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Modal strain energy-based structural health monitoring validation on rib stiffened composite panels

Jason S. Hwang[a,c]*, J. Marcelo Müller[b], Richard Loendersloot[c] and Tiedo Tinga[c]

[a]*Gas Turbines and Structural Integrity Department, Netherlands Aerospace Centre NLR, P.O. Box 90502, 1006 BM Amsterdam, The Netherlands;* [b]*Structures Technology Department, Netherlands Aerospace Centre NLR, Amsterdam, The Netherlands;* [c]*Dynamics Based Maintenance Group, University of Twente, Enschede, The Netherlands*

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Structural Health Monitoring (SHM) techniques are emerging more and more in aerospace composite structure applications. Validation of the SHM techniques is one of the issues to be addressed yet. Moreover, cost-efficient and reproducible ways to compare SHM methods can be attractive for researchers and end users. NLR has recently developed ways for automated production of rib-stiffened composite panels using an Automated Fibre Placement (AFP) machine. Without human interference and a fully automated manufacturing process, multiple panels can be cost-effectively laid-up with limited quality variation. In this paper, Modal Strain Energy Damage Index (MSE-DI) method is applied on two impact-damaged panels manufactured in this new way. The results show that the panels have slightly deviating natural frequencies even though the C-scan revealed no major quality variation. Secondly, the presence of the impact damage can be successfully detected by the MSE-DI algorithm, but the localization was inaccurate, contradicting the expectations. This is attributed to the specific characteristics of the panel and indicates a dependency of the performance of algorithms on structure to which they are applied. Future work will include a further exploration of this dependency.

Keywords: structural health monitoring; composite damage; modal strain energy damage index; automated fibre placement; composite aircraft structure

Introduction

Maintenance strategies in various fields of industry, including aerospace applications, are shifting from time scheduled to condition-based strategies. Structural Health Monitoring (SHM) techniques can play an important role in providing more concise information about the integrity of the target structure, while inspection time can be reduced. Within aerospace applications, impact damage monitoring on composite rib-stiffened panel structures imposes high potential for SHM techniques [1–3]. However, the number of SHM applications in the field of aerospace is still limited. One reason for this is the unknown performance of each individual SHM approach related to various damage cases. An effective way to compare different SHM techniques is to evaluate their performance with identical structures to which various damage cases are applied. The first challenge is to manufacture identical structures, which for composites is not a triviality.

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*Corresponding author. Email: jason.hwang@nlr.nl

Fully automated composite structure manufacturing can be attractive to make test objects with a limited variation in their internal structure. When relatively simple stiffened panels can be manufactured without human interference, quality variation of the manufactured panel will be small and the panel will be cost effective to produce. In the European project LOw COst Manufacturing and Assembly of Composite and Hybrid Structures (LOCOMACHS) [4,5], a method to reduce the amount of tooling and labour necessary for manufacture of rib-stiffened panels was investigated. The knowledge and experience that NLR has gained within that project is used to produce a series of fully automated rib-stiffened composite panels for SHM technique validation purposes.

This paper describes how such panels can be used for SHM comparison study. First, the manufacturing process of the rib panel is presented. Secondly, the description is given for the two panels with each one single impact damage. Thirdly, the modal strain energy-based damage identification method is explained in more detail. Fourthly, the damage identification method has been applied to both panels to see if the damage could be identified successfully. This paper concludes with a discussion on the results.

Test article

Manufacture of stiffening elements in composite structures usually requires different process steps and associated tooling and labour. Often, the composite skin and stiffening elements are manufactured separately and either subsequently co-cured in one single cycle, co-bonded or secondary bonded after cure. Complex tooling is necessary during laminating of the components as well as additional tooling during cure. The traditional manufacturing methods involve a reasonable amount of touch labour to either cut the materials, lay-up the plies and assemble the preform. Besides being labour intensive and thus relatively costly, the complex sequence of manufacturing can be subjected to variation due to human errors. Automation of the manufacturing process can reduce cost and improve reproducibility as well, leading to less variation in mechanical properties.

