Damage arrest design approach using stitched composites

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ABSTRACT

Although the deployment of carbon fibre structural components has enjoyed wide success on smaller aircraft, their acceptance on larger transport airframes is less sanguine – especially in the case of primary structure applications where the increasing out of plane loads found on the larger airframes have exposed the weak inter-laminar properties of the layered material system. This has led to an overreliance on mechanical attachments to suppress these through thickness failures, which ultimately degrades structural performance and increases manufacturing costs. Until these resin dominated failure modes can be dealt with more effectively, without adversely affecting the in plane properties of the laminate, the true weight-savings potential afforded by carbon fibre material systems will be difficult to attain. This paper describes how researchers at NASA-LaRC and The Boeing Company are working to develop a next generation stitched composite design solution that addresses the fundamental challenges in achieving improved structural performance and reduced fabrication costs for large carbon fibre airframe structures.

NOMENCLATURE

BVID	Barely Visible Impact Damage
CAPRI	Controlled Atmospheric Pressure Resin Infusion
DLL	Design Limit Load = maximum load encountered
DUL	Design Ultimate Load = 1·5 times maximum load encountered
PRSEUS	Pultruded Rod Stitched Efficient Unitised Structure

1.0 INTRODUCTION

The modern transport monocoque airframe design is the result of a century of materials development, design evolution, and in-service experience. The basic stiffened panel shell that comprises the airframe is an extremely complex array of skins, tear straps, stiffeners, frames, and fastening systems that are designed to react static and fatigue loadings while also maintaining residual strength under a wide variety of impact damage scenarios. Over the course of this evolution, the complex interaction between material properties, design criteria, and structural concept characteristics have progressed collectively into a highly integrated structural solution that results in the lightest, most reliable, and safest airframe that is economically possible. For today's state-of-the-art airframes, this evolutionary process started in the mid 1930s with the advent of the aluminium skin-stringer panel and was further refined over the years using specially tailored alloys and sophisticated fracture mechanics techniques that have resulted in unprecedented levels of damage tolerance, durability, and safety.

Though common for metallic structures, such levels of airframe design precision have yet to be realised for composite primary structures. Even though carbon fibre material systems possess superior mechanical properties, along with a higher degree of tailorability than metals, the full potential of these attributes have yet to be realised due to the weak interlaminar properties of the layered material system. In many cases, the resulting design strains are still too low to be weight competitive with existing metallic designs, and when paired with uninspired bolted together structural assembly approaches, the resulting 'black aluminium' designs are often a further departure from the aspirations of the composite design engineer.

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To get beyond these criticisms, researchers at The Boeing Company in Huntington Beach and at NASA Langley Research Centre are developing a next generation composite design solution that utilises through-thickness stitching to suppress interlaminar failures so that higher levels of fibre tailoring are possible, as well as damage arrest boundaries that will enable fail safe design approaches. This new design and manufacturing approach is called the Pultruded Rod Stitched Efficient Unitised Structure (PRSEUS) concept (Ref. 1). Beyond the integral construction and novel fabrication aspects of the design, the PRSEUS concept is the first composite structure that utilises through thickness reinforcements, stitching in this case, to facilitate the use of fail safe design methodologies in a manner similar to metallic structural design practices. Development of this new damage tolerance design space for a composite structure is described in the following sections of this paper.

2.0 CONVENTIONAL COMPOSITE DESIGN PRACTICES

The basic design requirements for a transport aircraft are set by FAR Part 25, Airworthiness Standards: Transport Category Aircraft (Ref. 2). These minimum design and compliance objectives, along with the manufacturer's past experience, forms the basis on which new airplane design criteria are developed. Through a combination of analysis and testing, strength and deformation requirements are validated over a wide range of critical design conditions. How these unique design requirements are satisfied is strongly influenced by the characteristics of the individual material systems chosen, and nowhere is the composites vs metals contrast greater than in the design methodology used to meet FAR Part 25.571 (Damage-tolerance and fatigue evaluation of structure). Here the layered arrangement of the composite material system forces the design engineer to utilise the safe-life design objectives as described in Paragraph (c) [Fatigue (safe-life) evaluation] rather than the fail safe approach in Paragraph (b) [damage tolerance evaluation] that is typically used for metallic structures.

Arguably, this single difference constitutes the largest weight penalty for composite materials in primary structure applications like the wing or fuselage. This is especially true for larger transport size aircraft because the weak resin interfaces are not scalable; i.e. making thicker parts does not increase the interlaminar resin strength, which must react the higher out of plane loads inherent in the larger airplane. This is the fundamental problem facing the structural designer and it cannot be overcome unless composite structures can be made more damage tolerant like metal structures; whereby they too would meet the crack growth and damage arrest methodology described in Paragraph (b). This forms the basis for the PRSEUS design approach; it uses through-thickness stitching to increase the out of plane tension strength of the layered material system so that the damage tolerant requirements described in Paragraph (b) can be met. Moving beyond the safe life, or no growth, design constraints of Paragraph (c) is the single most important factor necessitating the use of through thickness stitching for composite primary structures, and yet it remains one the least understood incentives for doing so.

