structure on medium sized commercial passenger aircraft, primarily leading and trailing edge access panels. Panels of this type are currently still produced using the traditional process of hand lay-up of layers of prepreg and curing the stack in an autoclave. It is intended that the mechanical performance of panels produced using the RFI process will be compared to those produced traditionally. A generic component has been designed that is representative of panels used in secondary structures. The generic panel has a plan area of 900 x 300 mm, with a carbon fibre face sheets of thickness approximately 3 mm and a Nomex honeycomb core. In-service the wing panels are bolted to the main wing beam, but are free to deflect on one edge. The panel must withstand the aerodynamic pressure load on the surface. The load condition has been simplified for this study to a pressure load of 0.0275 MPa on the surface. A mechanical testing rig has been designed and commissioned to be used in conjunction with an Instron servo-hydraulic test machine (Figure 1) to test the panels under realistic loads. The pressure load is induced by pulling the test panel over a water filled cushion that is supported by a table structure. The water cushion allows a distributed pressure load to be applied to the panel's surface without stress concentrations. The set-up is such that it allows optical access to the displaced face sheet of the panel enabling full field optical strain measurement, as shown in Figure 1.

In this paper the full-field optical measurement techniques that are proposed to be used with the full scale rig are assessed. The strain in the sandwich structure face sheet material is measured using two techniques: Thermoelastic Stress Analysis (TSA) [5] and Digital Image Correlation (DIC) [6]. A brief description of each of the techniques as applied in this work is provided in section 2. Five types of test specimen are used in this work comprising simple unidirectional (UD) carbon epoxy to more complex woven twill materials. The test specimens and their manufacture are described in section 3 of the paper. In applying DIC to the large sandwich structure panels, as described above, a efficient means for correlation must be established. Therefore section 4 of the paper examines the effect of different correlation speckle patterns and correlation cell areas. Section 5 of the

paper examines the application of TSA to the carbon fibre epoxy specimens and assesses thermoelastic response in terms of the applied strain [7, 8].

