IMPACT OF COMPOSITE MANUFACTURING CONSTRAINTS ON AEROSPACE STIFFENED PANEL DESIGN

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Abstract

This paper reports on a design study assessing the impact of laminate manufacturing constraints on the structural performance and weight of composite stiffened panels. The study demonstrates that maximizing ply continuity results in weight penalties, while various geometric constraints related to manufacture and repair can be accommodated without significant weight penalties, potentially generating robust flexible designs.

1 Introduction

Aerospace manufacturers are increasingly using laminated composites to replace metallic materials in primary structures with the objective of reducing aircraft weight and maintenance requirements [1]. The stiffened panel design of wing and fuselage structures seeks to take advantage of the high strength-to-weight, high stiffness-to-weight and design flexibility characteristics of composite materials. However, the manufacture of stiffened panels introduces additional design complexity and manufacturing geometric constraints over traditional metallic structures [2-4]. For example, co-cured manufacture requires minimum flange widths to allow pressure application, or maximizing ply area places constraints on laminate stacking sequences. The presented study assesses the influence of composite manufacturing constraints, both geometric and ply based, on the structural performance and mass of aircraft upper wing panels.

2 Design Problem

Typically, an aircraft structure such as a wing is idealized as a series of smaller panel units comprising a single stiffener and skin bay bounded by lateral ribs (Fig 1). The sizing of these panels must consider general panel design constraints related to multi-axial loading, material, strength, stability and durability, generic to either metallic or composite manufacture. The sizing of composite panels must additionally consider constraints that represent the industry practice of reducing manufacturing layup complexity and cost. To this end a series of optimization studies are carried out subject to the standard panel design constraints plus varying levels of composite manufacture and repair constraints applied. These composite specific constraints are in the form of either cross sectional geometric manufacture, assembly and repair requirements or ply continuity requirements which aim to maximize individual ply area.

3 Design Constraints

3.1 Standard Panel Sizing (Constraints Set I)

General aerospace stiffened panel sizing processes are subject to a recurring set of constraints considering the applied loads, the material utilized, strength and stability requirements and general structural best practices. These constraints are enforced on all studies presented.

Loading

The focus of the study is on compression critical upper wing panels and as such the loading distribution is aimed at being representative of various wing panel sections of a mid-size passenger aircraft. Axial compressive loads are the primary reference with additional loads a function of the compressive load;

- Axial compression $(N_x)_C$ loading intensities range from 1000N/mm – 4000N/mm.
- Axial tension loading intensity equal to axial compression, $(N_x)_T = (N_x)_c$
- Lateral compression $(N_y)_C$ and tension $(N_y)_T$ loads typically negligible across large portions of wing and taken as zero, $(N_y)_C = (N_y)_T = 0$
- Shear loading (N_{xy}) approximately 10% of axial compression loads, $(N_{xy}) = 0.1*(N_x)_C$

- Skin buckling is not allowed to occur at less than 87% maximum axial compression load, equivalent to 130% of limit load.

Material

Typically, the industrial design process would use correlated test data to represent laminate level stiffness and material properties as functions of laminate thickness and ply ratios. In the absence of experimental test data, this study presents laminate stiffnesses and material failure allowables developed using theoretical methods. Using non-proprietary lamina material properties[5], Classical Laminate Theory (CLT)[5][6] is used to generate laminate stiffness matrices and the Tsai-Hill fully interactive failure criterion to generate representative material failure allowables [6][7].

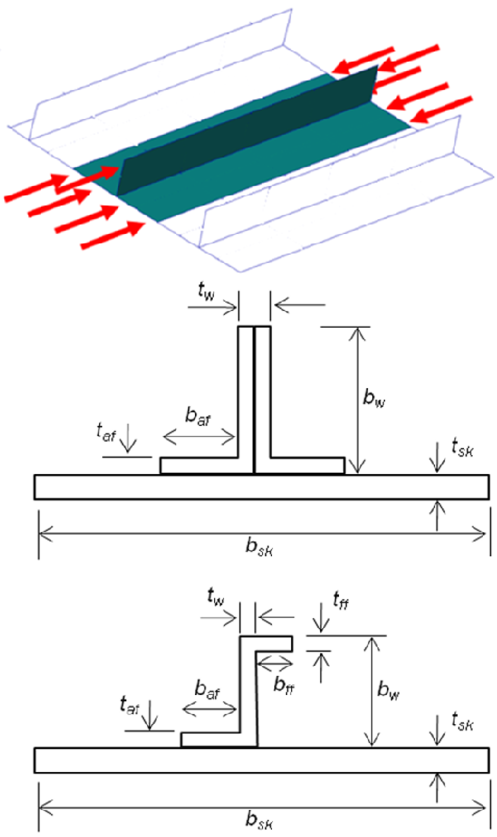

Fig. 1. Single skin-stiffener panel unit within a larger structure, an associated cross sectional dimensions for "inverted T" and "Zed" stiffener idealizations.