Rib stiffening

Grid stiffened structures, also known as grid, orthogrid, or isogrid stiffend structures, have been investigated since the 1970s, particularly in space structures [6,7]. The rib stiffening manufacturing method discussed in this article uses Automated Fibre Placement (AFP) to automatically place layers of a single thermoset tape on top of each other, creating a stack of tape that can serve as a stiffener. Two rib stiffened panels were manufactured using AFP. The chosen skin lay-up was semi-isotropic to keep the structure as generic as possible. See Table 1 for the technical specifications. Figure 1(A) shows the cured panel, while (B) shows the areas where the thicknesses are measured

Table 1. Specification of the test panels (all dimensions are given in mm).

Tow[1] width	Length	Width	Lay-up	No. of plies	Remarks[2]
6.35	600	400	$[45/90/-45/0/45/90/-45/0]_S$	16	Skin
6.35	600	12,7	[0]	18	Rib feet
6.35	600	6.35	[0]	147	4 Ribs

[1] Prepreg: CYCOM 5320-1FI/IM7.

[2] Panels: Two identical panels were fabricated with ID number 5565 and 5567.

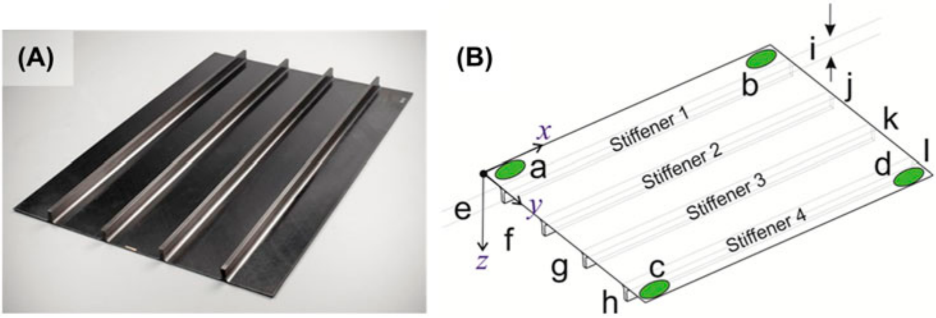


Figure 1. (A) Rib-stiffened co-cured panel (B) Thickness measurement area.

after curing. Table 2 presents the thickness measurements on two panels. Notice that panel 5567 is slightly thinner than panel 5565.

Non-destructive ultrasonic C-scan inspection after fibre placement and cure is performed on both panels, see Figure 2 for the composition picture of the results. These pictures show that both panels have some voids in the skin between the stiffeners. Skin-stiffener connections show no significant voids. Based on the thickness measurements and the C-scan results, it can be concluded that the panels show still some quality variation.

Method

In this study, the MSE-DI was implemented according to [8]. MSE-DI falls under the category of vibration-based modal-domain damage feature extraction methods, utilizing curvatures of the mode shape. The displacement mode shape is used to derive dynamic strain and the mode shape curvatures. Since the bending stiffness of the panel in y-direction is dominant, the panel is modelled as a beam-like structure, allowing to use the 1D formulation of the modal strain energy equation [9]. This assumption leads to modal the strain energy U as:

$$U_i^{(n)} = \frac{1}{2}\int_{x_{i-1}}^{x_i} EI_y \left(\frac{\partial^2 u_z^{(n)}}{\partial x^2}\right)^2 \mathrm{d}x \approx \frac{1}{2} EI_y \int_{x_{i-1}}^{x_i} \left(\frac{\partial^2 u_z^{(n)}}{\partial x^2}\right)^2 \mathrm{d}x = \frac{1}{2} EI_y \gamma_i^{(n)}, \tag{1}$$