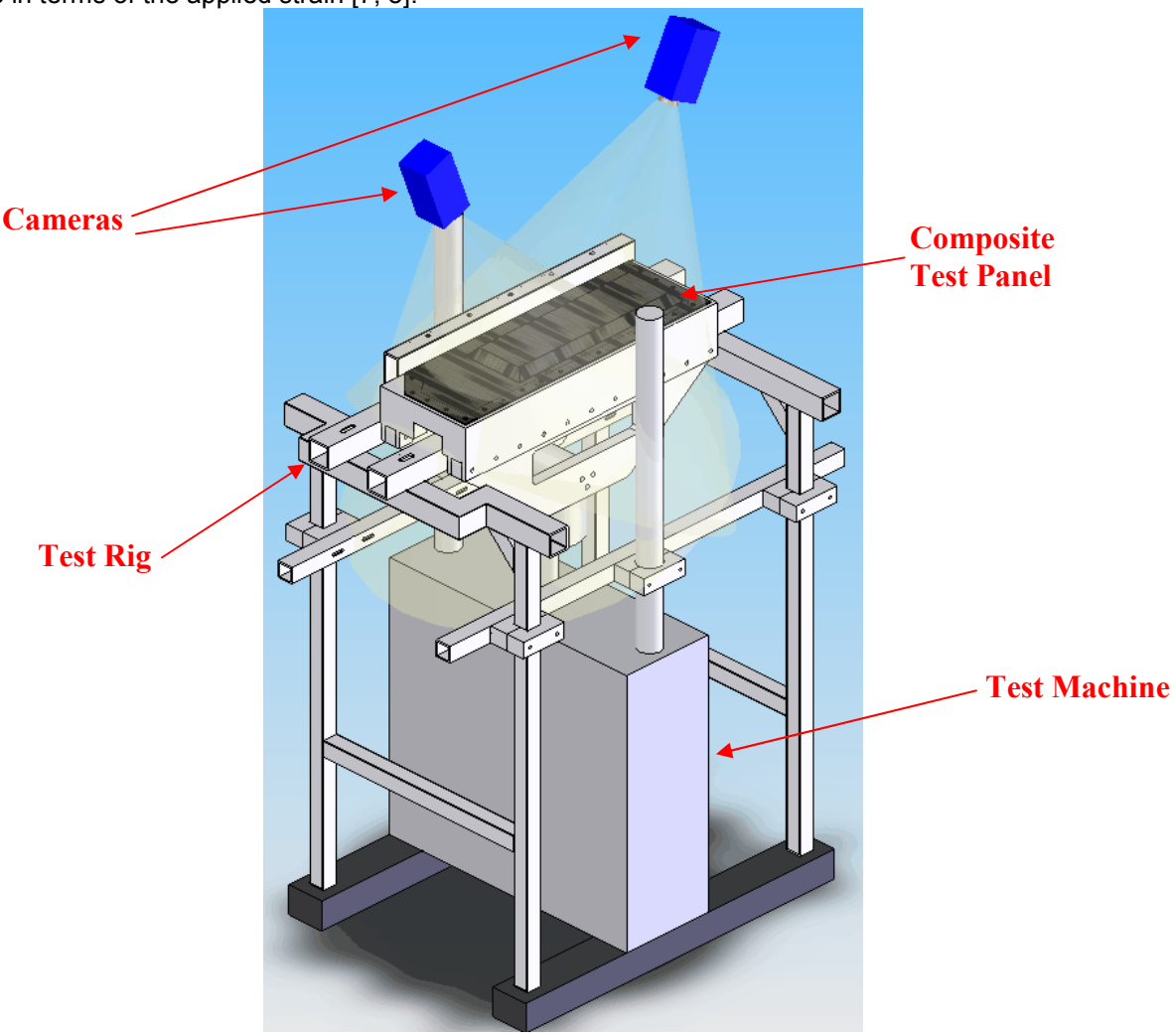


Figure 1: Pressure test rig

2. Experimental Techniques

2.1 Digital Image Correlation (DIC)

Digital Image Correlation (DIC) is a relatively new full-field, non-contact optical technique for obtaining the strain distribution across the surface of a deformed component. The technique has been used successfully for analysis of heterogeneous engineering materials such as composites [9]. DIC tracks the surface displacements of deformed structure by recognition of geometrical changes in grey scale distribution of surface patterns before and after a strain has been applied [9]. The system used in the present work was two digital cameras with 2 MPixel resolution that allows three-dimensional analysis of the deformations and measurement of the in-plane strains. The images are recorded and processed using DaVis 7.2 produced by LaVision. This system must be calibrated before it is used to produce a 'working volume', in which the surface height of the specimen from a known plane can be calculated. The processing of the images can be controlled by selecting optimum combinations of the two main processing parameters; cell size and cell overlap. An important part of the current work is to study the effect of altering these parameters is investigated.

All that is required for DIC analysis is an unstrained (reference image) and a strained (deformed image) that can be compared to the reference image [6]. The reference image is divided into 'cells', of a given number of pixels from 2 x 2 to 1024 x 1024. In Figure 2a, a 2 x 2 cell configuration with no overlap is shown. It can be seen that the undeformed image has a grey scale pattern within the 2 x 2 cell before deformation. After deformation the grey scale pattern is retained except its spatial position has moved. By recognition of the grey scale pattern from the undeformed to the deformed condition the deformation of the specimen can be obtained. In three-dimensional

analysis an extra initial step is requires where the out-of-plane deformation is obtained by calculating the distance of the surface under analysis from a calibration plane (usually in the form of plate with a known and regular geometric pattern enscribed on it); this defines the 'working volume' and the position of the two cameras relative to each other. Once this procedure has been completed, three-dimensional displacement information is obtained from which the in-plane strains are derived. The choice of cell size selection is a compromise between accuracy and spatial resolution; the larger the cell size, the more data there is to average over. A second factor to be considered in the compromise is cell overlap, as shown in Figure 2b. The strain is calculated from the deformation vectors in each cell, or in the case of overlapping data each sub-cell. This is indicated by the gauge length shown in Figure 2a and 2b; the overlap shortens the gauge length. Therefore the increase in spatial resolution shortens the distance over which the strain is measured. This inevitably will create more scatter as the measurement region decreases.