Strength and Stability

Aerospace stiffened panel structures generally comprise an external plate skin, divided and supported longitudinally and laterally by internal stiffeners. Given the thin-walled nature of stiffened panel structures, the ability to predict the local buckling, post-buckling and failure behavior of stiffened panel designs is essential.

Conventional aerospace structural sizing procedures evaluate and interrogate the various strength and instability modes of the panel elements to predict panel static behavior and performance at a laminate level. The procedure used within this study replicates the aerospace industrial methods, performing the following analysis checks;

- Static strength of skin and stiffener sections under compression and tension loading [8].
- Static strength of skin sections under shear and combined tension and shear loading [8][9].
- Uniaxial and biaxial compressive skin buckling, skin shear buckling and combined compression and shear buckling [9][10].
- Stiffener cross sectional buckling and crippling [10][11][12].
- Stiffener compressive Euler buckling [8][9] and combined flexure and local crippling using both Secant [8][9] and Johnson-Euler [9] methods.
- Combined stiffener axial compression and lateral pressure using beam-column analysis methods [2][12].

Panel structural performance is measured in terms of a Reserve Factor, R.F. (1). For a panel design to be deemed satisfactory R.F. ≥ 1 is required for all strength and instability modes.

$$R.F. = \left(\frac{Allowable\ Load}{Applied\ Load}\right) \qquad (1)$$

Best Practice Guidelines

Panel cross section design, Fig. 1, is also subject to general structural best practice guidelines [9]. Such guidelines include;

Damage tolerance preference: $0.3 \leq \frac{E_{st}A_{st}}{E_{sk}A_{sk}} \leq 1$

Flange crippling prevention: $3 \leq \frac{b_{af}}{t_{af}} \leq 15$

Web buckling prevention: $0 \leq \frac{b_w}{t_w} \leq 22$

Skin-Stiffener attachment: $0.7 \leq \frac{t_{af}}{t_w} \leq 2$

3.2 Ply Continuity (Constraints Set II)

The integration of multiple individually sized panel units across a larger global structure necessitates the inclusion of skin ply continuity constraints to reduce manufacturing costs. The two key constraints include;

A) Maximizing global ply area by maximizing the number of plies that can be continuously laid across multiple zones. This is enforced by ensuring that each stacking sequence has a 70-80% level of continuity when measured against any other laminate. I.e. for any two possible adjacent laminates, at least 70-80% of plies in the thinner laminate can be continuously laid into the thicker laminate [2].

B) Removing "bow-tie" effects that can introduce potential weak points at panel interfaces. As Fig. 2 demonstrates, it is possible to have plies dropping off in both directions across two laminates with the outcome being an interface region which as fewer plies than either adjacent laminate. This is enforced by ensuring that plies only drop-off in one direction across any combination of two stacking sequences.

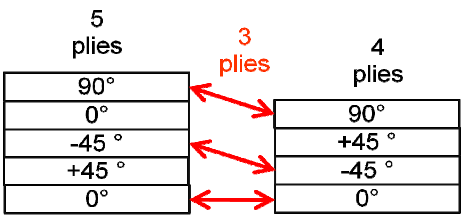

Fig. 2. Example of ply continuity between a 4-ply and 5-ply laminate.

With regards to the selection or optimization of the laminate stacking sequences, the tabular method using pre-defined databases of stacking sequences can provide more control over ply continuity. It has been demonstrated that designs generated using this approach can produce designs of equivalent mass, yet significantly higher levels of ply continuity than designs generated using Genetic Algorithms and bi-level optimization methods [2][13][14]. In addition, the tabular method can be considered an industrial compromise between a complex design procedure and reducing manufacturing cost.