3.0 USING STITCHING TO ACHIEVE DAMAGE TOLERANCE

The overarching goal of this new design approach is to create a composite structural assembly that has an equivalent level of safety and crashworthiness to that of a conventional state-of-the-art aluminium structure: multi load path, redundant, damage arresting, and capable of redistributing internal loads to the undamaged regions of the panel assembly. To achieve this with composite materials a new level of damage tolerance design must be defined – one where the relationship between damage initiation and growth is more precisely defined to characterise a tightly choreographed failure progression that not only encompasses material failure but also the inherent structural capabilities of the entire panel assembly. Such relationships are commonplace for metallic structures where the evolution of damage tolerance design practices has influenced everything from the alloy compositions, to the inclusion of tear straps, and even down to the individual stringer and frame spacing. Such a comprehensive level of design sophistication should also be employed for next generation composite structures because it brings about the substantial weight reduction that comes from moving beyond conventional no growth design practices.

As one might expect, the logic for such an approach comes directly from the existing requirements established in FAR Part 25, and closely parallels the methodology already used for metallic structures. It is based on two fundamental principles: (1) the structural response up to

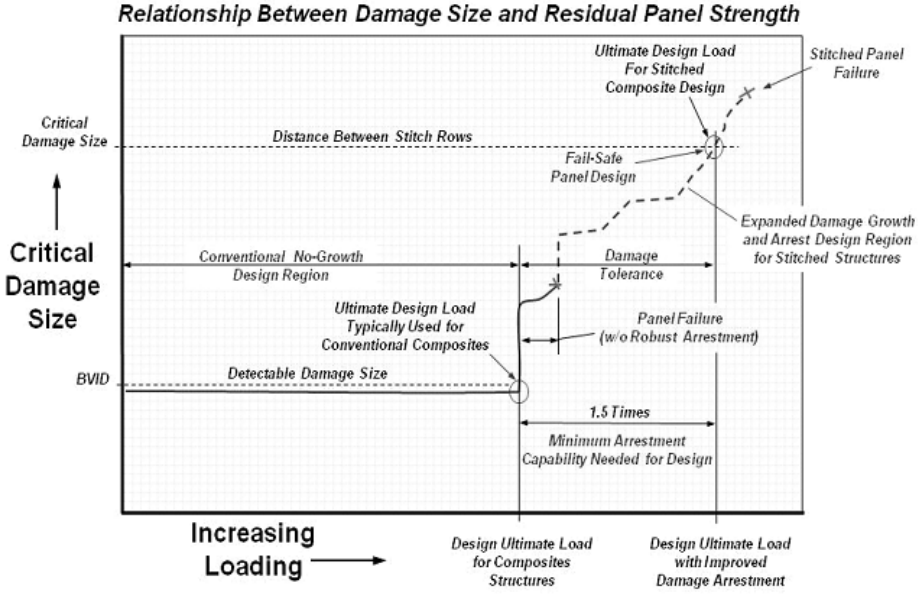


Figure 1. Notional improvements possible with robust damage arrestment.

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limit loading must remain linear and, (2) damage must be arrested between limit and ultimate loading to maintain the residual strength of the panel assembly up to ultimate load. For composite materials, the result of enforcing linearity up to limit load essentially means that BVID levels of damage are not permitted to grow until limit load is reached. Above this point, damage propagation would be permitted as long as it is arrested and the internal panel loads are redirected to the undamaged regions of the structure so that the residual strength of the overall panel assembly is maintained up to ultimate load. Such an approach is graphically depicted in Fig. 1, where the dotted region of the curve represents the theoretical improvement that is possible by designing within the damage tolerance design regime.

The figure shows that by permitting Barely Visible Impact Damage (BVID) sites to propagate at limit rather than ultimate load, and by arresting damage growth between limit and ultimate loading at the stitched interfaces, higher load levels could be possible. Although this method of load redistribution is common for metallic structures, it has yet to be used for composite structures because stopping damage propagation and preventing out of plane failures is difficult in laminated material forms. To avoid spreading uncontained delaminations, conventional composite structures are typically designed to lower operating strain levels so that BVID levels are not permitted to grow until ultimate loading is reached. This compromise leads to much thicker skin gauges than would otherwise be necessary if damage were permitted to grow at limit load, and could then be arrested by adjacent structural members on the way up to ultimate loading. The final outcome of the conventional no growth design approach is a heavier structure that tends to be governed by resin dominated failure modes that occur at the onset of damage growth and propagate rapidly at higher sustained static load levels. And yet even with the substantial improvements afforded by toughened resin systems, still higher levels of through thickness strength and toughness are needed to span the entire damage arrestment design regime between limit and ultimate loading as proposed in Fig. 1.