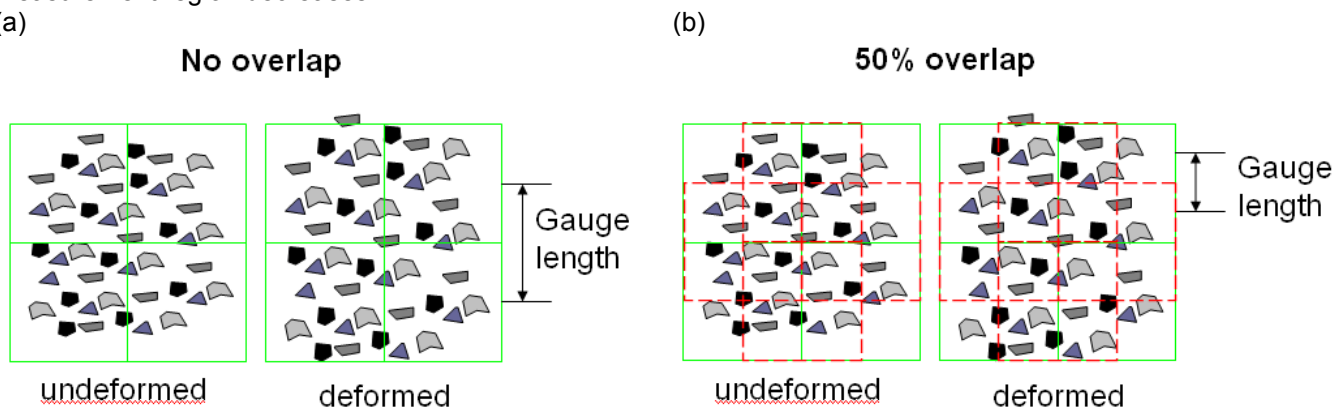


Figure 2: Cell size and cell overlap

2.2 Thermoelastic Stress Analysis (TSA):

Thermoelastic Stress Analysis (TSA) is a well-established, non-contacting technique for evaluation of stresses in engineering components, e.g. [10], where an infra-red detector is used to measure the small temperature change associated with the thermoelastic effect. The output from the detector provides the change in surface temperature, ΔT, which is related to the change in the sum of the principal stresses. For an orthotropic material, such as laminated composites ΔT can be related to the strains on the surface material, ε_L and ε_T, as follows [8]:

$$\Delta T = -\frac{T}{\rho C_p}\left[\left(\alpha_1 Q_{11} + \alpha_2 Q_{12}\right)\varepsilon_L + \left(\alpha_1 Q_{12} + \alpha_2 Q_{22}\right)\varepsilon_T\right] \quad (1)$$

where α_1 and α_2, are the coefficients of linear thermal expansion in the principal material directions, Q_{11}, Q_{12}, and Q_{22} are the laminate stiffnesses, $Q_{11} = \frac{E_1}{1-\nu_{12}\nu_{21}}$, $Q_{12} = \frac{\nu_{21}E_1}{1-\nu_{12}\nu_{21}}$, $Q_{22} = \frac{E_2}{1-\nu_{12}\nu_{21}}$, T is the ambient temperature, ρ is the density and c_p is the specific heat at constant pressure.

The overall goal of the work is to apply TSA to quantify the strains developed in the generic panels whilst loaded in the full scale rig. The application of TSA for quantitative stress/damage analysis of composite materials is not straightforward. It is necessary to calibrate the system to directly measure the surface stresses [7] and, as damage evolves the specimen heats, leading to localised temperature changes at the damage site. This affects the thermoelastic signal, but a means for correcting the temperature increase has been devised [11]. In the current work the Silver 480M system manufactured by Cedip Infrared Systems is used to collect the TSA data. This system is radiometrically calibrated so it is possible to obtain any temperature changes that occur during testing and correct as described in [11]. It is also possible to obtain directly the ΔT values given by equation (1). In the current work the application of TSA to the carbon fibre materials will be validated using equation (1) as the basis.