where EI_y stands for bending rigidity in y-direction (see Figures 1 and 3 for the sign convention), $U_i^{(n)}$, $u_z^{(n)}$ and $\gamma_i^{(n)}$ stand for the strain energy, displacement in z-direction at the element i and the right-hand side nth mode shape integral respectively. Assuming that the beam can be divided in N elements, $U_i^{(n)}$ is calculated for each element. By dividing each $U_i^{(n)}$ over the total sum of the modal strain energy in the beam, the ratio of the fractional modal strain energy can be extracted. The DI is extracted by comparing the fractional modal strain energy for each element and mode shape before and after impact loading is applied to the structure:

$$\beta_i = \sum_{n=1}^{N} \left(\frac{\tilde{\gamma}_i^{(n)}}{\tilde{\gamma}^{(n)}}\right) \Big/ \sum_{n=1}^{N} \left(\frac{\gamma_i^{(n)}}{\gamma^{(n)}}\right). \tag{2}$$

Table 2. Dimensions in mm measured on different area of the panels (see Figure 1(B)).

	Skin thickness				Stiffener height							
Location	**a**	**b**	**c**	**d**	**e**	**f**	**g**	**h**	**i**	**j**	**k**	**l**
Panel 5565	2.14, 2.16, 2.16	2.16, 2.17, 2.15	2.16, 2.17, 2.16	2.13, 2.13, 2.15	24.37	24.04	23.81	24.59	23.50	23.70	24.58	24.10
Panel 5567	2.13, 2.10, 2.10	2.11, 2.08, 2.10	2.13, 2.11, 2.08	2.06, 2.08, 2.07	24.25	24.37	23.65	23.76	23.50	23.83	23.54	23.21

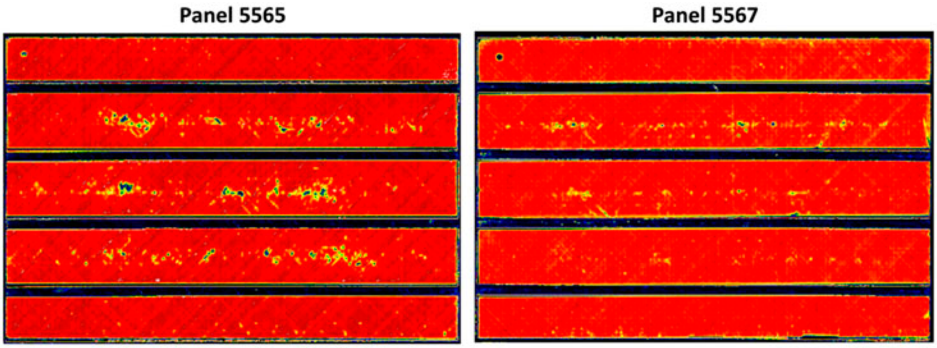


Figure 2. Ultrasonic C-scan inspection results on the skin and rib skin. This is a composition picture of multiple inspection results.

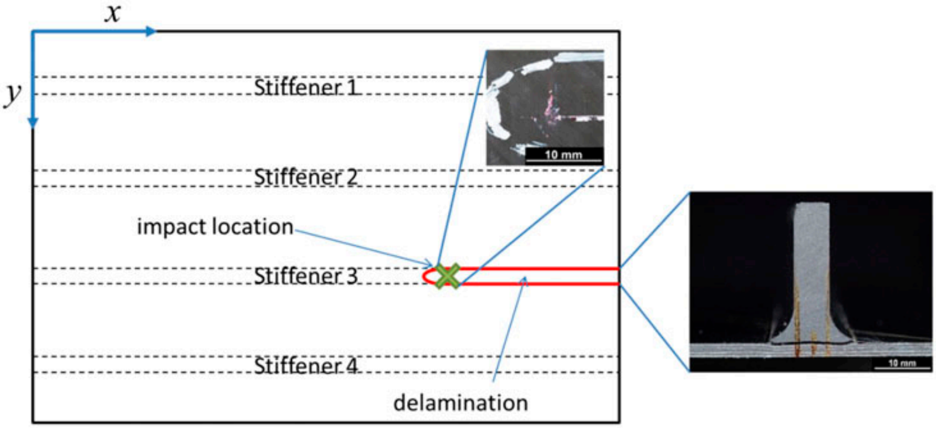


Figure 3. Schematic drawing of the rib-stiffened panel. Delamination size and location are similar to both panels.

