Composite Structures 177 (2017) 119–128

Contents lists available at ScienceDirect

Composite Structures

journal homepage: www.elsevier.com/locate/compstruct

Laminate design for optimised in-plane performance and ease of manufacture

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ARTICLE INFO

Article history:
Received 12 November 2016
Revised 23 June 2017
Accepted 26 June 2017
Available online 28 June 2017

ABSTRACT

New structural efficiency diagrams are presented, showing that current design practice incurs additional mass because: (i) laminate balancing axes are not aligned with principal loading axes and (ii) principal loading ratios vary within a part with fixed ply percentages. These diagrams present significant opportunities for fibre steering and laminate tailoring in aerospace design. Moreover, it is shown that standard ply angles (0°, +45°, −45° and 90°) have incompatible modes of deformation between adjacent sublaminates in their uncured state (during forming); such modes can promote the occurrence of wrinkling defects during manufacture which reduce part strength significantly. A new formulation is presented to enable any standard angle laminate to be replaced by a laminate consisting of two non-standard angles, $\pm\phi$ and $\pm\psi$, with equivalent in-plane stiffness. Non-standard ply angles are shown to promote compatible modes of deformation and offer significant potential, in terms of formability, thereby increasing production rates and reducing the need for so-called manufacturing knockdown factors.

1. Introduction

Netting analysis [1], in which fibres only carry load in their longitudinal direction and the resin matrix is ignored, indicates that designs with fewer than three fibre directions produce mechanisms when subject to small disturbances in loading. This reveals the robustness of established aerospace laminate design practice which uses four standard angles (0°, +45°, −45° and 90°) to provide a level of redundancy in load carrying whilst allowing for the manufacturing requirement of balanced angle plies. Laminates for aerospace components are currently designed using standard ply angles whilst following established design rules [2]. These rules include: ply angle symmetry about the laminate mid-plane, equal numbers (balancing) of +45° and −45° angle plies, 10% thickness in each of the 4 ply angles, and ply blocks of identical angles must be a maximum of 1mm. Additionally, ±45° plies are usually positioned at the outer surface for enhanced damage tolerance. The percentage of 0°/±45°/90° plies in a laminate is a function of the typical loading a component will carry; for example in wing skins, stiffeners and wing spars, target percentages are typically 44/44/12, 60/30/10 and 10/80/10, respectively. Unfortunately, such rules do not account sufficiently for manufacturing processes and can also limit the possible laminate designs that have the required curvature-stable manufacturability and stiffness coupling [3,4]. Restriction of ply orientations to the four standard fibre angles can also contribute to the development of manufacturing induced defects during the curved laminate forming process. This is because, in its pre-cured state, the resin matrix has an extremely low transverse modulus and so the unidirectional fibres within layers either separate (are pulled apart) or rotate in shear (scissor) to enable a change in geometry. The general scissoring behaviour of cross-plied UD can be modelled by using pin-jointed-net theory [5], originally created to analyse shearing of woven fabric. Limitations to formability arise in the fibre direction, where fibres cannot extend, nor can they resist compression without causing a buckling (wrinkling) defect. The combination of all four standard angles within a laminate therefore makes it difficult to form the laminate into a curved shape from flat. Indeed Hallender et al. [6] discovered that defects occurred during forming of a C-Section spar when +45° and −45° plies were separated by a 0 ° ply. In contrast wrinkling defects were not produced when ±45° and 0/90° plies were grouped separately as these separate groups were able to deform independently, as shown in Fig. 1. Hence, the properties of ply groups within the laminate (sublaminates) were seen to be a critical feature of formability and can be linked to the compatibility of sublaminate modes of deformation.

Clearly, any shift in design practice toward non-standard angles cannot come at the cost of laminate performance, where laminate tailoring and tow-steering are pushing the boundaries of minimum mass composite structural design [8]. Efforts are also being made

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http://dx.doi.org/10.1016/j.compstruct.2017.06.061
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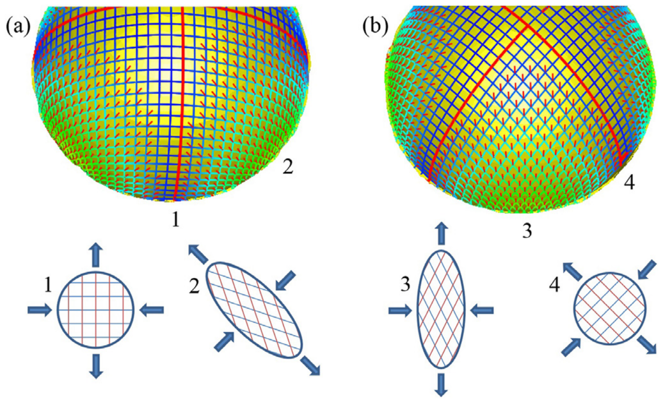

Fig. 1. Forming of (a) 0/90° and (b) ±45° plies over a sphere (after [7]) and local modes in regions 1–4. The shearing of fibres is orthogonal in regions 1 (2) and 3 (4) and so forming needs to allow slip between 0/90° and ±45° plies.

to make the composite laminate design process more straightforward [9].

Fig. 2 illustrates the conflict between manufacturing and performance in an energy landscape. The laminate manufacturing process imposes a fixed deformation on the uncured laminate arising from consolidation of complex parts or by forming such parts from an initially flat state. The objective is both to minimise the strain energy U_m for an imposed strain, and to avoid orthogonality in these low energy sublaminate modes. In contrast, improved performance requires minimisation of strain energy for an imposed stress, see U_p in Fig. 2.