To measure the effect of skin ply continuity constraints two skin laminate databases have been developed for panel sizing exercises;

- Skin database A contains all possible stacking sequences up to 24mm thickness that obey general stacking sequence rules, but with no continuity constraints applied. A total of 127 laminates comprise the database.
- Skin database B has been filtered to enforce the two ply continuity constraints (A and B), reducing the number of laminates in the database to 34.

The unique stiffener database has general stacking sequence rules and ply continuity constraints enforced, comprising 13 laminates up to 8.8mm thickness.

3.3 Geometric Manufacture & Repair (Constraints Set III)

As shown in Fig. 1 the presented study focuses on "Inverted T" and "Zed" section stiffeners on a uniform thickness skin bay. The rib pitch is fixed at 500mm. The manufacturing concept of the Inverted T section involves co-curing two back to back "L" pre-formed parts without an insert between, therefore enforcing the constraint $t_w = 2t_{af}$, with t_{af} being the attached flange laminate as defined in the stiffener database. The corresponding constraint $t_w = t_{ff} = t_{af}$ is enforced on the "Zed" section panel concept. A co-curing process requires a minimum width of attached flange to adequately apply the necessary pressure, with practical recommendations suggesting $b_{af} \geq 30$mm as a constraint.

Repair requirements place additional geometric constraints on cross sectional geometry. These constraints arise from the need to accommodate metallic repair features by traditional fastening methods. Typically, the thickness of a feature under repair dictates the size of bolt required, which in turn drives the minimum width of the feature needed to safely accommodate the bolt. Considering the critical case of a broken stiffener:

- Inverted T sections use repair angles bolted through web and flange, providing the constraints;

$$\left(\frac{b_w}{t_w}\right) \geq 3.3, \quad \left(\frac{b_{af}}{t_w}\right) \geq 2.8$$

- Zed sections use repair straps attached to the web only, providing the constraint;

$$\left(\frac{b_w}{t_w}\right) \geq 5$$

4 Analysis

4.1 Panel Sizing Tool

Stiffened panel structure can be idealized as a series of plate (skin) and column (stiffener plus effective width of skin) elements, and analyzed using industry standard empirical and semi-empirical methods. The buckling analysis methods typically employ closed formed solutions that assume the laminates are specially orthotropic, balanced and symmetric. In addition, all plate and column instability analyses assume simply supported boundary conditions at structural interfaces.

The panel design process employed in this study uses a sizing tool housed in Microsoft Excel with analysis calculations carried out through Visual Basic (VBA) code. The sizing operation of the tool utilizes full factorial design principles, employing response surface methodologies without regression that calculates the performance of every potential design within a user defined design space. The selected design is that of minimum volume that satisfies all loading, laminate stacking sequence and geometric constraint requirements. For the purpose of further in depth study of results, all designs analyzed are stored for further interrogation.

4.2 Laminate Databases

The laminate databases are pre-defined based on general sequencing rules, target ply ratio distribution and the target application, and then integrated into the panel level sizing process. Target ply percentage distribution for 0°, ±45° and 90° material in the skin laminates are 35%, 50% and 15% respectively, and 50%, 40% and 10% in the stiffener laminates respectively (with a 5% tolerance). The developed stacking sequences are also subject to general stacking sequence guidelines [2][6], which include for example;

- Each laminate must contain at least 10% thickness of each orientation.
- No more than 3 plies can be laid consecutively together.

In addition, stacking sequences also accommodate a separate dedicated ply on both outer surfaces. This ply is typically a robust satin fabric designed to maintain surface integrity during the assembly process.

