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ADVANCED COMPOSITE DEVELOPMENT FOR LARGE TRANSPORT AIRCRAFT

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Abstract

Advanced composite materials promise significant cost and weight savings when applied to aircraft structure. For example, an internal Boeing study has shown a cost reduction compared to aluminum when an advanced composite wing is manufactured using the automated methods currently under development. A similar study has shown a cost reduction for a fuselage shell manufactured of advanced composites rather than aluminum. Both the wing and the fuselage shell advanced composite concepts reduce the structural weight by 20% to 30%. Even greater weight and cost reductions are believed possible with resized aircraft, emerging material improvements, and innovative designs that exact more performance from the advanced composite material.

The subject of design strain levels for advanced composite structure is a basic issue. Ultimate design strains are influenced by damage tolerance criteria in both tension and compression. Tension designed structure is controlled primarily by large-area damage. Compression designed structure is controlled by either large-area damage or residual strength after impact. Ultimate design strains of 0.006 in/in in tension and compression are feasible with today's materials and designs.

Design criteria as well as design concept and material selection have a significant influence on ultimate design strain levels and manufacturing cost. The emergence of new materials and the influence of design simplicity on structural efficiency and projected manufacturing costs are reviewed. The discussion focuses primarily on large wing structure but considers empennage and fuselage applications as well.

The future for advanced composite materials in large transports is projected and related to new material developments and future new programs.

Need

In recent years the distribution of airline costs has changed from dominance by fuel and other direct operating costs to a situation where ownership costs are equal in importance (fig. 1). New technology incorporation in modern commercial transports has contributed to an escalation of new airplane prices. Consequently, airlines are saying new airplanes cost too much, thereby making an acceptable return on investment more difficult. They look upon airplanes as nothing more than a tool used to deliver a product to their customers. If the tool costs too much, they will look for other ways to produce a profitable product.

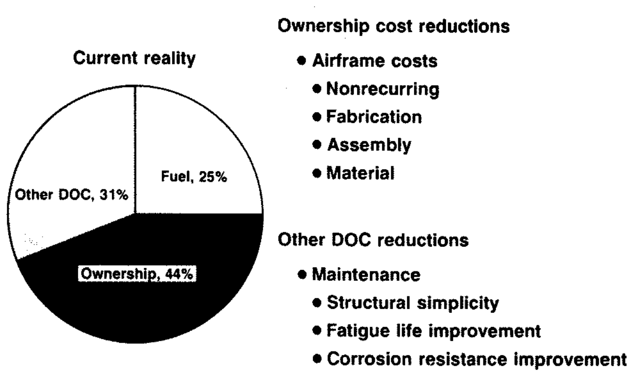

Figure 1. Change in Direction: Composites' Influence on Direct Operating Cost

The cost issue has presented a major hurdle to the incorporation of composites in heavy primary aircraft structures. Material costs are high. Fabrication has largely been labor intensive with hand layup and many fasteners.

As a contrast to the rising airplane costs, composites offer the potential to reduce direct operating costs significantly. Maintenance costs associated with corrosion and fatigue will become greatly reduced since composites are not sensitive to these phenomena. A simpler structural configuration will mean fewer problems simply due to fewer parts and less complex assemblies.

Potential

The cost and weight reduction potential is illustrated in Figure 2. The wing and fuselage account for equal fractions of aircraft structural weight, i.e., they have equal importance in weight consideration. The fuselage, however, is the highest cost per pound of structure and offers the greatest potential for cost reduction. A reduction in cost down to the average cost for the total airplane would represent a savings of approximately 25%. It can be shown that a 1% savings in aircraft weight empty plus operator's items will increase available revenue payload by almost 4%.

Cost reduction opportunities can be grouped in four categories: structural simplicity, improved materials, improved processing, and automation. Simplicity in design and assembly, i.e., a more monolithic, less structurally tailored laminate orientation and ply dropoff scheme, results in fewer parts and therefore fewer parts to fasten together. A part/part card comparison (fig. 3) among innovative composites and conventional construction shows reductions of approximately 80% to 85% of parts and part cards. This can translate into as much as a 35% cost saving on a production run of approximately 300 airplanes. Toughened materials that simplify layups and require less concentrated damage tolerance features also influence design simplicity and thus cost. A more tolerant fabrication process that has a larger "window" for process variables such as material outtime, temperature, and pressure variation can influence cost by allowing more flexibility in the manufacturing processes. Automation features including material kitting, handling, placement, and trimming also offer great potential for lower production costs.

