ADVANCED COMPOSITES RESEARCH AND DEVELOPMENT FOR TRANSPORT AIRCRAFT

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Abstract

This paper highlights past experiences, lessons learned, state-of-the-art and current research activities directed at providing an integrated "affordable" data base for composite structures. Composite secondary and empennage structures are in production on several transport aircraft. The weight reduction potential of composite structures is well documented. However, the cost to develop and produce composite structures remains the major barrier to increased application of this technology to transport aircraft. Specific technology items that are being developed under the NASA Advanced Composites Technology Program are described. Materials, design concepts, structural mechanics methodology and manufacturing processes and equipment are under development or are emerging that are expected to lead to an integrated "affordable" data base. Technology verification for the next decade is expected to require fabrication and testing of full-scale wing-box and fuselage-section components before certification can occur and production committments can be made.

State-of-the-Art

The application of composite structures to transport aircraft is proving to be an evolutionary process. The National Aeronautics and Space Administration (NASA) has sponsored the development, certification and flight service evaluation of composite secondary and empennage structure(see Table 1).[1] An excellent bibliography that contains synopses of over 600 references on NASA sponsored research between 1976 and 1986 was published in 1987.[2] Spoilers for the Boeing 737, one of the components in the NASA flight service evaluation program, have been in service since 1973 and tests after 15 years of flight service indicate no loss in residual strength (see Fig. 1).[3] Lessons were learned during these development and flight service experiences and the level of confidence was built that made possible the production commitment of secondary composite structures on the Boeing 767 and the baseline designs of numerous components for the McDonnell-Douglas MD-11 and the C-17 (see Figs. 2 and 3).[4]

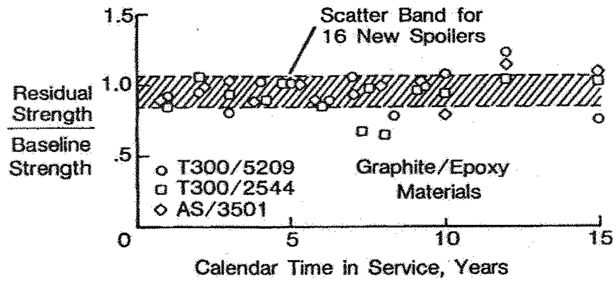

Fig. 1 Residual strength of B-737 graphite/epoxy spoilers.

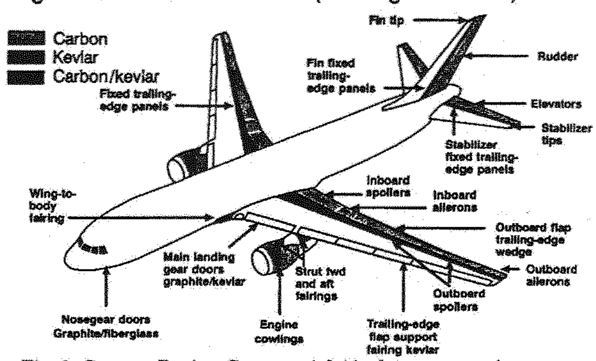

Fig. 2 Current Boeing Commercial Airplanes composite usage.

Table 1 Composite structures developed under NASA ACEE program.

Component	Secondary: DC-10 rudder	Secondary: 727 elevator	Secondary: L-1011 aileron	Empennage: DC-10 vertical stabilizer	Empennage: 737 horizontal stabilizer	Empennage: L-1011 vertical fin
Size (root X span), ft.	3.2 X 13.2	3.4 X 17.4	4.3 X 7.7	6.8 X 22.8	4.3 X 16.7	8.9 X 25
Metal design weight, lb.	91	130	140	1,005	262	858
Composite design weight, lb.	67	98	107	834	204	622
Weight reduction, %	26	25	24	17	22	28
Number of production units	20	11	12	3	11	2
Start flight service	4/76	3/80	3/82	1/87	3/84	-

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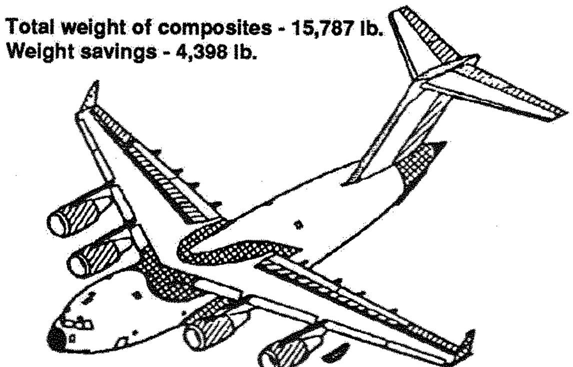

Fig. 3 Advanced composites applications to C-17.