4.0 STRENGTH AND DEFORMATION REQUIREMENTS

Damage arrestment is the single most important breakthrough needed to realise the true load-carrying potential of carbon fibre based material systems. Reliance on the conventional no-growth

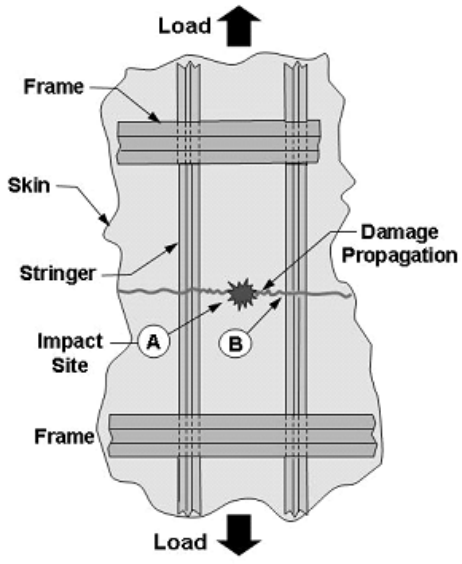


Figure 2. Conventional failure sequence.

design practices, typically employed for brittle composite designs, reduces their effectiveness compared to ductile material systems like metals, which are designed using more favourable fail-safe design criterion. This single difference in design methodology, driven primarily by the brittle-versus-ductile material response, has a substantial effect on structural weight. For composite material systems to be more competitive in primary structure applications, they must be capable of moving beyond the limitations imposed by no growth design philosophies. Recognising this principle forms the basis of why stitching is so relevant in developing improved next generation composite designs; it provides a means of arresting damage and redistributing internal loads, thus making brittle material fractures appear to more closely resemble those of ductile material failures. This in turn, enables the designer to employ those more efficient fail-safe design practices typically reserved for metals, where well-proven multi-load path design schemes are used to control damage growth and redistribute loads to undamaged regions of the structure to maintain the overall structural integrity of the combined material-structural arrangement.

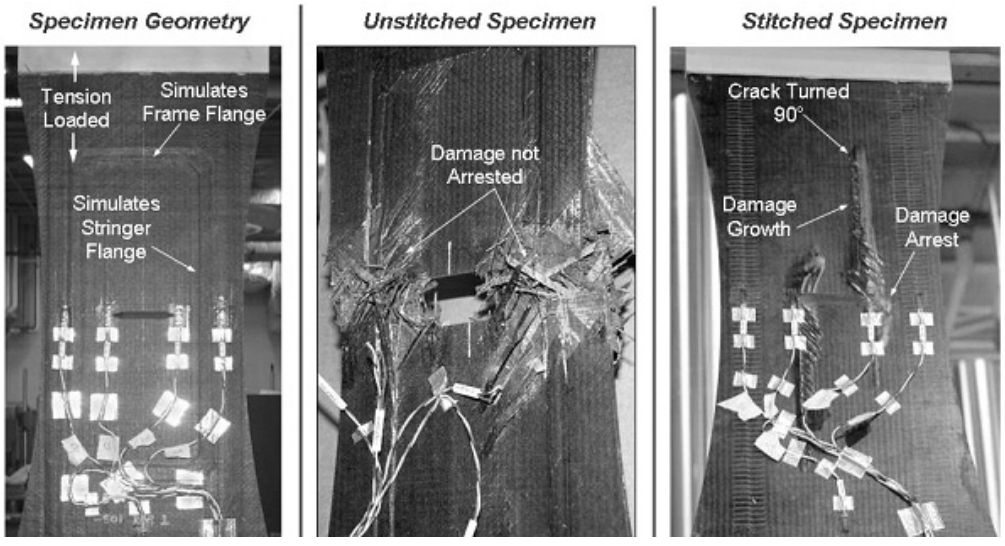


Figure 3. Comparison of unstitched and stitched laminate failures.

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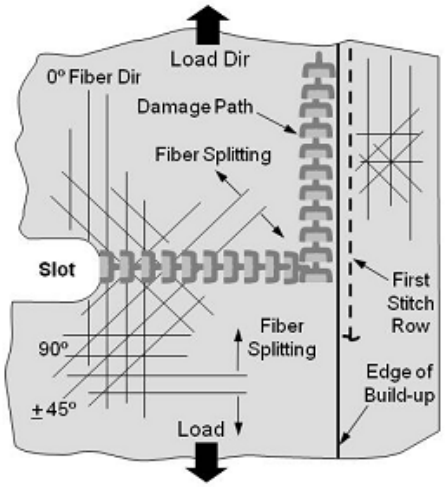


Figure 4. Fibre failure sequence.