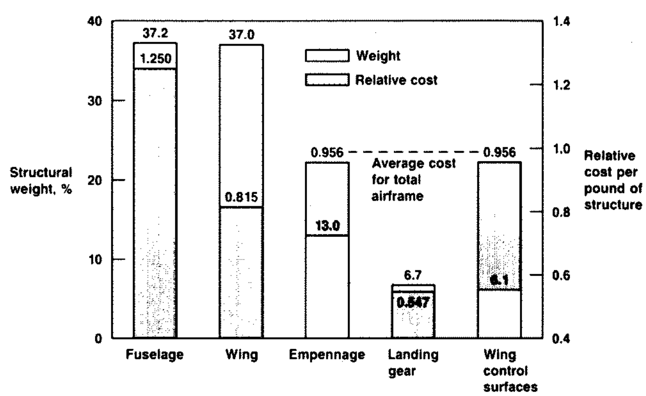

Figure 2. Typical Commercial Transport Component Weight Distribution and Relative Cost Breakdown

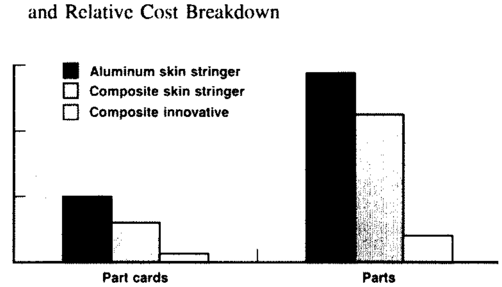

Figure 3. Part/Part Card Comparison

Figure 4 traces apparent costs of typical composite wing structure over the last decade and projects future costs that are equal to or less than conventional aluminum wings. Reductions in all areas itemized and previously discussed contribute to an overall cost reduction.

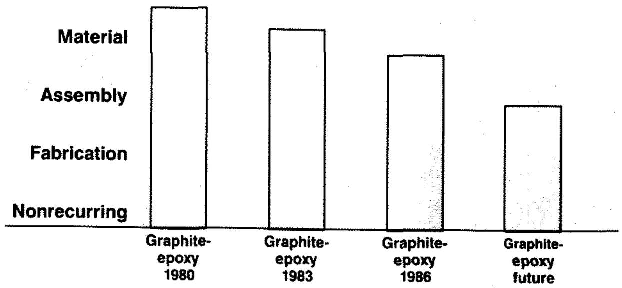

Figure 4. Typical Wing Structure Comparative Cost

Weight reduction potential is directly proportional to magnitudes of design loads. At the higher end load range for wing structures (20 to 30 kip/in) structures are thick and stable and not easily affected by impact damage. Empennage and fuselage structures are dominated by thinner, postbuckled designs that are more influenced by impact damage possibilities. Fuselage structure is punctuated by many cutouts, support of interior systems, and attachments for wing, empennage, and gear and has complicated requirements for pressure containment and damage tolerance. The fuselage has a high potential for cost and weight reduction but presents a greater challenge for accomplishment when compared to wing or empennage structure. Figure 5 illustrates current and future potential weight distributions and reductions.