However, the full exploitation of the benefits of composites requires application to wing and fuselage structures which account for approximately 75 percent of the aircraft structural weight.

Potential Benefits

Composite structures are recognized as enabling technology for making significant advances in the performance of subsonic transports and are essential to achieving an economically viable supersonic transport. Prior research and development programs have demonstrated weight savings greater than 25 percent and, when the effects of improved damage tolerant materials and resizing the aircraft are taken into account, the weight savings for some components may approach 40 percent. Composite structures that have been designed as replacements for metallic structures contain significantly fewer parts and numerous studies have noted the direct relation between part count and cost. Cost estimates that range from 10 to 40 percent less than metallic structure have been reported for the projected acquisition cost for composite structure in future production.[5] The weight savings of composite structure translates into about 10 to 15 percent savings in fuel.[6] Outstanding resistance to environmental degradation and residual strength after exposure to cyclic fatigue loading offer the potential for building aircraft with longer life and reduced maintenance cost.[7] Recent studies on supersonic aircraft indicate that composite structures are essential to holding the gross takeoff weight to less than 750,000 pounds for a 5000-nautical-mile-range economically-viable design.[8]

Inhibitors to Application

The major obstacle to increasing the application of composites in aircraft structure has been acquisition cost. In the mid 1980's, data indicated that the cost of composite structures was substantially more than the cost of similiar type metallic components (see Fig. 4).[9] Other obstacles have included: the lack of an integrated data base with a maturity similar to metals, the high initial cost of materials and the lack of automated affordable manufacturing processes. Several learning experiences

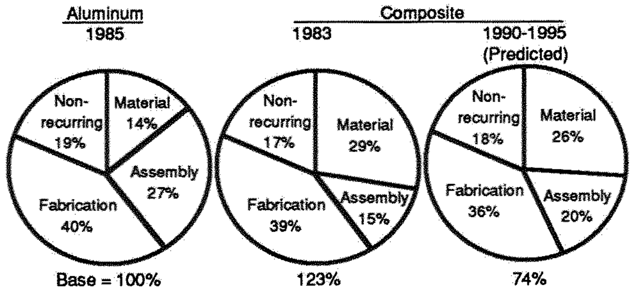

Fig. 4 Comparison of costs for metal and composite wing structure.

on the effects of moisture absorption, damage tolerance and hail damage have also slowed the introduction of composites but are viewed as manageable barriers.

Approaches for Overcoming Inhibitors

In 1984 NASA requested the Aeronautics and Space Engineering Board of the National Research Council to form a committee chartered to assess the status and viability of organic matrix composites technology for aircraft structures. The committee findings and recommendations were reported in 1987 and recommended that NASA, the United States Department of Defense (DoD) and the Federal Aviation Administration (FAA) establish a joint program to develop an integrated "affordable" composites technology data base.[10] The term "data base" used herein relates to materials and structural matters required to reduce the risks of new composite designs to levels acceptable to designers and chief engineers. Pertinent data include materials properties, test methods, analytical methods for predicting structural response and strength, manufacturing and inspection methods.

Perhaps the most important approach for achieving an "affordable" data base is through the integration of the design and manufacturing disciplines. During the last several years this concept has become popular and is referred to in the literature under several labels that include "Total Quality Management", "Concurrent Engineering" and "Design To Build Teams".

In 1988 NASA initiated a new program, " Advanced Composites Technology" (ACT), in response to the

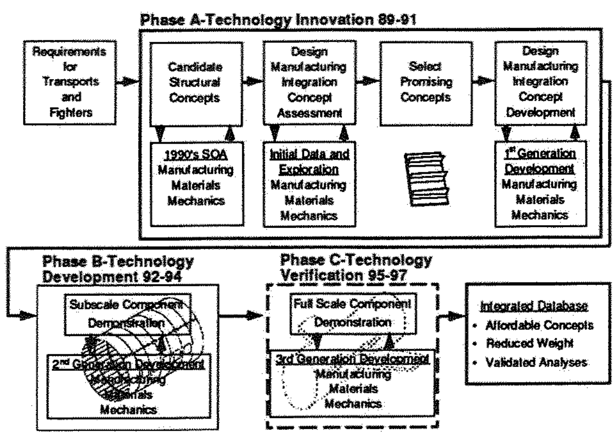

Fig. 5 Advanced Composites Technology Program logic chart.