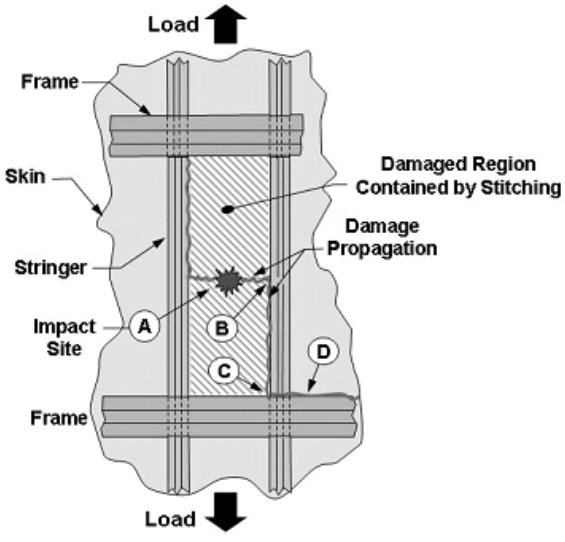


Figure 5. Stitched panel failure progression.

The key aspect of this new damage-arrest design approach that is missing from conventional composite design solutions is the ability to retard damage growth by either blunting or redirecting cracking by reducing the crack-tip stress intensity. Without this, increasing stresses at the crack tip causes the damage to grow unabated. The failure sequence for a conventional composite tension-loaded panel with induced damage is illustrated in Fig. 2. Here, as the static load is increased, damage grows from the induced damage site (Point A) and propagates horisontally across the net section of the panel in an uncontained manner, crossing the adjacent stringer at Point B – simultaneously breaking 0° fibres in the skin and failing the weak resin interfaces along the way as it skirts beneath the stringer. Load sharing between the fractured skin and intact stringer member are not as prevalent as in ductile structures because the local deformations and load sharing that normally occur are absent. Even with the most effective adhesives binding the details together, the resulting first ply failures in the parent materials are still dominated by brittle resin failures – an inherent characteristic of the unreinforced layered material system. This inability to unload the crack tip by redistributing loading is the essential characteristic that shackles conventional composite structures to the use of the inefficient, no-growth design schemes typically reserved for only the least ductile material forms.

Stitched composite concepts on the other hand, while essentially still composed of the same brittle resin and fibre material constituents, are able to replicate the fundamental crack-turning and load-sharing phenomena ubiquitous in metallic structures due to the load path provided by the stitching thread. This capability was demonstrated using a series of dog bone tension specimens with built-up edges (to simulate stringer and frame flanges on the skin) and a centre slot to start damage progression (Fig. 3). As expected, the unstitched specimens (centre photograph) consistently failed in a horizontal progression across the net section of the panel without any damage arrest or crack turning behaviours. Next, a series of stitched specimens were tested, and the failure response was markedly different (right photograph). As loading was increased in the stitched specimens, damage initially emanated from the centreline slot where it was first arrested horizontally at the stitched skin buildup. The crack then turned vertically, as it split the 0° dominated fibres, before being arrested again at the horizontal stitched skin buildups near the top and bottom of the test panel.

Once the damage was stopped in the opposite corners of the skin bay, increasing load levels caused the skin to tear horizontally across the upper corner of the built up skin. This unique damage

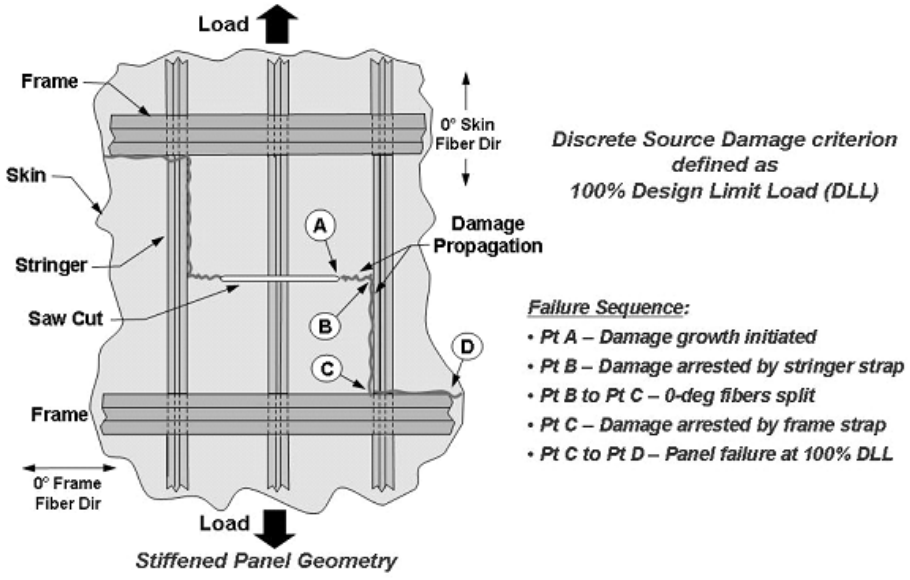


Figure 6. Stitched panel discrete source damage failure mode.