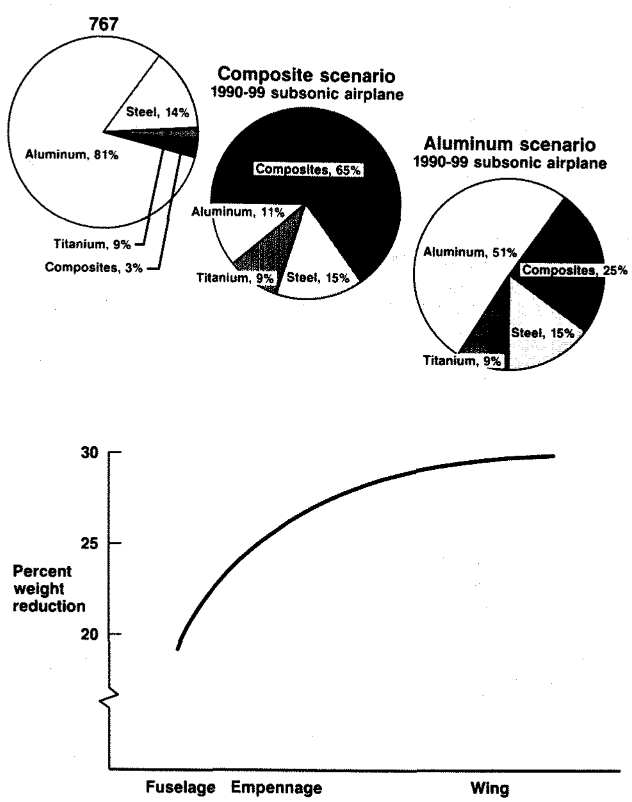

Figure 5. Materials Weight Distribution and Reduction

Current airplanes, such as the Boeing 767, utilize up to 3% of structure weight with composites. With dedicated effort and design of airplanes using composites as baseline material, it is projected that 65% of the structure could be composites. This projection presumes vigorous composites development success, improving materials database, and a continuously improving manufacturing development success. It also presumes that wing, fuselage, and empennage all share in this development success.

If aluminum remains as baseline material for structure design and composites technology development is not at a ready status for consideration on all structures, it is estimated that up to 25% of airplane structure weight could be composites.

Technical Issues

Many of the principal technical issues facing composites incorporation into transport aircraft primary structure are shown in Figure 6. Some of the concepts are common to wing, empennage, and fuselage and others are more specific to each component. The following discussion will focus on three major issues: impact/damage tolerance, postbuckled structure design, and economic repair in the field. Surface panel development and tests will be reviewed. Figures 14 and 15 show typical design details.

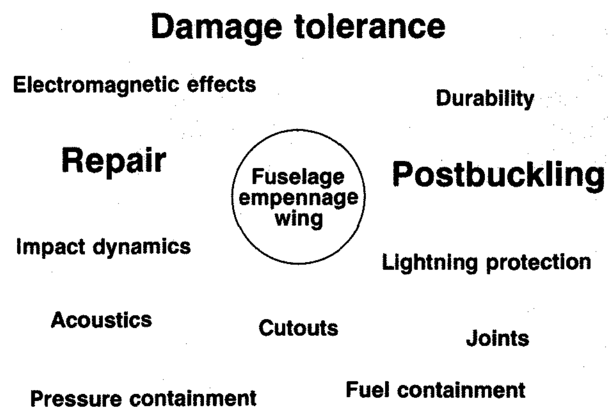

Figure 6. Technical Issues

Primary structure surface panels can be made from stiffened laminate or honeycomb construction. Damage tolerance impact damage criteria and manufacturing cost typically dictate the selection. In general the stiffened laminate construction satisfies more requirements in the heavy end load range. Honeycomb panels are more damage sensitive and repair considerations have to be weighed in the structural configuration decision making. The use of aluminum core provides much better damage tolerance but may add risk of possible corrosion in the presence of service damage.

One-piece spars and ribs with cocured or cobonded stiffening are low-cost structure. However, in ribs especially, honeycomb construction has generally shown to be lowest cost. Ribs inside the box or shell structure are protected from damage exposure and nonmetal core is the preferred construction.

A typical wing panel validation program is outlined in Figure 7. An assessment of panel damage potential in manufacture and service contributes to a panel design criteria formulation. Material candidates are screened at the coupon level for strength and damage tolerance characteristics to meet specific design requirements. Cyclically loaded small coupon tests assess damage growth potential under the design load spectra.

After the material choices are narrowed to one or two, larger elements such as three-stiffener panels are tested to evaluate damage growth potential in a more complex structure. In addition, methods to enhance performance, such as stitching stiffener to skin, are evaluated at the larger element level.