National Research Council committee's recommendations. The ACT program logic emphasizes the Design/Manufacturing Integration (D/MI) approach (see Fig. 5). The ACT program focuses on development of advanced materials, mechanics of materials, structural mechanics, manufacturing process methods, and innovative concepts. The remainder of this paper will provide additional detail on the following items that are contained in the ACT program and are essential to achieving an "affordable" composites data base: D/MI, improved materials and materials forms, simplistic and tailored concepts, improved and validated structural mechanics analyses and innovative lower cost manufacturing methods.

Design/Manufacturing Integration

All disciplines that have an influence on the performance requirements and cost of airframe structure are brought together at the initiation of the design process in the D/MI approach (see Fig. 6). This is a

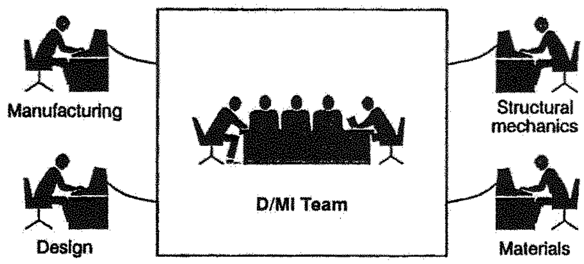

Fig. 6 Design/manufacturing integration (D/MI) progress.

cultural change for most organizations where one discipline tends to be viewed as the leader. The lead discipline has varied from one organization to another. In some organizations the leader has been engineering, in others the leader has been manufacturing, in others the leader has been finance, or some other discipline. A D/MI team will assess the design requirements and constraints in the beginning of the design process and evaluate their influence on performance and cost. Trade studies will be conducted, requirements will be

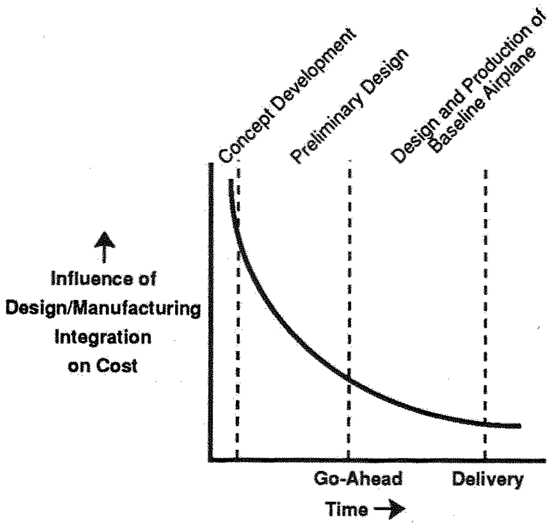

Fig. 7 Potential influence of design/manufacturing integration.

questioned and changed if appropriate, and the best possible concept will be defined by the D/MI team. The potential influence of the D/MI team on the cost of producing structure is greatest at the beginning of the design process and decreases rapidly thereafter (see Fig. 7). Once the design concept is selected, the D/MI process probably cannot reduce cost more than one-third. The D/MI team approach is also an excellent approach for conducting cost and weight trade assessments. A schematic of the DM/I process for a simple stiffened skin compression panel shows the influence of design concept on cost and weight (see Fig. 8). Data from a comparison of "I" and "J" stiffened

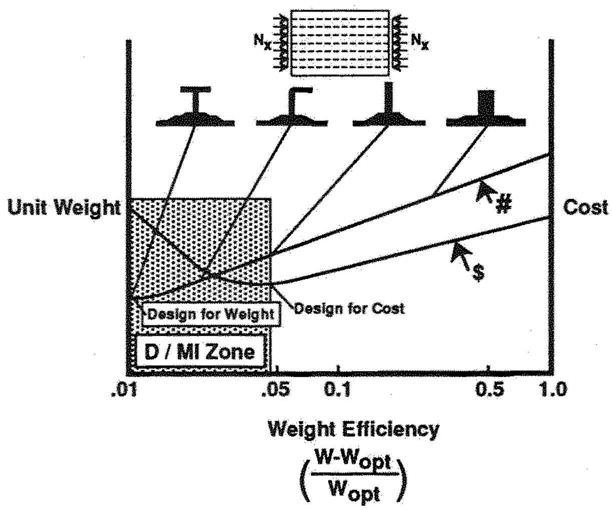

Fig. 8 Influence of design/manufacturing integration.

damage-tolerant panels that have been fabricated and tested indicate that the "J" stiffened panel is more expensive but lighter than the "I" stiffened panel (see Fig. 9). The airframe companies have begun to practice the D/MI approach and large benefits from this approach will appear in the next generation of aircraft.

