

Process Change: Redesign of Composite Parts for Structural Integrity

Jean-Philippe Lachance and Jean-Evrard Brunel
Bombardier Aerospace

ABSTRACT

The objective of this document is to present the methodology used to verify the structural integrity of a redesigned composite part. While shifting the manufacturing process of a composite part from pre-impregnated to a liquid resin injection process, the Composites Development team at Bombardier Aerospace had to redesign the component to a new set of design allowables. The Integrated Product Development Team (IPDT) was able to quickly provide a turnkey solution that assessed three aspects of airframe engineering: Design, Materials & Processes (M&P) and Stress. The focus of this paper will be the stress substantiation process led by the Stress Engineers. It will also bring up the synergies with M&P that are unique to the IPDT approach.

The stress substantiation process required three distinct checks be confirmed. These were that the global behavior of the surrounding structure remained unaffected, that all local failure modes still exhibited positive margins of safety, and finally to organize detail level testing to provide complete substantiation for the certification authorities.

The paper also exposes how this method was used successfully on components of programs currently under development at Bombardier Aerospace. The IPDT approach as well as the novel mean of substantiation allowed for a quick turnaround. This resulted in the rapid preparation of a positive business case for both weight and cost.

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INTRODUCTION

It is a well-documented fact that, in the case of composite materials, material properties are vastly dependent on the manufacturing process. This is initially true because of the characteristics the materials must possess for the target process to function. Requirements for resin rheology are very different if it is to be used as in a pre-preg or a Vacuum Assisted Resin Infusion (VARI) process. This in turn yields varying mechanical properties of the cured laminate. Furthermore, with a given process, parameters and robustness are key in determining the failure load of components once assembled on an aircraft. Laminates with the same degree of cure, but subjected to different cure cycles can exhibit different cross-linking patterns which can adversely affect the properties of the material. Consequently, converting components from a process to another can be a very arduous task for both Manufacturing & Stress Engineers. This is most efficiently done through an IPDT approach as documented in Pahud, O. [5].

Concurrently to the development of a new liquid resin manufacturing process, an opportunity arose to convert Carbon Fiber Reinforced Polymer (CFRP) components from a pre-preg to a Resin Transfer Molding (RTM) process. Typically, non-recurring costs associated with high performance RTM manufacturing preclude small aeronautical production volumes. However, repeating part numbers created a business case where the non-recurring costs were amortized over enough components that the RTM process yielded decreased cost in addition to a shorter cycle time. As previously stated, the shift in process was accompanied by a shift of material allowable. While process robustness and qualification was tackled by Manufacturing Engineers as documented in Salek, H. [6], there was still a need to size the part to a complete new set of design allowables. These included a drastic variation in lamina Cured Ply Thickness (CPT), as well as notable variations in matrix type failures such as Compression After Impact (CAI) and InterLaminar Shear and Tensile Stress (ILSS/ILTS). As a starting point for this exercise, both the material and process were qualified and allowables were readily available.

Therefore, the redesign necessary to account for the changes in material properties concentrated on two aspects to verify substantiation. The requirements were that, following the redesign:

- The global behavior of the surrounding structure remained unaffected
- All local failure modes were still met.

Comprehensive substantiation still requires a pyramid block approach test plan be completed. Material and process must be deemed equivalent to current practices using coupon and element level testing. Furthermore, allowables derived from detail testing must be confirmed. However, in the context of aircraft substantiation by analysis supported by test evidence, this represents a very succinct and low cost means of compliance compared to full qualification and developments of a full set of allowables.