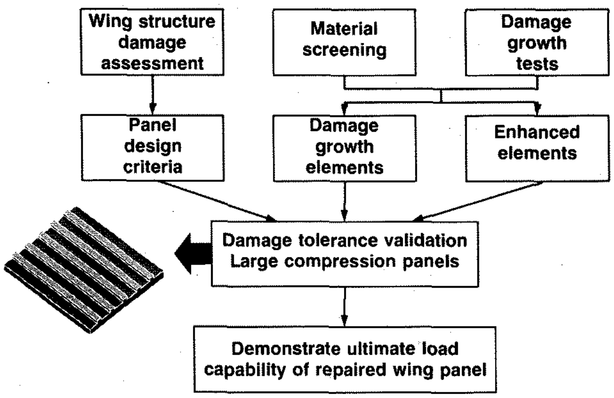

Figure 7. Wing Panel Damage Tolerance Validation Program

A material and method for panel performance enhancement are then selected for large five-stiffener panel verification. A series of tests to evaluate the various levels of damage and associated design loads are conducted to validate the design.

The results of studies by Boeing on heavy wing panel construction are shown in Figure 8. Materials ranging from older systems such as AS4/3501-6 to current IM7/8551-7 have been investigated for capability in wing design. In general the modern toughened materials are capable of design panel strain approaching 0.0060 in/in in compression and possibly higher in tension. Tests show that compression structure design in this range is much harder to achieve than tension design.

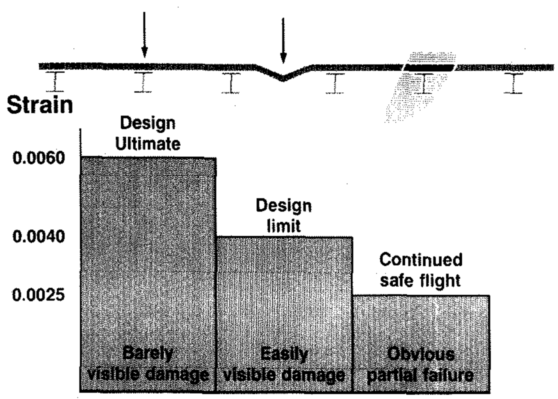

Figure 8. Impact Damage Tolerance Validation for Large Compression Panels

The following damage definitions are offered for discussion as they relate to design load requirements. Barely visible damage is that which is just below normal visual inspection detectability and is presumed to go undetected, unless it should grow in size and be discovered by some means. Boeing criteria require structure with barely visible damage to be capable of withstanding design ultimate loads (DUL) without failure.

Easily visible damage is defined as that which will be found during planned maintenance inspection. Damage growth characteristics would be such that it would not grow to unsafe proportions prior to discovery. Structure damaged thusly must withstand two-thirds DUL or design limit loads.

Damage caused by uncontained foreign object damage such as engine burst fragments or other uncontrolled high-energy rotating machinery within the airplane and whose effects on the airplane would be noticed by the pilot is defined as obvious partial failure. This damage would be dispositioned before the next flight of the airplane. This type of damaged structure must withstand continued safe flight loads of approximately 40% DUL.

The ability to repair service damage is a critical issue with airline operation. Relatively quick and economical repairs are necessary to minimize airplane downtime and not have a significant effect on maintenance costs.

Repair capability for thin and thick structure made from newer toughened materials has been demonstrated for tension and compression loaded structure. Repairs using mechanical fasteners to attach thin steel or titanium plates, for example, are ideal to fit airline needs and have been tested to panel design strain values ranging from 0.0035 to 0.0060 in/in for empennage and wing respectively.

Figure 9 shows a cross section where a section of skins and stringers was repaired using the technique described above. Splice members form a double shear joint utilizing very tight fitting titanium fasteners. Protruding head fasteners are used for best splice effectivity.

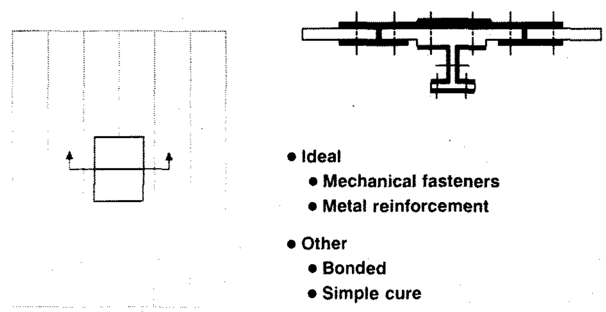

Figure 9. Wing Panel Repair

If bonded repairs are needed in critical detail locations, they should be based on typical field repair depot capability, i.e., simple cure cycle at lower temperatures.