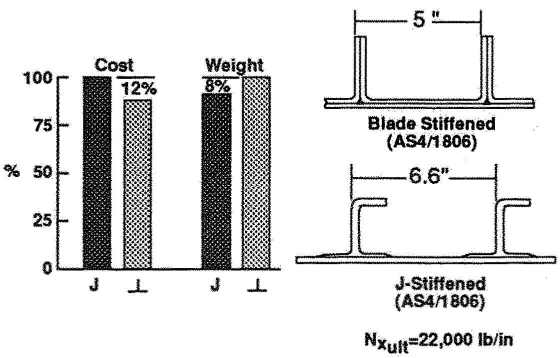

Fig. 9 Weight/cost comparison for "I" and "J" stiffened compression panels.

Materials and Material Forms

For 21st century aircraft, polymer matrices and carbon fiber reinforced composites must exhibit a favorable balance among such properties as toughness, damage tolerance, processibility, good mechanical properties under hot/wet exposure, and acceptable cost. The major resin matrix for almost all the current secondary and medium primary subsonic aircraft structure in service today is based on a brittle 350°F curing epoxy system such as contained in Narmco's 5208, Hercules' 3501-6 and 3502, Fiberite's 934, and Hexcel's F263 epoxies. These brittle matrices in conventional composite form do not exhibit the needed toughness and damage tolerance required of heavily-loaded primary aircraft structure.

Consequently, new 350°F cure toughened epoxy composites have been developed to overcome the damage tolerance problem. Representative examples include Hercules' 8551-7, ICI-Fiberite's 977 and Toray-Hexcel's 3900-2 toughened epoxies. Hot/wet unidirectional (0°) compressive strengths and compressive strengths after impact of these toughened materials equal or exceed typical materials specifications and far exceed those of the first generation brittle epoxy composites cited above.[11,12] However, prepregs of these toughened materials are at least three times the cost of those of the standard brittle systems and are considered to be too expensive for widespread use on commercial aircraft (see Fig. 10).[13] Although prepreg cost is related

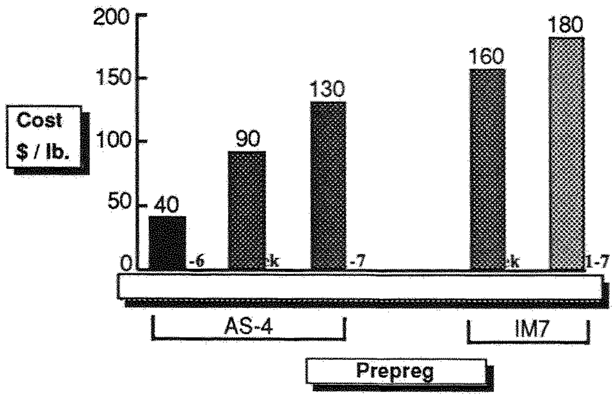

Fig. 10 Composite materials cost (today).

to market quantity and composite costs result from a complex set of factors, and not just material costs alone, it seems reasonable that use of these advanced materials would have a major impact on final product cost.

Thermoplastic composites have received considerable attention over the past 6 years because they offer an attractive combination of mechanical properties, toughness, and the potential of low-cost manufacturing.[14,15] Relative to thermosets, they offer simple formulations, excellent prepreg stability with infinite out-time and no cold storage requirements, and fast thermoformability. A few of their more obvious drawbacks include tackless/boardy prepreg, high processing temperatures, poor fiber alignment, fiber waviness, relatively poor 0° compressive strengths, difficult prepregging and a small data base. Again, as with toughened thermosets, thermoplastic prepreg is much more expensive than the standard brittle epoxy prepreg (see Fig. 10).[13] Thus, efforts are underway in the ACT Program to develop affordable thermoset and thermoplastic composites for subsonic aircraft. These efforts will be described below and involve textile preform technology, resin transfer molding (RTM) materials and fabrication technology, powder coated towpreg, tape prepreg fabrication, and commingled towpreg fabrication.