STRESS SUBSTANTIATION PROCESS

Global Considerations

As previously stated, the first consideration whilst redesigning a component is to ensure the redesigned structure behaves identically as the baseline. In spite of the many advancements in mechanics, materials, analysis and manufacturing, aircraft structures are still as simple as their pre-war ancestors. Fuselage structures are simply thin skin sheets supported by stiffening members. These members serve two major roles. Firstly, they provide some area to the fuselage cross-section to carry loads. This is presented extensively in Chapter 11 of Niu, M. [3] where it is shown on a typical aircraft fuselage the longitudinal stiffening members, or stringers, can represent up to 50% of the cross-sectional area. However, the second and most important function of the stiffening members is to provide support to the skin and prevent it from buckling. This is phenomena is visible in [Figure 1](#), where the buckling waves are constrained to small bays by the orbital frames and longitudinal stringers. For thin sheets subjected to shear or compressive loading, the loads at which the plates buckle are much lower than the material strength. The stiffening members address this two ways as viewed in [Equation 1](#) and [Figure 2](#).

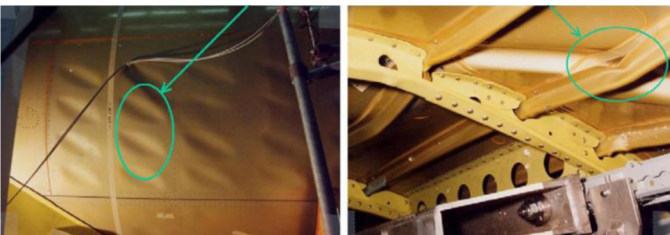


Figure 1. Containment of Buckling Waves to Bays by Stiffening Members

$$\sigma_{cr} = \frac{K\pi^2 E}{12(1-\nu^2)} \frac{t^2}{b^2}$$

Equation 1: Euler Buckling Stress. Bruhn [1].

The previous [Equation 1](#) corresponds to critical Euler buckling stress. In addition to the skin thickness t , the buckling stress is given by the dimension of the panel b as well as the level of support K given by [Figure 2](#). The stiffening members serve as panel breakers and their span therefore dictates the panel buckling load. While this is an essential consideration during preliminary design phases, these were far beyond the scope of redesigning single parts. However, it was still necessary for the stiffeners to achieve the required level of support for the skin.

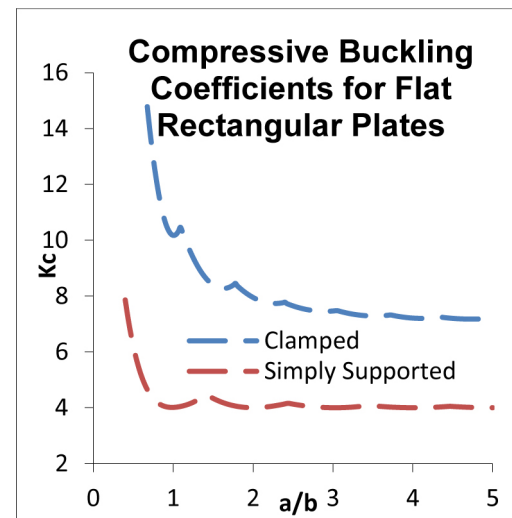


Figure 2. Buckling Coefficients of Flat Rectangular Plates under Compressive Load. Bruhn [1].

The stiffening members being fastened to the skin and possessing some torsional rigidity, they provide to the skin boundary conditions somewhere between simply supported and clamped as illustrated in [Figure 2](#). When redesigning the component, it is mandatory that this does not decrease. The level of support is given mostly by the out-of-plane bending stiffness and torsional rigidity. Consequently, analytical beam theory can be used to compare the rigidity of iterations of redesign and quickly determine candidate laminates.

While it is a good practice to target the baseline design's stiffness for longitudinal axis and out-of-plane bending, this is not necessary. This is due in part by the fact the rigidity of the fuselage is only given in fraction by the stringers, who represent at most 50% of the total area. Furthermore, as illustrated in [Figure 3](#), the stiffness of the fuselage is given most importantly by the total area of skin & stringer of the fuselage and its distance from the centroid. Large variations in Area-Moment-Of-Inertia (AMOI) relative to the stringer are negligible to the dimensions of the fuselage and do not contribute to a redistribution of loads with the surrounding structure. This opens up the design space significantly, and