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progression was readily observed as an indentation progressively forming on the outer moldline surface of the stitched specimens – pictured just before the failure (right photograph in Fig. 3) – as it traveled along the simulated stringer flange buildup before finally tearing horizontally along the simulated frame flange buildup near the top of the panel. This crack-arrest and turning phenomenon, which only occurred in the stitched specimens, represents a fundamental shift in the progressive failure pattern typically observed in composite laminates. Since the through thickness stitching was the only difference between the two specimen configurations, the ability to initially arrest and ultimately change the direction of the oncoming cracking can only be attributed to the stitched interfaces.

Embedded within the layered arrangement of the warp-knit fabric, there seemed to be a sequential order of fibre failures occurring as the static loading was increased. As each successive 0-degree fibre tow (starting at the slot edge) was overloaded to failure (Fig. 4), the damage progressed horizontally toward the stitched buildups at the panel edges. Sequentially, as the 0° fibres failed, the resin dominated or fibre splitting failure modes began to occur across the 45° and 90° fibre layers which were visible as indentations in the OML surface (up to the edge of the stringer flange buildups).

As damage approached the edge buildups, the influence of the first stitch row enabled better load sharing between the 0-degree fibres in the skin and the buildup region of the panel, such that the ultimate strength of the adjacent fibres was not exceeded, and thus remained intact to continue carrying load and unload the crack tip. Once this occurred, the damage no longer propagates in the horizontal direction and the crack-turning mode had started – now, the 0° fibre splitting began along the buildups and progresses in the vertical direction parallel to the stitching (Fig. 4) until it was arrested at the edge of the horizontal buildups that also contained stitching.

Having demonstrated the basic attributes of arresting damage using stitched interfaces, the next step was to incorporate this methodology into a next generation composite design approach that would not only offer superior performance, but would also provide better fail-safety and structural redundancy under an extensive array of damage scenarios. Such a methodology would also closely resemble the fundamental design principles established for metallic structures, but it would have to differ slightly to account for the brittle fatigue and fracture mechanics inherent in carbon fibre material system. The key aspects of this new damage-arrest design approach are shown in Fig. 5 and summarised below:

1. Maximum BVID damage level at Point A.
2. No damage growth up to Limit Load with BVID.
3. Growth from Point A-to B after Limit Load.
4. Growth from Point B-to C up to Ultimate Load.
5. Failure from Point C to D above Ultimate Load.

The ability of the stitch rows to contain damage growth within the open skin bays between the principal structural elements (stringer and frame elements) is the essential underpinning of this new approach. This serves an important purpose as it reduces the stress intensity at the crack tip (reducing the likelihood of further progression), while simultaneously maintaining the structural integrity of the stiffening elements to redirect load around the damaged region. This newfound capability also has an important implication for the overall structural optimisation of the panel assembly. It tends to drive the skin gauges down and concentrates the axial load carrying material in the stiffening elements (stringers and frames). The result is a substantially lighter composite panel assembly that more fully exploits the highly orthotropic lay ups used for the stringer and frame elements (in their respective directions) that are stitched onto the softer skin laminate. The ability to mix hard and soft laminates within the same integral panel assembly is now feasible because stitching is used to suppress the out of plane failure modes that normally occur at the edges and transitions of highly tailored composite laminates.

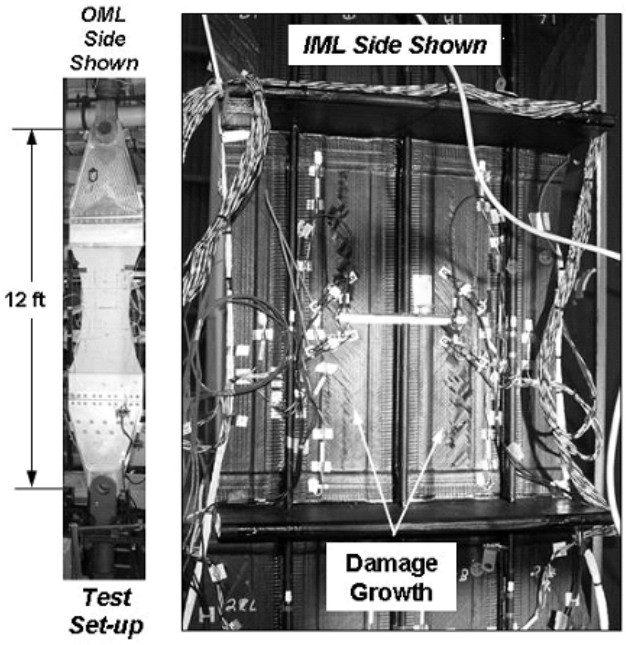


Figure 7. Damage growth at 100% DLL.