An exciting approach to the toughness/cost trade-off problem is the use of net-shaped fiber preforms which can be infiltrated by RTM processes and cured by conventional means.[16,17,18] Automated textile processes such as through-the-thickness weaving, braiding, knitting and stitching offer the potential for reduced fabrication costs of aircraft structural components. Complex fiber architectures, multilayer fabrics and structural composite preforms have already been demonstrated using automated textile technologies (see Fig. 11).[17] The

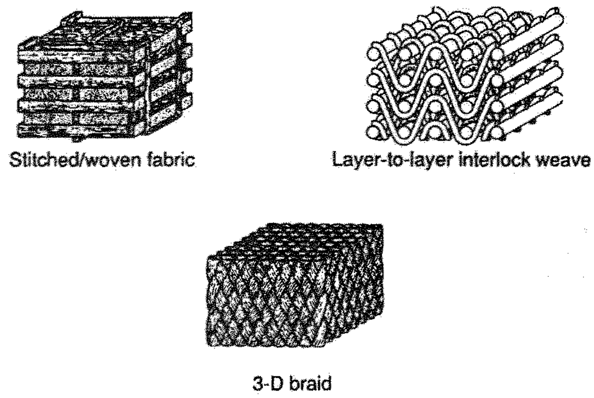

Fig. 11 Through the thickness reinforced textile forms.

preforms lend themselves to economical composite fabrication techniques such as RTM and pultrusion. Because they often have fibers in the through-the-thickness direction, composites fabricated from these new material forms and lower cost standard brittle epoxy matrix resins offer outstanding damage tolerance and out-of-plane load capability.

For example, the compression failure strain after impact of a stitched uni-woven fabric flat composite panel made with the standard brittle Hercules 3501-6 epoxy resin was excellent and far above the accepted design goal of 0.006 (see Fig. 12).[13] The toughened epoxy composite, Hercules 8551-7, made with a more expensive resin, also meets this goal in both tape and fabric form. Assuming layup and other fabrication costs

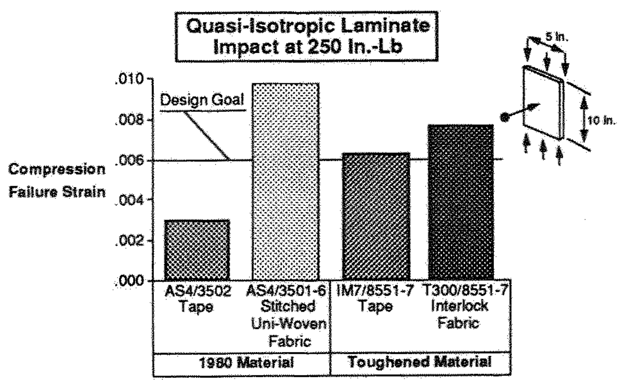

Fig. 12 Standard compression after impact for quasi-isotropic laminates, impact at 250 in.-lb.

for the two different systems are similar, the one with through-the-thickness fiber clearly offers a cost-effective alternative.

The technique for constructing stitched dry preforms from uni-weave fabric subelements is described in detail elsewhere but is briefly illustrated herein for a reinforced panel (see Fig. 13).[19,20] The resin can be either vacuum

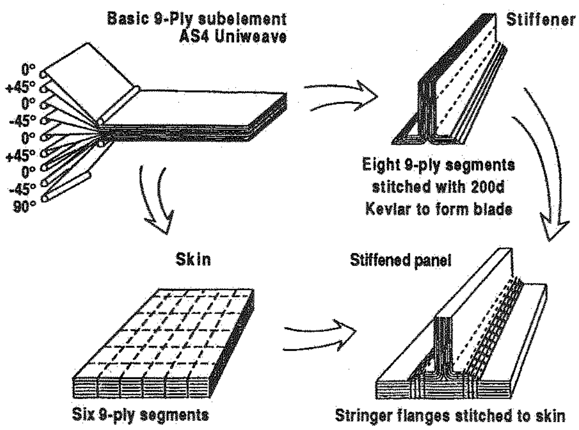

Fig. 13 Damage-tolerant stiffened panel concept

or pressure infiltrated into the preform which is bagged and supported by a die consisting of easily fabricated plate metal inserts. The layup was cured using standard autoclave procedures.