3. Test specimens

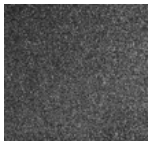
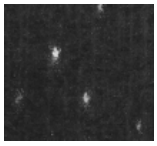
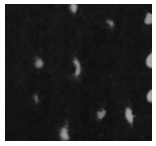
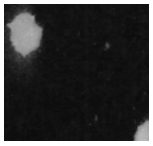
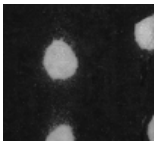
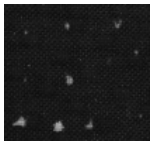
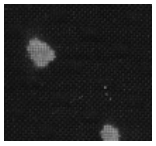
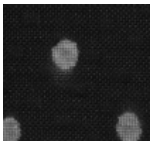
Specimens manufactured from five materials have been used in this study to investigate the feasibility of using DIC and TSA for assessing the strains in the full-scale panels. Initially flat panels of each of the generic panel face sheet materials were manufactured; these were then cut into strip specimens. Material 1 (M1), is a

unidirectional (UD) prepreg that was autoclave cured. The individual layers of prepreg are 0.125 mm thick, therefore 24 layers were required to form a 3 mm thick laminate. Two panels were made; first with all layers aligned along the longitudinal direction, therefore a UD panel, and the other with equal number of plies in 0°, 45°, -45°, and 90° directions, forming a quasi-isotropic (QI) panel. Material 2 (M2), is a five harness satin weave prepreg, again cured in an autoclave. The woven prepreg has a cured ply thickness of 0.25 mm, therefore 12 layers were required to form the 3 mm laminate. Two panels were manufactured to be similar to those produced for M1. Material (M3), is dry non-crimp fabric with separate resin film. Each layer of the dry fabric contains four UD plies stitched together in a QI configuration, therefore only 6 layers were required to form the 3 mm laminate. Only one QI panel was manufactured from this material. Material 4 (M4), is a 2 x 2 twill woven semi-preg that was cured in an oven. The lay-up and panel manufacture were similar to M2. Finally material 5 (M5), uses the same raw materials to M3, but cured in an oven. Tensile specimens were cut from the panels to dimensions from ASTM D3039. These are 15 mm wide and approximately 250 mm in length. To prevent specimen crushing in the jaws of the test machine, glass fibre end tabs were adhered to the ends of the specimens.

4. Application of DIC to carbon fibre face sheets

The DIC method discussed in section 2.1 uses a random pattern on the surface of the specimen to correlate the deformation. In the future when the large panels are tested a means of applying a suitable pattern to the panels is required. Therefore in this study various types of pattern have been investigated as shown in Table 1. Six types of random pattern are used in the analysis; a fine paint speckle, the natural surface roughness that is caused by the peel ply during manufacture, small dots with 5% coverage, small dots with 10% coverage, large dots with 10% coverage and large dots with 20% coverage. These patterns were tested on UD specimens from materials M1 and M2. To assess the effectiveness of these patterns a stain was applied to each specimen and measured using a long gauge extensometer and DIC and the results compared.

Table 1:Surface patterns for correlation

	Paint Speckle	*Natural surface*	*Small dots, 5% coverage*	*Small dots, 10% coverage*	*Large dots, 10% coverage*	*Large dots, 20% coverage*
UD						
Woven				n/a		

Firstly the optimum combination of cell size and overlap was investigated by loading M1 specimens in uniaxial tension. The surface pattern was the recommended fine paint speckle surface pattern. The specimen was strained by approximately 0.1% in an Instron servo mechanical test machine and the strain recorded by the long gauge extensometer attached to one side of the specimen with DIC data being collected from the other. The DIC recorded an image before deformation and a second at the end of the test. The two images were correlated using 10 combinations of processing parameter; the cell size was 128 or 64 pixels, while the cell overlap was set to 0%, 25%, 50%, 75% or 87%. The test was run three times to check repeatability of DIC strain values. Figure 3 plots the strain from the DIC obtained in the longitudinal direction for each processing combination normalised against the extensometer value and it can be seen that there is a variation between both the parameter combinations and the tests. It is expected that under simple tension the strain would be constant. However, by studying the strain plots, large variations were observed particularly when using larger cell overlaps. To obtain a measure of the variation centreline plots were taken and a value of coefficient of variation (COV) in each plot was calculated. Figure 4 plots the value of the COV for each processing parameter for test 1; the same trend was produced from all three tests. The COV increases markedly with cell overlap, and there is also a step change when the cell size decreased from 128 to 64 pixels. Therefore it was decided that a good compromise between accuracy and spatial resolution was a cell size of 64 pixels and an overlap of 25%.