For fuselage type structure, the effects of curvature, biaxial loads, shear, and out-of-plane pressure loads present a complicated design problem. Tear straps, i.e., local concentrated masses of material, serve to arrest or contain damage until it is discovered. Figure 10 shows that load capability is directly proportional to percent stiffening and tear strap spacing.

Damage oriented longitudinally along the fuselage shell would be critical for hoop pressure loading. Tear strap effectivity would be provided by circumferential frames and strap material buried in the skin beneath the frames.

Damage oriented circumferentially would be critical for longitudinal loads. Longitudinal stiffeners would provide the required tear strap material.

Figure 10 shows the trend of hoop pressure combined with longitudinal inplane tension loads in the presence of longitudinal damage. The failure mode for this condition is circumferential tearing at each end of the damage. This mode of failure is more acceptable than one that progresses fore and aft along the fuselage. The damage is self-relieving from the local effects of internal pressure, and pressure leakage will allow easier and earlier discovery of the damage.

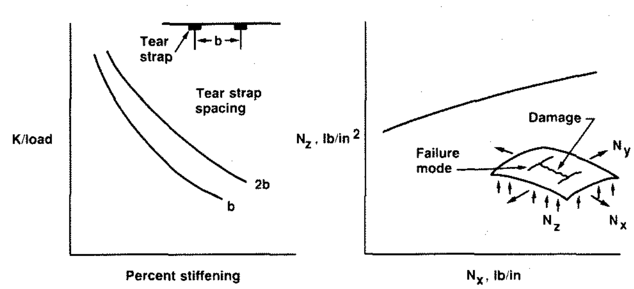

Figure 10. Pressure Containment/Damage Tolerance

Figure 11 shows the test setup representing fuselage internal pressure application on a curved panel.

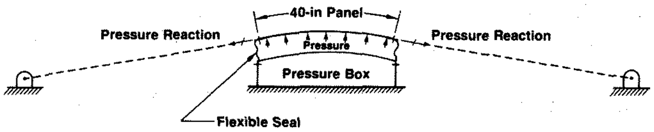

Figure 11. Curved Fracture Panel Pressure Test Setup

A postbuckled skin panel design, for lower end loaded empennage, for example, depends on specific skin/stiffener interface characteristics. Boeing experience has shown that required damage tolerance criteria can typically be met with designs where initial buckling of the skin occurs above 50% of design ultimate load. The effects of impact damage are considered in the postbuckling design. Figure 12 shows results of combined compression and shear load tests on Boeing proposed 7J7 empennage panels.

The systematic method of screening materials and damage tolerance features through successive levels of specimen size and complexity has proven to be an economical process of arriving at cost-effective panel designs that meet design requirements.