Damage tolerant single and triple blade-stringer panels were successfully fabricated from dry stitched preforms and standard Hercules 3501-6 brittle epoxy using hard tooling and RTM. For example, a 22-inch-wide by 22-inch-long three-stringer panel demonstrated excellent post-impact compressive strength (see Fig. 14).[19]

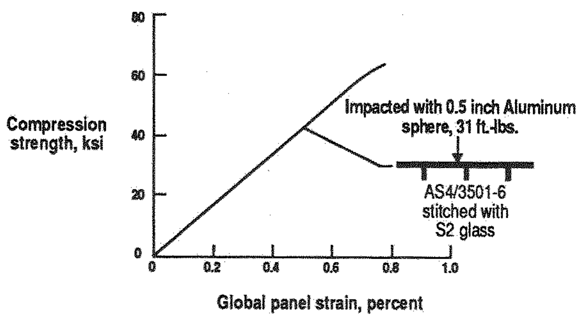

Fig. 14 Post impact compression strength of blade-stiffened panel

These and similar results in other applications have spawned the development of a wide variety of RTM resins from Shell Development Company, Dow Chemical Company, BP Advanced Materials, 3M Company, and many others. The resin requirements for RTM impregnation are very stringent. Notably, resin viscosity during impregnation must be very low to penetrate tow bundles, yet the resin must have a relatively short cure time and a relatively low exotherm at elevated temperature. Neat resin fracture toughness need not be very high, just sufficient to prevent microcracking, but all other mechanical properties must meet standard hot/wet specifications for aircraft structural applications. New toughened RTM resins are being developed under the ACT Program, including cyanate based resins, crosslinkable epoxy thermoplastics, multifunctional high temperature epoxies, and vinyl esters of epoxy novolacs.

Several new BMI resins modified for RTM processing are now available for evaluation in structural applications requiring higher temperatures such as those projected for future supersonic transports. These resins include Shell's Compimide 65 FWR, Ciba-Geigy's Matrimid 5292 A/B and Narmco's 5250 and 5260, among others.

Alternate methods for impregnating fibers with resin while maintaining the textile preform concept have been developed. In one process, polymer powder is homogeneously coated and permanently attached to 3K to 12K carbon tow and the resultant towpreg woven into fabric. The coating process is relatively simple and involves a tow spread chamber, a powder recirculating chamber, and a convection furnace for powder fusion.[21] Advantages of powder-coated towpreg as a new method to combine fiber with resin are given herein (see Fig. 15).

- Towpreg has good drape.
- Powder is permanently attached to fiber.
- Process eliminates solvents and outgassing.
- Process eliminates use of hot/melt prepregging.
- Process eliminates use of refrigeration/freezers.
- Process eliminates waste due to out-time, storage, loss of cure.
- Process is ideal for resins not suited to RTM.
- Process is applicable to thermoplastics and thermosets.
- Towpreg is easily woven.

Fig. 15 Advantages of powder-coated towpreg.

The process has been demonstrated with a variety of powders, some of which are listed in Table 2. It is especially useful for thermoplastics which otherwise are difficult to prepreg onto carbon fiber without the use of difficult hot/melt or solvent coating procedures. The mechanical properties of AS-4/LARC-TPI polyimide

Table 2 Polymers utilized in powder coating of carbon fiber tow.

Name	Source/Grade	State	Melt Flow	Process Temp., °F
1. LARC-TPI	MTC/2000	Semicry.	Moderate	650
2. LARC-TPI	MTC/1500	Semicry.	High	650
3. "New TPI"	MTC/400	Semicry.	Moderate	800
4. PMR-15	Dexter/Preimidized	Amorphous	High	600
5. BMI	Various Sources	Amorphous	High	350-425
6. Epoxy	Various Sources	Amorphous	High	350
7. PEEK	ICI	Semicry.	High	720

unidirectional and woven composites made from powder-coated towpreg are given in Table 3 and nicely demonstrate the efficacy of this new impregnating process.[22]

Table 3 Mechanical properties of AS-4/LARC-TPI composites made from powder-coated towpreg.