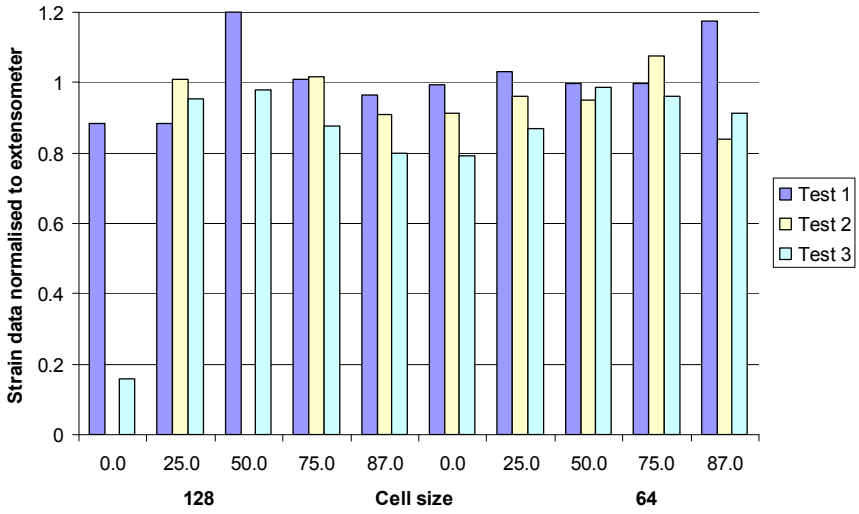


Figure 3: Effect of varying processing parameters

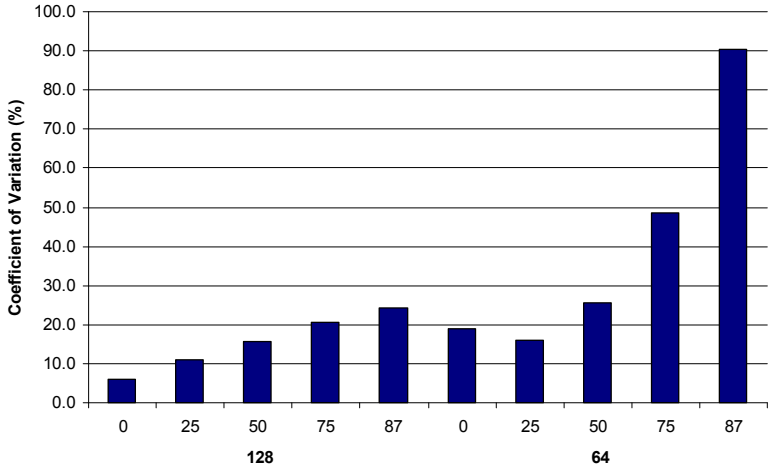


Figure 4: Coefficient of Variation

Having established the cell size and overlap, the next step was to assess the different surface patterns shown in Table 1. Once again the specimens were strained by about 0.1% in the test machine and the longitudinal strain obtained from the DIC normalised to that from the extensometer. Figure 5 plots the normalised strain data for each pattern on both M1 and M2. The paint speckle pattern showed 3.1% error in comparison to the extensometer for the UD specimen and 1.4% error for the woven specimen, while the natural pattern gave 0.7% and 2.8% error for the UD and woven specimens respectively. However the other paint patterns produced errors in excess of 10% and as such were discarded. For the purposes of the full scale tests, it will be possible to use the natural surface pattern on the panels to correlate the strain. This will remove the need for time consuming and problematic application of fine paint speckle.