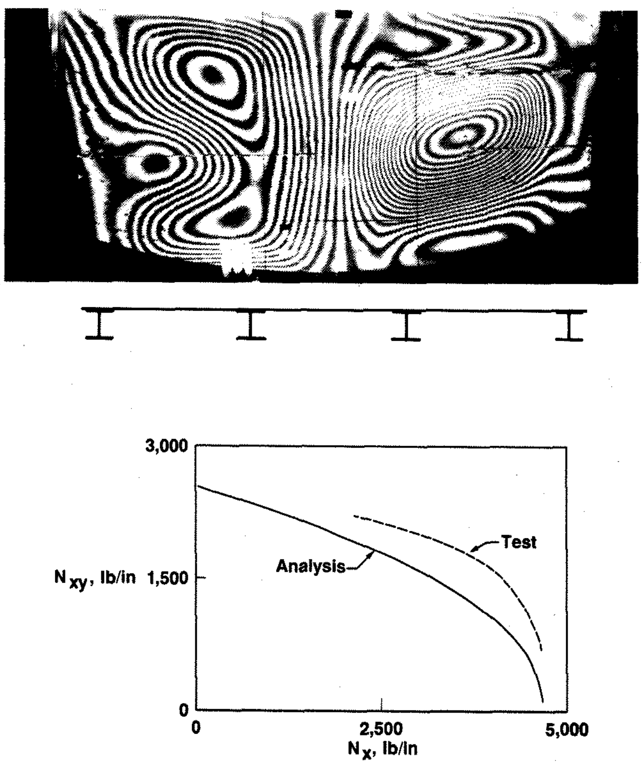

Figure 12. Postbuckled Panel Test for Combined Compression and Shear

Achievements

Boeing current production airplanes use composites as shown in Figure 13. These composite parts account for 3% of airplane structural weight. In addition, Boeing has had five shipsets of 737 composite horizontal stabilizers in commercial service for approximately 4 years.

Figures 14 and 15 illustrate the composite structural development at Boeing Commercial Airplanes in recent years. Some NASA-sponsored development at Boeing is included in the summary.

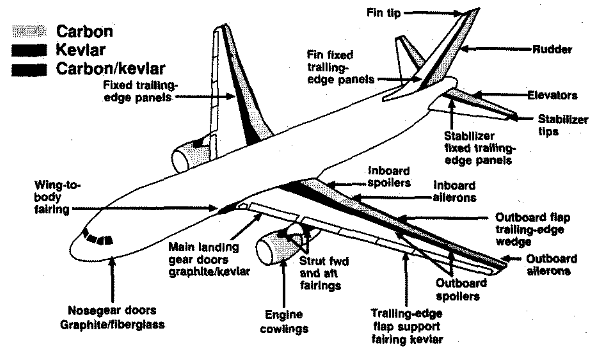

Figure 13. Current Boeing Commercial Airplanes Composite Usage

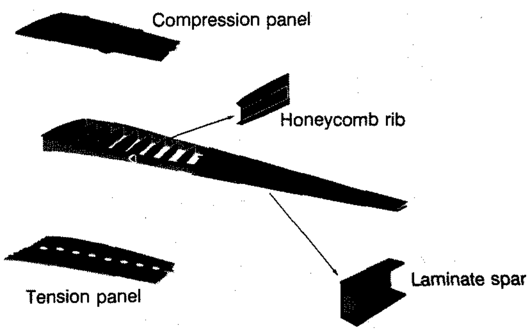

Figure 14. Wing and Empennage Component Development

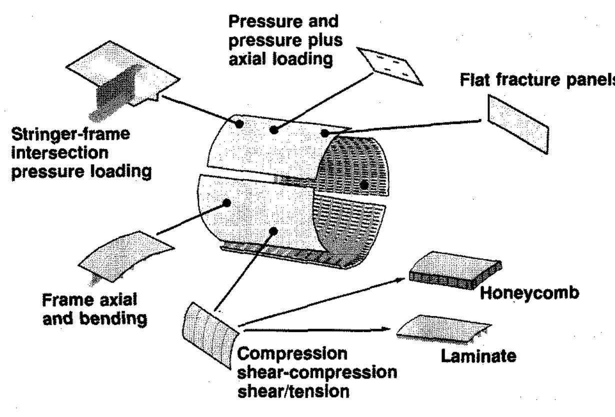

Figure 15. Fuselage Component Development

Wing, fuselage, and horizontal and vertical stabilizer design, analysis, and test development have been accomplished. Materials used include those with extensive industry database as well as new toughened systems.

Near-term application of composites to primary structures was to have included the new Boeing 7J7 airplane horizontal and vertical stabilizers. Advanced composites were ideally suited for the vertical stabilizer because of the need for sonic fatigue resistance and good damage tolerance in the presence of the aft-mounted unducted fan engines. Since the postponement of the 7J7, Boeing has concentrated on composites application to future derivative and new airplanes.

Future

Figure 16 illustrates the future potential of weight and especially cost savings associated with design and manufacturing techniques of unitized structures. The relatively high-part-count composite 737 stabilizer in service is compared with a design where honeycomb stabilized skins and spars are combined to form essentially a one-piece construction. Likewise, a similar construction is envisioned to replace the high-part-count, high-cost conventional fuselage shell.