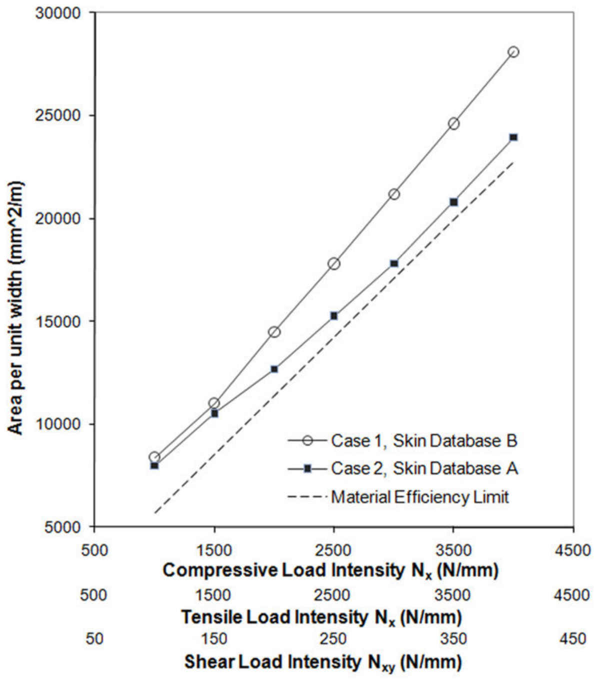

Fig. 3. Cross sectional area comparison of "Inverted T" stiffened panels for a range of loading intensities with two different skin laminate databases.

5 Results & Discussion

5.1 Influence of Ply Continuity Constraints

Fig. 3 presents minimum mass designs (measured as cross sectional area per unit width) of "Inverted T" stiffened panels, for a range of loads, using both the ply continuity constrained (Case 1) and unconstrained (Case 2) skin laminate databases. In both cases Constraint sets I and III are also enforced. It is clear that when continuity constraints are not considered lighter designs are achievable than when continuity constraints are enforced. At higher loads (>2000N/mm) the designs without constraint set II applied are consistently 13-14% lighter than the equivalent designs with constraint set II applied. This highlights the significance of ply continuity considerations when designing a compromise between structural performance and manufacturing complexity and cost. At lower loads (<2000) ply continuity constraints have less impact on minimum mass designs, with any mass savings typically less than 4%. The reduced influence at lower loads may be a result of the limited number of potential stacking sequences available for low skin thicknesses after applying the standard sequencing rules. Up to a skin thickness of approximately 5mm the laminate databases with and without ply continuity constraints are almost identical.

Interestingly, when comparing load equivalent continuity constrained and unconstrained designs the skin thickness is often identical in both panels. In most cases the heavier continuity constrained design attempts to balance an off-optimal stacking sequence not by increasing skin thickness to the next laminate, but by adding material to the stiffener section. Skin thickness is a large contributor to panel mass and it only becomes efficient to step up to a thicker skin laminate when tailoring the stiffener cross section fails to meet the structural performance required.

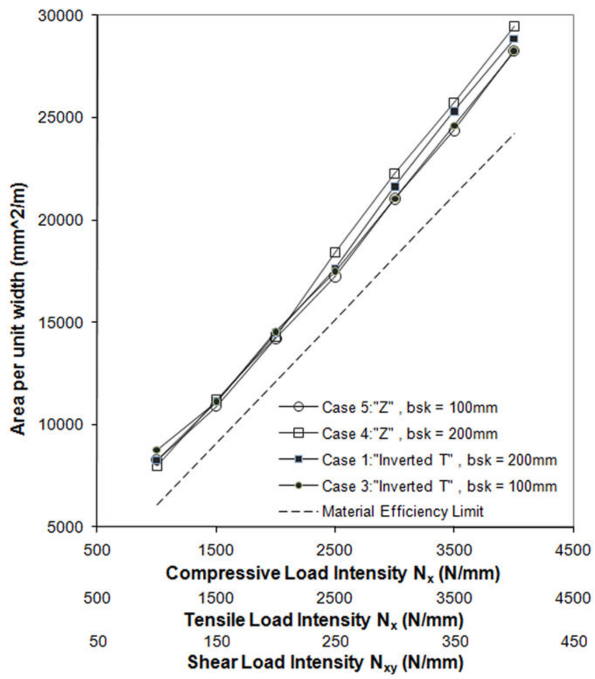

Fig. 4. Cross sectional area comparison of “Z” and “Inverted T” stiffened panels for a range of loading intensities.

5.2 Stiffener Idealization

Fig. 4 presents minimum mass designs for “Z” and “Inverted T” stiffened panels at stiffener pitches of 100mm and 200mm. All geometric and continuity constraints (Constraint sets I, II and III) have been applied. At higher loads the inverted T designs are marginally lighter, 1.6% at b_{sk}=200mm and 0.6% at b_{sk} = 100mm, with the “Z” stiffened panels demonstrating more sensitivity to stiffener pitch. At lower loads minimum mass designs are relatively insensitive to stiffener pitch and idealization. Inspection of the entire design space studied showed that of all potential designs attainable in the global design envelope, “inverted T” panels had up to 33% more designs satisfying the various constraints than “Zed” panels. The more acceptable design options associated with the “inverted T” panels may have contributed to the lighter designs over “Z” panels at higher loads.