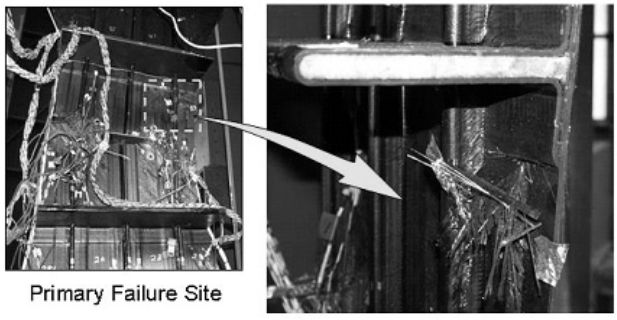


Figure 8. Primary failure site at load at 130% DLL.

The same fundamentals are also used to meet the discrete source damage loading conditions; here too, the basic methodology established for metallic structures is closely followed (Fig. 6). The basic requirement would be set at 100% Design Limit Load (DLL) with a principal structural element removed. The overall failure mode and damage progression sequence is similar to the previous damage growth discussion: the damage arrest and crack turning aspects of the design are used again to maintain overall panel integrity by permitting load redistribution around the damaged two bay region of the panel. The key aspects of this failure sequence are listed in Fig. 6.

The hypothesis for this new design approach was validated by test (Fig. 7, Refs 3 and 4) using a three stringer dog bone tension specimen with a two bay crack (slot across central stringer) that was statically loaded to failure (Fig. 8). The results were consistent with the earlier coupon panel testing (Fig. 3). Like those tests, the stiffened panel specimen was also able to arrest propagating damage in both the horizontal and vertical directions as it emanated from the slot edge. Because damage was arrested at the stitched interfaces and successfully contained within the two bay damage zone bordered by the adjacent stitched stringers and frames, the panel was able to continue bearing load beyond the 100% design limit load (DLL) requirement. The specimen ultimately failed at the frame flange at 130% DLL, and 60% higher than the onset of damage propagation at the slot edge. This test clearly demonstrated that substantially higher loads are possible (60% higher) when stitched interfaces are used to arrest damage and turn cracks, thereby enabling a fail-safe design methodology to be successfully deployed for composite material systems.

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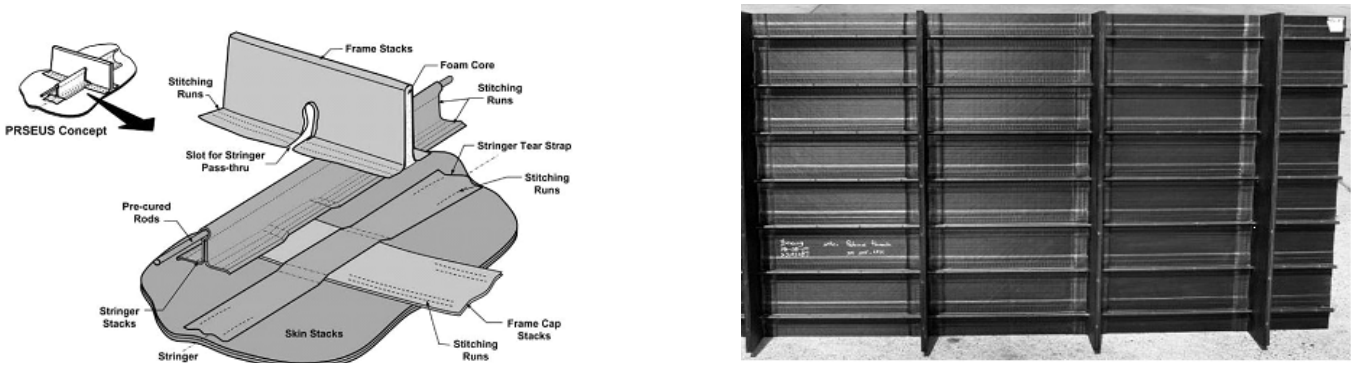


Figure 9. Pultruded rod stitched efficient unitised structure (PRSEUS).