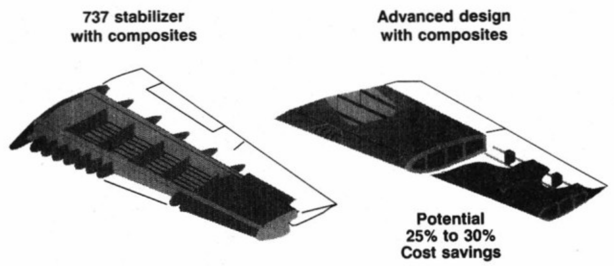

Figure 16. Advanced Composite Design Concept

Improvements in composite matrix materials and stronger high-modulus fibers have substantially changed the design possibilities for today. Figure 17 compares hot/wet compression strengths with compression strengths after impact (CAI). Recent toughened material test data indicate strengths almost twice what was achievable a few years ago in CAI.

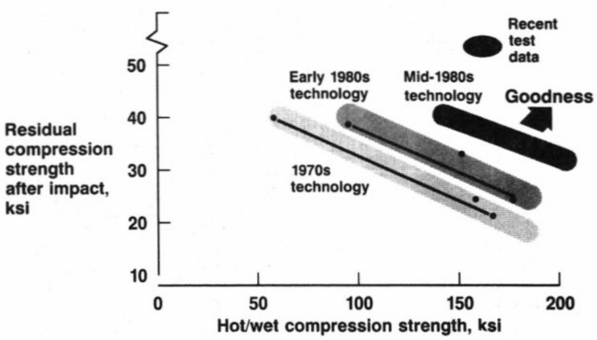

Figure 17. Composite Matrix Improvements—Coupon Test Data

Figure 18 compares damage areas on typical thick heavily loaded structure made from toughened and state-of-the-art materials. Generally primary structure allowables are a function of damage size and CAI. Lighter, simpler, and less costly structures as shown in Figure 19 are more achievable with the new materials.

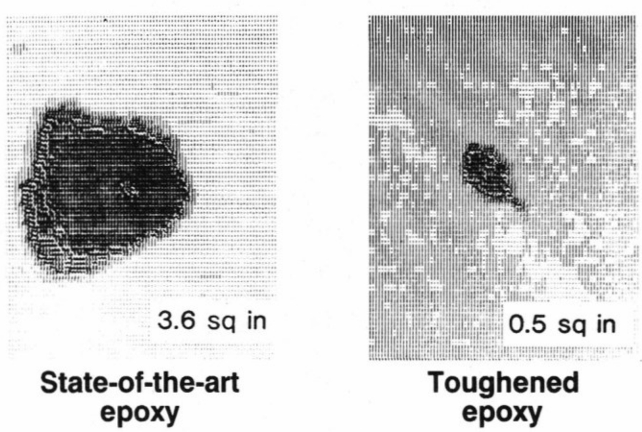

Figure 18. Composite Damage Tolerance

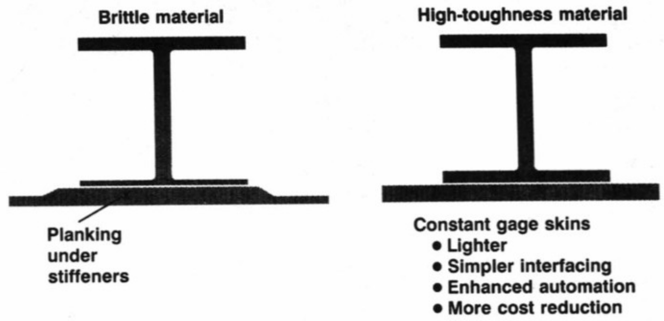

Figure 19. Potential Design Simplification for Toughened Composite Materials

Summary

The basic technical issues facing large-scale heavy composite primary structure incorporation in production are understood. A large body of test data is available. Large composite component tests are expensive. To reduce test costs we must have a greater understanding and validation of analysis methods, especially damage tolerance methodology, and a predictive allowables capability. We must be able to predict service environment response and design for it.

Above all, cost is the basic driver affecting the success of advanced composites in large commercial transport production. The implementation of simplicity and innovativeness in design together with an automated, cost-effective manufacturing scheme will make the success story for composites real.