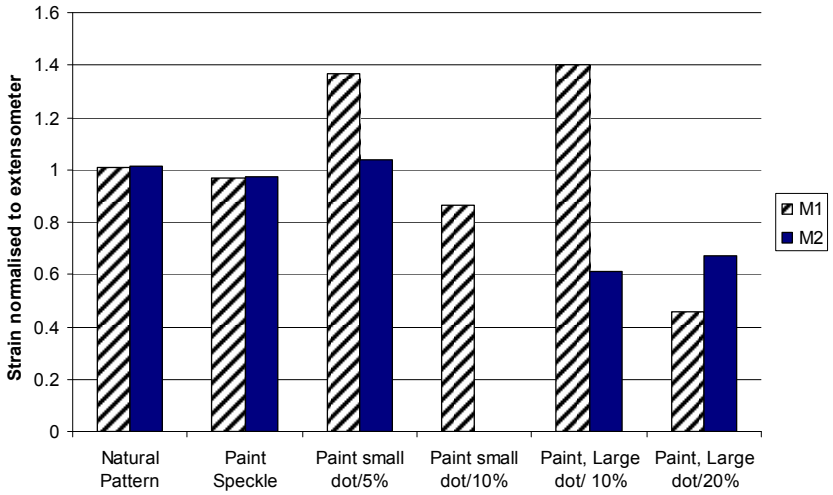
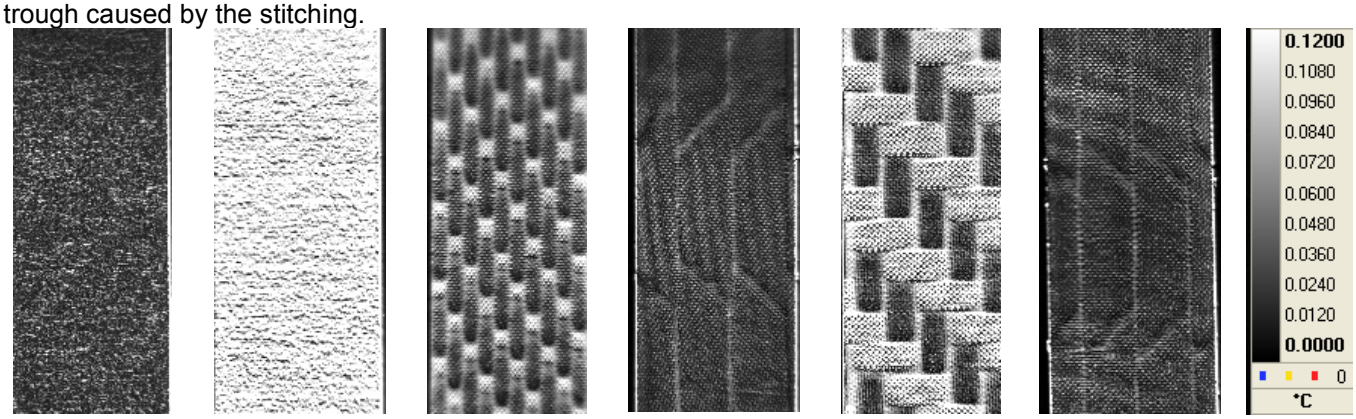


Figure 5: Normalised strain data for the UD specimens

5. Application of TSA to carbon fibre face sheets

To investigate the possibility of using the TSA technique to measure full field strain from the orthotropic materials, a Silver 480M system (manufactured by Cedip Infrared Systems) was used. Specimens made from material M1-M5 were loaded in uniaxial cyclic tension to a level of 3.5 ± 3 kN at a frequency of 10 Hz in an Instron servo-hydraulic test machine. ΔT was obtained from the surface of each specimen using the Cedip system. To obtain temperature measurements from the Cedip system it was necessary to input an emissivity. A value of 0.92 was chosen because the surfaces of the carbon fibre specimens are matt black. Figure 6a shows the TSA image of a quasi-isotropic laminate manufactured from M1 with a surface ply at 0°, which is practically uniform with a ΔT of 0.0351 °C. Figure 6b shows the TSA image of a similar quasi-isotropic laminate from M2, but with a surface ply at 90°; here the response is also uniform but much greater with a ΔT of 0.1253 °C. Figures 6c and 6e are from the woven composite materials M2 and M4 and the tows at 0° show a similar response to that in Figure 6a and the tows 90° show a similar response to Figure 6b Figures 6d and 6f are for the NCF material of M3 and M5. The surface is uniform and the response is similar to that given by Figure 6 for the UD material, however the stitching in the NCF to hold the plies together is evident in the image, possibly because the resin has accumulated in the trough caused by the stitching.