Without the benefits of stitching, one would have expected the damage to continue growing at the 80% DLL load level (point at which crack propagation began during this test) in a manner similar to what occurred in the flat coupon test described earlier, and then fail shortly thereafter across the net section. This would have been 20% below the 100% DLL requirement, and roughly 60% below the actual stitched panel test failure load. By using stitched interfaces to arrest damage and turn cracks, a more favourable internal loading distribution was established that unloaded the crack tip and enabled the undamaged elements of the structure to continue carrying additional load; similar to what happens in a conventional aluminium panel design.

5.0 PRSEUS STRUCTURAL CONCEPT

The PRSEUS structural concept was specifically developed to exploit the superior out of plane strength and structural integration advantages made possible with through thickness stitching. In addition to the damage arrest advantages already discussed in this paper, stitching also enables a higher level of panel integration and laminate tailoring than would otherwise be possible using conventional composite materials. By examining the highly integrated nature of the PRSEUS concept (Fig. 9), it is evident that the strategic placement of hard and soft fabric layers along the principal loading directions could only be accomplished when those interfaces are reinforced with stitching. The dry warp-knit fabric, pre-cured rods, and foam-core materials are assembled and then stitched together to create the optimal structural geometry for pressure and biaxial loading by maintaining load path continuity at the stringer-frame intersection in both directions. The 0° fibre dominated pultruded rod increases local strength/stability of the stringer section (Ref. 5), while simultaneously shifting the neutral axis away from the skin to further enhance the overall panel bending capability. Frame elements are placed directly on the IML skin surface and are designed to take advantage of carbon fibre tailoring by placing bending and shear-conducive lay-ups where they are most effective. Now, all of the out-of-plane loads caused globally by internal pressure loading, or locally by structural discontinuities, can then be reacted and ultimately controlled by the stitched interfaces.

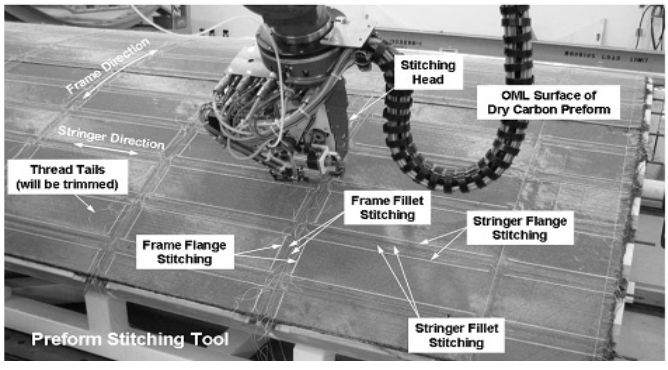


Figure 10. Single-sided stitching operation.

Once the stitching thread becomes a crucial part of the overall damage control approach within the design scheme, it must be accurately controlled and inspected in much the same fashion as mechanical fasteners are in conventional aluminium panels. This now mandates precise control, repeatability, and inspection methods be followed during the stitching operation. The location and placement of stitch seams is accurately controlled using a robotic stitching system that utilises a single sided, two needle end effector that forms a modified chain stitch on the interior surface of the preform (Fig. 10). An industrial robot arm gives the end effector six degrees of freedom for stitching in 3D space. The stitching seams are visually inspectable from either side of the preform, and can easily be removed or redone prior to moving the preform.

Stitching seams are strategically placed at all the critical transitions where stress concentrations are prevalent. Each pass of the sewing end effector lays down a three dimensional seam consisting of two rows of stitching: one to trap the flange edge and the other to trap the fillet radius. The six rows of stitching in the stringer member are shown in Fig. 11 (for a cured panel). In the photograph, the backside thread loops are clearly visible in the stringer and frame flanges.

Once the stitching is complete, the dry fabric preform assembly is placed on a mold tool, vacuum bagged, infused with resin, and cured in an oven using the CAPRI (Control Atmospheric Pressure Resin Infusion) low pressure resin infusion scheme. (Ref. 1)

The completed one piece panel construction (Fig. 12) is a radical departure from the conventional multi detail, laminated and bonded composite panel assemblies that represent today's state of the art designs. The PRSEUS next generation panel design is a single piece structural assembly with seamless transitions between the skin, stringer, and frame members that are reinforced with through thickness stitching to better react out of plane loading and damage propagation. The stitched interfaces afford higher levels of damage tolerance that permit visual inspection methods because the strength of the structural interface can be visually assessed based on the appearance of the stitch rows; in much the same manner as mechanical joint strength can be evaluated by inspecting the quality of the fastener installation. Maintaining these high levels of damage tolerance, structural redundancy, and visual inspection schemes is the hallmark of transport airframe design, and until the advent of stitching, this was absent from conventional composite design practices, ultimately limiting their effectiveness in primary structure applications.