12K Undirectional, 2000 powder*	
Short beam shear strength, ksi (MPa)	12.3 ± 1.3 (84.8 ± 8.9)
Flexure strength, ksi (MPa)	323 ± 14.1 (2227 ± 97)
Flexure modulus, Msi (GPa)	19.4 (134)
3K 0/90 4-Harness Woven, 1500 Powder**	
0° Beam Specimens:	
Flexure Strength, Ksi (MPa)	134 ± 3.5 (924 ± 24)
Flexure Modulus, Msi (GPa)	8.1 ± 0.5 (56 ± 3)
0° Fill Specimens	
Flexure Strength, Ksi (MPa)	83 ± 4.0 (572 ± 28)
Flexure Modulus, Msi (GPa)	5.7 ± 0.6 (39 ± 4)

* Molding Conditions: 1hr/660°F/800psi; Resin Content: 33-34 wt.%
** Molding Conditions: 1hr/700°F/300psi; Resin Content: 29 wt.%

Another alternate process involves commingling thermoplastic and carbon fibers to form a towpreg which can be woven as desired, then consolidated by conventional techniques, the thermoplastic fiber becoming the matrix material. Polyarylene ether thermoplastic fibers such as PEKK (DuPont), PEEK (ICI), and PEKEKK or ULTRAPEK (BASF), among others, have been developed for this process. Composite properties of three forms of the latter are given in Table 4.

Table 4 Properties of composites made from carbon fiber/Ultrapek commingled tow, BASF Structural Materials, Inc., NAS1-18834

	Unidirect.	Uni-fabric	8 harness satin fabric
RT flex. strength, ksi	336	255	151
RT flex. modulus, Msi	15.8	16.2	9.3
RT short beam shear strength, ksi	----	----	13.5
RT compression after			
1500 in lb/impact: strength, ksi	----	49.8	----
strain, %	----	0.77	----
RT 90° Tensile strength, ksi	----	13.3	----
RT 90° Tensile modulus, Msi	----	1.5	----
RT 90° Tensile strain, %	----	0.9	----

Design Concepts

Many earlier composite structures for aircraft applications were designed following standard metallic aircraft stiffened-skin design practices. These earlier composite structures were usually developed with the design objective of reducing structural weight, and they successfully demonstrated the weight-savings potential of composite structures when compared to the stiffened metallic structures that they replaced. The lessons learned from these earlier composite structures have stimulated structural designers to begin searching for design concepts that further exploit the benefits of composite materials for aircraft structural applications.

Structural design concepts are currently being developed that can reduce both structural weight and cost. Satisfying the design requirements for reduced structural weight and reduced cost often requires a compromise between a minimum weight design and a minimum cost design (see Fig. 8). As a result of making this compromise, valuable experience is being gained that is teaching structural designers which structural design parameters can be varied to take the best advantage of new and existing materials and fabrication procedures to reduce cost without unduly sacrificing structurally efficiency. Designers are beginning to recognize that the cost of a structure is influenced by more than just the material cost. Fabrication and assembly costs also significantly affect structural cost. Design concepts that reduce fabrication and assembly costs can often be cost-effective even with more expensive materials. Simpler design concepts that reduce the number of parts or elements that have to be assembled are usually lower cost structures. For example, several structural elements can be integrated into a structure during subcomponent fabrication by cocuring the stiffeners with the skin which significantly reduces the number of fasteners, clips and shear ties needed to assemble conventional metallic stiffened structures. Sandwich structures are being considered because they have the potential for minimizing or even eliminating the need for stiffeners. Designers are beginning to develop design concepts that exploit the advantages of new and existing fabrication procedures to reduce costs. For example, automated advanced fiber placement machines can be used to fabricate integrated structures by rapidly winding or placing fibers or tows on fabrication tools or mandrels. Stiffeners can be wound into grooves in the mandrel that are oriented in any suitable stiffener orientation, such as a geodesic stiffener pattern, and then the skin can be wound over the stiffeners to produce a stiffened shell structure with minimum part count (see Fig. 16). Resin transfer molding with textile preforms is another example of a fabrication process that can be used to fabricate integrated structural components (see Figs. 11 and 13).

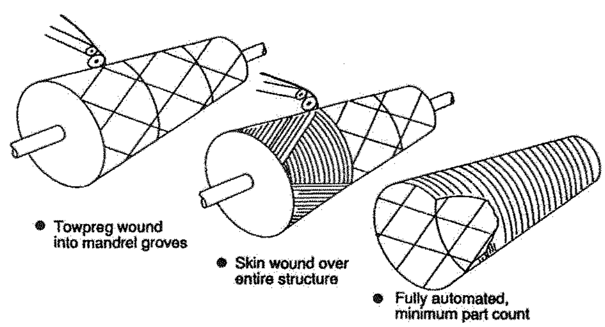

Fig. 16 Automated fuselage fabrication.