The tilde indicates the damaged state. $\gamma^{(n)}$ and $\tilde{\gamma}^{(n)}$ stand for the integral over the whole length of the beam. Statistically relevant outliers in the DI can be extracted by:

$$Z_i = \frac{\beta_i - \bar{\beta}}{\sigma}, \tag{3}$$

where $\bar{\beta}$ and σ stand for the average and standard deviation of the DIs for all mode shapes and elements, respectively. In general, a minimal damage detection threshold can be set as Z_i larger than two.

Impact loading

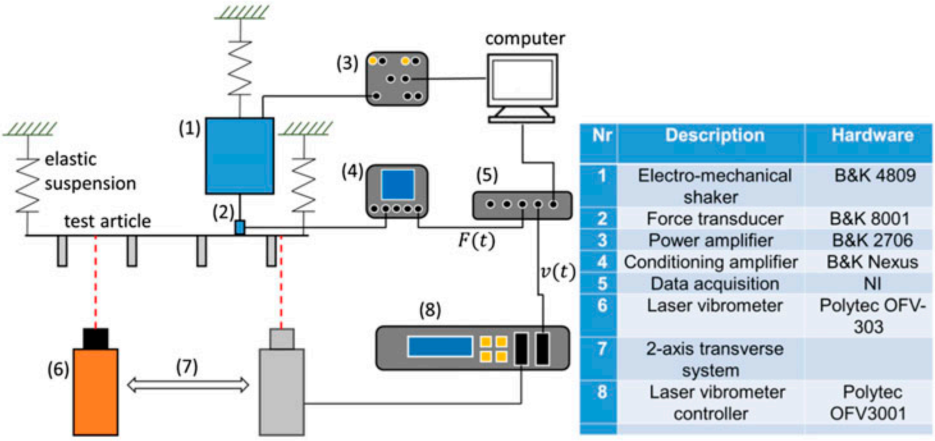
Panels 5565 and 5567 have been subjected to a single impact load each. The panels were clamped on four edge sides, while stiffeners were kept unsupported such that a typical aircraft skin-stiffener condition is created. The impact was applied on the skin side, directly underneath stiffener 3, see Figure 3. The impact was applied with a hemispherical metallic head with a diameter of 12.7 mm and a mass of 2.267 kg. The impact location on both panels is identical: $x = 450$ mm and $y = 250$ mm. The impact energies on panels 5565 and 5567 were registered to be 17.8 and 13.7 J, respectively. Nondestructive inspection with an ultrasonic probe revealed that both impacts have caused a similar delamination at the skin-stiffener connection. The impact loading on the outer skin side has caused no visible damage from the outside. However, skin-stiffener connection is deteriorated significantly over a length of 150 mm, see Figure 3.

Experimental setup

Before and after impact loading, the natural frequencies and the Operational Deflection Shapes (ODS) are extracted by using a Laser Doppler Vibrometer (LDV) and an electro-mechanical shaker. The complete dynamic set-up and data acquisition scheme used for the experiments are presented in Figure 4. The panel has been suspended using two thin rubber bands attaining a free-free mounting condition. The electro-mechanical shaker has been coupled to the panel with a slender rod and a circular disc glued on the skin. The shaker has been aligned perpendicular to the surface avoiding the introduction of in-plane force as much as possible. Furthermore, the shaker has been suspended with a spring to preserve a free-free condition. The excitation consisted of series of five sine sweeps from 200 to 4000 Hz, while the LDV measured the out-of-the-plane velocity with the sampling frequency of 48 kHz. The measurement grid consisted of 23 × 11 points.