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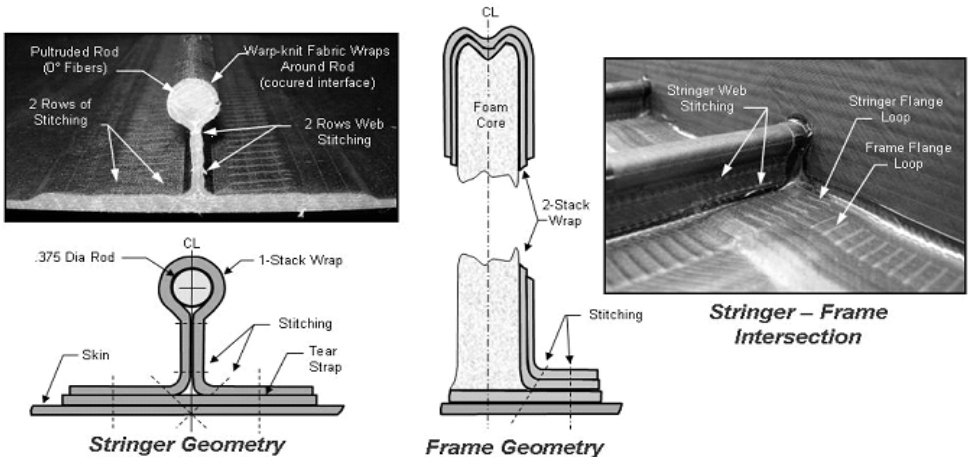


Figure 11. Stitch seam locations in stringer and frame members.

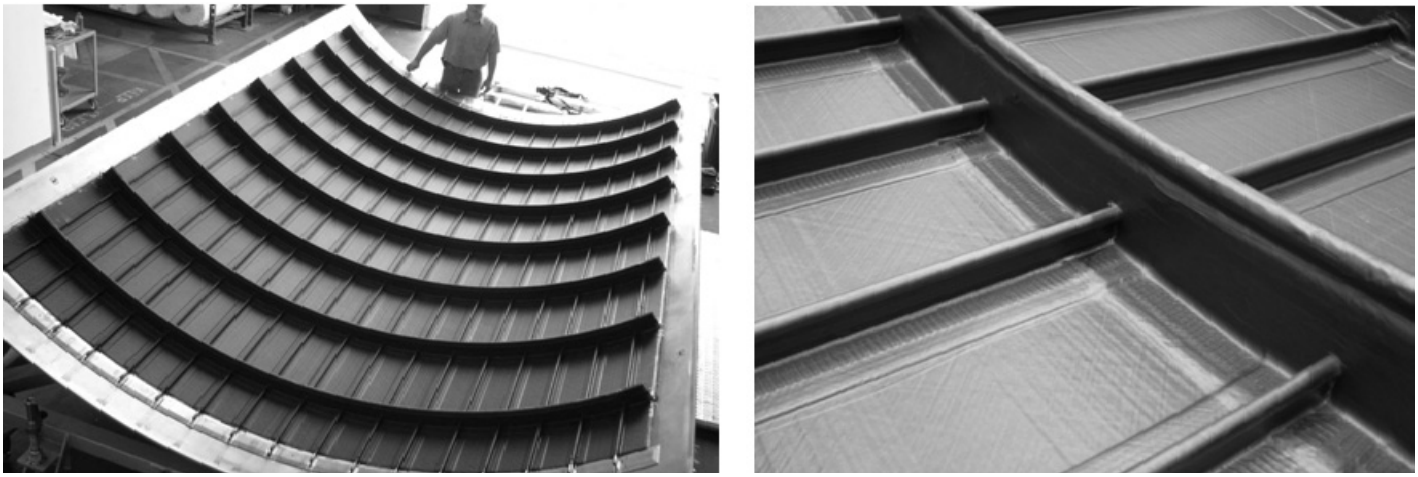


Figure 12. PRSEUS integral fuselage panel fabrication.

6.0 CONCLUSION

To meet the challenges of tomorrow, researchers at NASA and Boeing have developed a highly engineered stitched composite structural solution that moves beyond traditional composite design practices to offer a highly effective design that can be operated beyond conventional no-growth design constraints. The result is a very efficient airframe structure that combines the skin, stringers, and frame elements into an integral panel solution that is optimised for reacting the complex loadings of a transport airframe, as well as addressing the producibility challenges intrinsic in the fabrication of composite structures. This unique design-and-manufacture approach represents a bold vision in composite design theory and manufacturing methods. It is a radical departure from the conventional multi detail, laminated, and bonded composite assemblies because it evolves beyond the constraints of pre-impregnated materials and fully embraces the potential afforded by dry carbon fabrics to design larger, more integrated, cocured structures with seamless transitions and damage-arrest interfaces that offer superior performance at reduced fabrication costs.

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