5.3 Influence of Geometric Manufacture & Repair Constraints

Fig. 5 presents minimum mass designs for “Inverted T” panels both with (Case 1), and without (Case 6), geometric manufacture and repair constraints applied. In both cases Constraint Sets I and II are enforced. At higher loads (>2500N/mm), as expected, the addition of various geometric constraints results in a weight penalty, though relatively small at 1.2-1.6%. Interestingly, at lower loads the minimum mass designs are relatively insensitive to the addition of the geometric constraints. Designs with and without constraint set III enforced are mass equivalent.

Considering the various geometric constraints, in particular the dimensional ratios, every cross sectional design variable is linked via these rules. This may potentially drive the design process into a narrow region of the design space and thus limited the design flexibility. However, while only a fraction of the entire design space satisfied the various constraints there is little variation in optimal mass, suggesting that the most mass efficient designs tend to lie within the geometrically constrained design space.

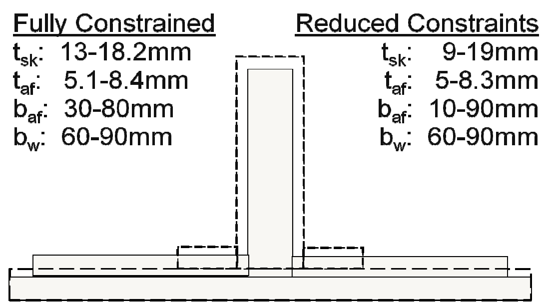

Fig. 6. Demonstrating design variation for all designs within 2% of optimum Inverted T panel at $(N_x)_C$ = 3500N/mm.

5.4 Geometric Design Flexibility

Inspection of the actual cross section topology of the optimal and near optimal designs also highlights interesting levels of geometric variation. From inspecting all designs with mass lying within 2% of the optimal design, Fig. 6 presents an example of the design variation available for a N_x = 3500N/mm "inverted T" panel. Also outlined are the ranges of design variable dimensions encountered within the 2% mass range. It demonstrates high levels of variance across multiple designs that are almost mass neutral, and this is evident for both a reduced and fully constrained design space. It suggests a level of robustness with composite panel design which offers the sizing process a degree of flexibility to generate multiple designs of varying topology without suffering a weight penalty.

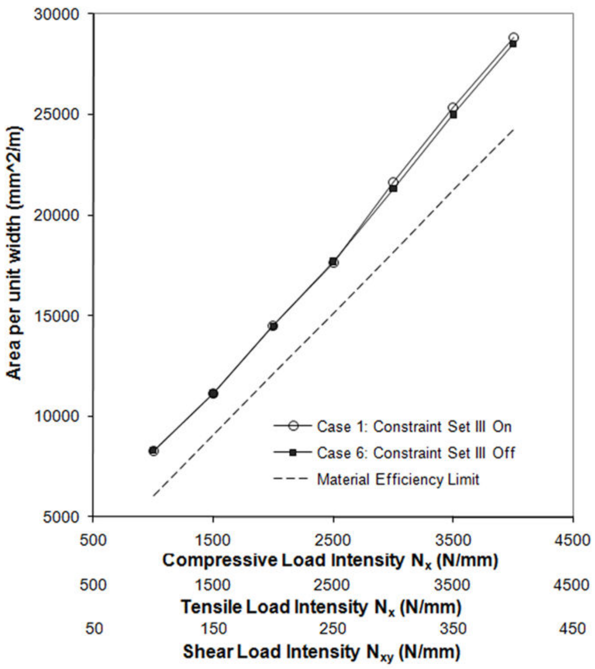

Fig. 5. Cross sectional area comparison of "Inverted T" stiffened panels for a range of loading intensities with different levels of constraints.

6 Conclusions

The results of the design studies have established, as expected, that introducing manufacturing geometric constraints and enforcing ply continuity across a stiffened panel structure can significantly drive the panel cross sectional / dimensional topology. Interestingly, it emerged that weight penalties occurred in highly loaded wing zones which were tightly constrained by manufacturing. However, in medium to low loaded zones the design freedom offered by composite materials allowed more robust designs which accommodated the manufacturing constraints without weight penalties.

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