Results

The natural frequencies and their corresponding ODS have been extracted from both plates before and after the impact loading. The first eight relevant natural frequencies are listed in Table 3. The amplitude normalized mode shapes of the bending modes parallel to the stiffeners (parallel bending mode hereafter) are taken for the normalized damage index calculation, following the approach of Ooijevaar [9].



Nr	Description	Hardware
1	Electro-mechanical shaker	B&K 4809
2	Force transducer	B&K 8001
3	Power amplifier	B&K 2706
4	Conditioning amplifier	B&K Nexus
5	Data acquisition	NI
6	Laser vibrometer	Polytec OFV-303
7	2-axis transverse system	
8	Laser vibrometer controller	Polytec OFV3001

Figure 4. Experimental test setup.

Table 3. The first eight natural frequencies.

Mode	Panel 5565 Pristine	Panel 5565 Damaged	Panel 5567 Pristine	Panel 5567 Damaged
1st *x*-bending	220	214	215	210
1st *x*-shear	266	245	259	249
2nd *x*-bending	468	458	456	445
2nd *x*-shear	528	522	514	501
1st *y*-bending	557	556	533	542
3rd *x*-shear	640	638	623	613
3rd *x*-bending	1018	–	998	–
2nd *y*-bending	1209	1121	1170	1139

Figures 5 and 6 present the ODS of the panel 5567 indicating the first two parallel bending modes and the calculated MSE-DI for both panels 5565 and 5567, respectively. Notice that some natural frequencies could not be determined properly. Furthermore, the natural frequency of first parallel bending mode seems to have gone up after the damage. This mode may have been influenced by different boundary condition caused by the different tension in the elastic chord suspension.

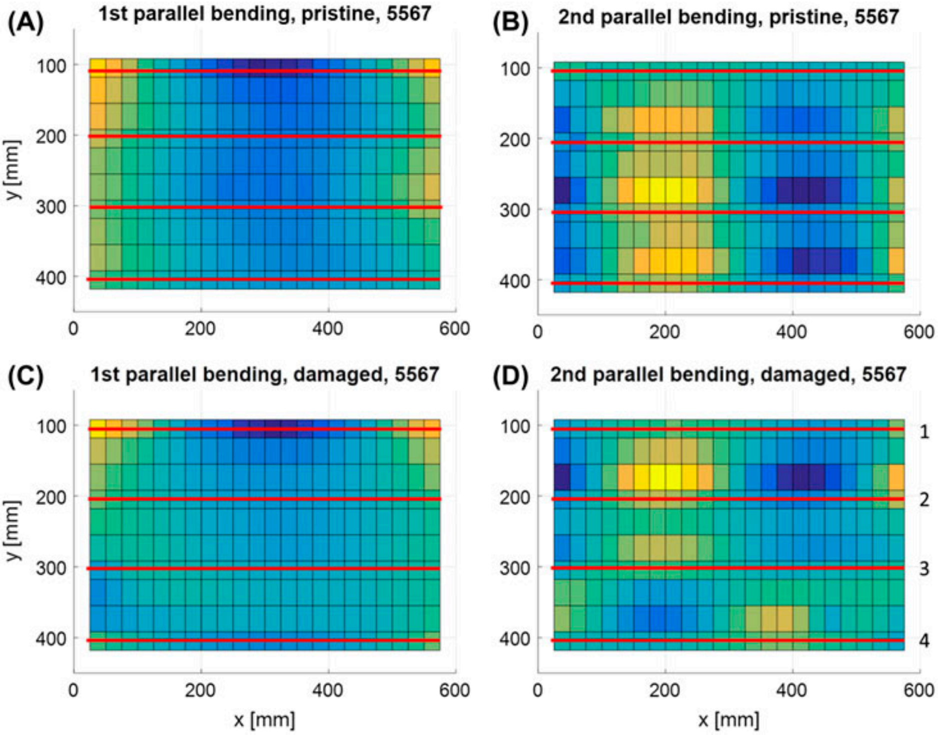


Figure 5. ODS of the plate 5567, relevant for the parallel bending moment mode shapes. (A) and (C) represent 1st bending moment before and after the impact damage, respectively, (B) and (D) represent second bending moment before and after the impact.