(a) (b) (c) (d) (e) (f)

Figure 6: TSA images of tensile specimens; (a) M1 with surface ply at 0°, (b) M1 with surface ply at 90°, (c) M2, (d) M3, (e) M4, (f) M5

It should be noted that these specimens were subjected to a constant applied stress of the same value. However this is a global measure and the TSA provides values that are related to the stress in the surface ply. In order to compare the response it was decided to obtain the strain sum in each specimen, which should be constant through the thickness and apply equation (1) to obtain a theoretical ΔT for each of the ply orientations. The cyclic

stress applied to each specimen is known, therefore using a measured global Young's modulus, E_L for each material it was possible to calculate ε_L. The ε_T value was calculated by multiplying ε_L by a measured value for ν_{LT} (see Table 2). To calculate Q_{11}, Q_{12}, Q_{22}, the laminate stiffness, it was assumed that the individual fibre tows had the same UD properties as the UD laminate produced from M1. M1 UD had a longitudinal stiffness (E_1) of 134 GPa, transverse stiffness (E_2) of 8.99 GPa, ν_{12} of 0.316 and ν_{21} of 0.048. Therefore, Q_{11} was calculated to be 1.36 x 10^{11}, Q_{12} to be 6.53 x 10^9 and Q_{22} to be 9.13 x 10^9. Finally values for C_P, and α_1 and α_2 were found in the literature. C_P was taken to be 1130 J/K °C, [12], while α_1 was taken to be -0.9 x 10^{-6} /°C and α_2 taken to be 27 x 10^{-6} /°C [13]. The values for the coefficient of linear thermal expansion were selected to be representative of a carbon fibre with similar stiffness and strength values. Table 2 presents both measured (m) and calculated (c) surface temperature changes for comparison.

Table 2: Comparison of measured (m) against calculated (c)

Spec	Δσ MPa	E_L GPa	ν_{LT}	ε_L	ε_T	$\varepsilon_L+\varepsilon_T$	ΔT_L		ΔT_T	
							m	c	m	c
M1 QI 0°	122.8	48.7	0.086	0.0025	0.00022	0.00274	0.0351	0.0307	n/a	n/a
M1 QI 90°	124.4	50.4	0.149	0.0024	0.00037	0.00284	n/a	n/a	0.1253	0.1002
M2 QI	116.5	47.1	0.262	0.0024	0.00065	0.00312	0.0387	0.0472	0.0971	0.1028
M3 QI	113.2	44.5	0.317	0.0025	0.00081	0.00335	0.0327	0.0540	n/a	n/a
M4 QI	105.7	42.2	0.241	0.0025	0.00060	0.00310	0.0329	0.0457	0.1119	0.1037
M5 QI	112.8	45.2	0.32	0.0025	0.0008	0.0033	0.0326	0.0533	n/a	n/a

The measured change in temperature for M1, with 0° surface ply, compares well with the calculated value, even though such low values are towards the bottom end of the thermal resolution for the system. Similarly the temperature change for M1, with 90° surface ply, is in excellent agreement. In these two cases the material elastic properties in equation (1) have been obtained using identical material, the other properties have been estimated from the literature. For the woven materials M2 and M4 ΔT_L differ by 23% and 36% respectively with the calculated value being greater in both cases. ΔT_T for M2 and M3 are in much closer agreement. The only explanation for this is that the estimation of stiffness parameters based on a UD material does not sufficiently model the longitudinal weave. With the non-crimp M3 and M5 materials ΔT_L have a larger error or almost 40%. It is noteworthy that in all cases the experimental values ΔT_L are within 16% of each other. Further investigations of the reasons for the differences are underway.

6. Conclusions:

This paper has presented work completed on a feasibility study on the use of Digital Image Correlation (DIC) and Thermoelastic Stress Analysis (TSA) on composite sandwich face sheet panels. These techniques are to be used on full-scale tests of panels under a pressure load. The study has made use of specimens loaded in uniaxial tension, manufactured from five different carbon fibre epoxy composite materials that have been described. Tests were performed using the DIC system to compare measured strains taken with an extensometer. These were used to investigate the type of surface speckle pattern to be used for correlation and the combination of processing parameters for the strain calculations. It was found that the natural surface roughness caused by the peel-ply during the composite manufacture was as good as a fine paint speckle that is recommended for use with the correlation system. This negate the need to apply the paint speckle to the full scale panels that would possibly be time consuming and problematic. For a compromise between accuracy and spatial resolution it was decided to use a cell size of 64 pixels and a cell overlap of 25%. The TSA system was used to measure the surface temperature change of specimens under a cyclic load. The predictions for the temperature change generally compared favourably with the measured values. Of particular interest was the observation that a similar thermoelastic response was obtained from the longitudinal and transverse tows of a woven composite to that of longitudinal or transverse surface plies of non-crimp UD material.

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