Minimisation of elastic strain energy allows laminates to be designed that store the least energy, creating the stiffest configuration for a given design loading, see Fig. 2. Prager and Taylor [10] first outlined optimality criteria justifying the technique of minimisation of elastic energy to produce a structure with optimal efficiency. Pedersen [11] subsequently applied this technique to composite materials to find analytical solutions for orientation of a single ply angle subject to in-plane loading. This is a logical design concept as the material is made to work as hard as possible to resist deformation under load, and thus is efficiently used, potentially allowing lower mass designs to be produced. However, such design does not directly convert to minimum mass, as failure is nonlinear and complex; comprising damage to both resin and fibre, which is induced by mechanisms such as delamination [12], buckling [13], bearing, edge effects [14] and manufacturing defects.

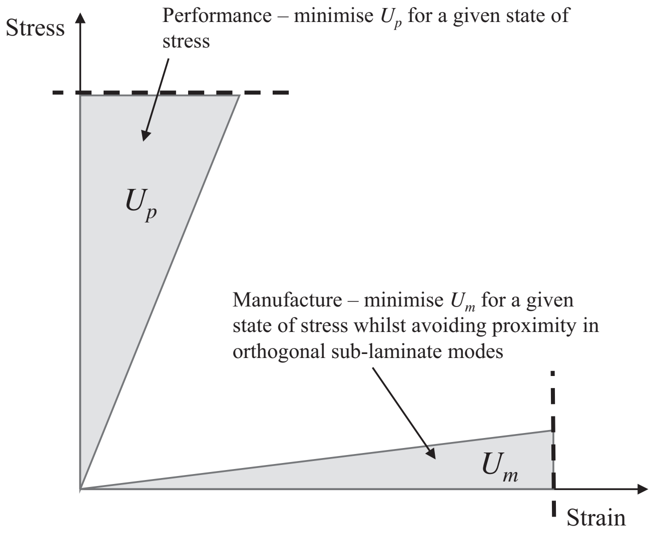

Fig. 2. Illustration of laminate strain energy relating to (cured) performance U_p and (uncured) manufacture U_m. Note that minimisation of U_p maximises the load-carrying capacity of fibres whilst minimisation of U_m maximises deformation by resin-dominated modes.

In this paper, elastic energy is considered to be an indication of performance to assess the potential of different design approaches to reduce laminate mass. Further to the above, manufacturing constraints mean that lay-up axes and principal loading axes are not necessarily aligned. Hence results are presented to illustrate the effect of aligning (and misaligning) the laminate balancing axes with the principal loading axes. Thus, in combination with a new method for finding non-standard ply angles that match the in-plane stiffness of standard ply angles, a simple strain energy (compliance) minimisation is used to compare performance of standard and non-standard laminates. Eigenvalues and eigenvectors of uncured sublaminate stiffness matrices are used to describe compatibility between the resin-dominated modes of sublaminate deformation and thus indicate the ease of manufacture of laminates in their uncured state. The potential for non-standard plies to improve sublaminate deformation compatibility is also explored.

2. Theory

The following outlines the theory required (i) to create non-standard ply laminates with matching in-plane stiffness to standard ply laminates and (ii) to assess the comparative manufacturability and performance of standard and non-standard ply laminates, where performance is qualified by elastic energy.

2.1. Equivalent representations of ply and laminate stiffness

The material specific in-plane stiffness of a single ply, linking in-plane stress components $\{\sigma\}$ to in-plane strain $\{\varepsilon\}$, is given by Classical Laminate Theory as

$$\{\sigma\} = \begin{Bmatrix} \sigma_{11} \\ \sigma_{22} \\ \tau_{12} \end{Bmatrix} = [C]\{\varepsilon\} = \begin{bmatrix} C_{11} & C_{12} & 0 \\ C_{12} & C_{22} & 0 \\ 0 & 0 & C_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_{11} \\ \varepsilon_{22} \\ \gamma_{12} \end{Bmatrix} \tag{1}$$

$$\left.\begin{matrix} C_{11} = \frac{E_{11}}{1-\nu_{12}\nu_{21}} & C_{22} = \frac{E_{22}}{1-\nu_{12}\nu_{21}} \\ C_{12} = \frac{\nu_{12}E_{22}}{1-\nu_{12}\nu_{21}} = \frac{\nu_{21}E_{11}}{1-\nu_{12}\nu_{21}} & C_{66} = G_{12} \end{matrix}\right\} \tag{2}$$

where E_{11} and E_{22} are longitudinal and transverse stiffnesses respectively, ν_{12} and ν_{21} are major and minor Poisson's ratios respectively, and G_{12} is the shear modulus. Subscripts 1 and 2 relate to local ply axes as shown in Fig. 3.

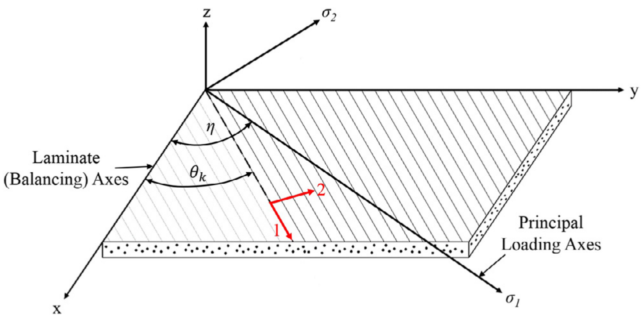

Fig. 3. Local ply axes (1–2) and laminate (balancing) axes (x-y-z). The principal loading axes and misalignment angle η from the balancing axes are also shown.