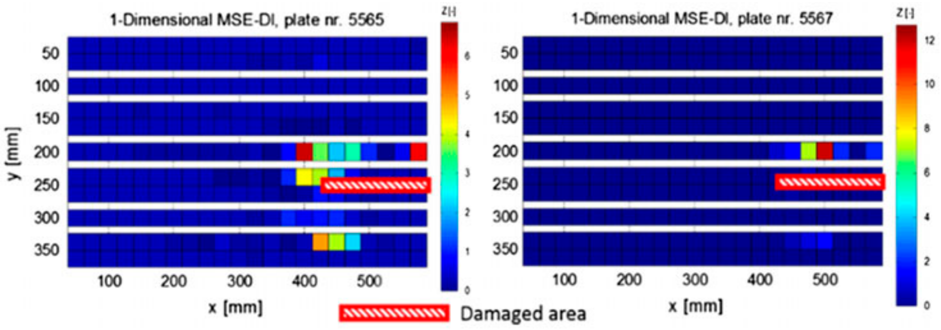


Figure 6. MSE-DI extracted from panel 5565 and 5567.

According to Ooijevaar et al. [9], the mode shapes that are most sensitive to the skin-stiffener delamination are the bending modes in the direction of the stiffeners. Therefore, the bending modes parallel to the stiffeners are used to calculate the damage indices. However, this panel is, compared to the work of Ooijevaar et al., much stiffer, due to both skin thickness and stiffeners dimensions. This causes small displacements of the relevant mode shapes, resulting in limited measurements of the mode shapes.

Figure 6 shows that damage has been detected successfully on both panels, although the peaks appear at different locations, indicating inaccurate damage localization. The highest normalized damage index is found adjacent to the damaged area between stiffeners 2 and 3. However, a number of additional peaks in the MSE-DI appeared in panel 5565, despite the impact being applied at the same location. The modal strain energy is based on the second-order spatial derivative of the mode shapes and therefore is even more inaccurate. The limited number of measured ODS has contributed even more to the inaccuracy.

Discussion and future work

Two panels were manufactured for SHM technology demonstration and validation purposes. C-scan inspection showed that both panels have some thickness variety as well as some porosities in the skin. The natural frequencies of the panels show slight deviation to each other, which can be explained by the thickness difference. Even though the quality variation of the manufactured panel is small by lack of human interference, the dynamic properties of a panel can still vary, which has to be taken into account when an SHM system is implemented. In the LOCOMACHS project, another set of panels were manufactured using a different composite material system [4] than the one described in this paper. Panels with better C-scan results were obtained.

The MSE-DI results show that the damage localization is inaccurate. The considerations in [9] apparently do not apply for this structure, where the stiffeners are overdimensioned. Unidirectional strain measurements parallel to the stiffener will help to extract the bending mode shapes parallel to the stiffeners more accurately, since the bending modes perpendicular to the stiffeners will not or hardly influence these measurements. A candidate sensor network for this purpose may be for example, optical fibres with Fiber Bragg Gratings (FBG) engraved inside.

Using these panels, various SHM techniques can be compared to each other under operational-relevant environments. The panels are designed such that they fit into the SHM test bench which can apply bi-axial loading to the panels while the climate chamber can be fit around the panel during the test. This gives unique opportunities to study performances of various damage feature extraction methods and sensor networks.

Disclosure statement

No potential conflict of interest was reported by the authors.

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ORCID

Jason S. Hwang http://orcid.org/0000-0001-8677-0513

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