Designers are also beginning to explore structural concepts that exploit the benefits of composite materials by tailoring the structural properties and parameters to improve structural performance. Cost-effective fabrication procedures, such as the automated fiber placement procedure, can lay down fibers or tows in any desired orientation. As a result, designers can tailor the mechanical properties of the structure to provide desired structural response characteristics without unnecessarily increasing fabrication costs. Structures can be tailored to control the internal load distribution so that the deleterious effects of a known discontinuity, such as a cutout, can be minimized.[23] In some cases, the performance of a tailored structure with a cutout can exceed the performance of the structure without the cutout (see Fig. 17).[24] Undesirable aeroelastic effects

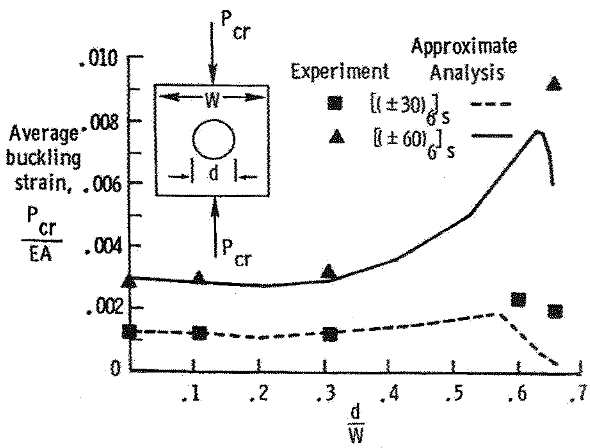

Fig. 17 Effect of a cutout on the buckling load of composite plates.

often associated with high-aspect-ratio wings and forward-swept wings, such as the X-29 wing, can be prevented by tailoring the wing box design by using composite materials to provide the torsional and bending stiffnesses necessary to control the aeroelastic behavior of the wing without significant weight increases. A tailored high-aspect-ratio composite wing concept offers the potential for reducing drag and, as a result, increasing vehicle performance.

Structural Mechanics

A fundamental understanding of the response and failure characteristics of composite structures is needed to design safe, reliable primary structures. Once these response and failure characteristics are understood, accurate structural analysis and design methods can be developed that are based on sound structural mechanics principles. Experience with earlier composite structures has shown that these structures are sensitive to local three-dimensional stress gradients that are associated with discontinuities in the structure and out-of-plane loads. Earlier structural analyses based on classical two-dimensional plate and shell models are proving to be inadequate for predicting some structural response phenomena and critical failure mechanisms that are affected by through-the-thickness or three-dimensional effects. The brittle nature of composite materials requires much more accurate analyses to predict the local stresses in composite structures. New structural concepts that exploit cost-effective fabrication procedures will have features that will require fundamental structural mechanics understanding and structural analysis and design methods that accurately represent the structural behavior of the new cost-effective concepts.

Structural mechanics technologies for composite structures are currently focused on understanding critical response phenomena and critical failure mechanisms that limit the performance of composite wing and fuselage primary structures and on developing structural analysis and design methods that accurately predict these behaviors. Nonlinear behavior, such as postbuckling response, offers challenges for predicting both response and failure. Accurate response prediction is necessary in the nonlinear postbuckling load range to determine the critical three-dimensional stress fields needed to predict failure of the structures (see Fig. 18).[25] Unstiffened panels fail along nodal lines due to

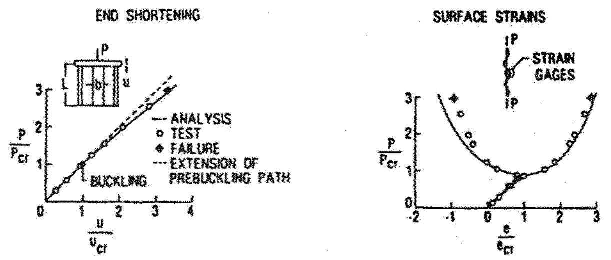

Fig. 18 Compression postbuckling of a flat stiffened graphite-epoxy panel.

interlaminar shear stresses[26,27] and stiffened panels fail by skin-stiffener separation,[25] stiffener crippling[28] or interlaminar stress failure in the skin.[26,27] Conventional design practice for pressurized fuselage structures usually assumes a linear behavior. Experience has shown that the nonlinear response of a pressurized curved panel is important for predicting the interlaminar stresses that cause failure in a composite skin panel.[29] Pressure containment for stiffened fuselage structures may require nonlinear analysis to determine accurately the local stresses in the skin at the frame-skin interface regions or at other discontinuities such as cutouts, joints or splices. Detailed local stress analyses of large-scale structural components can be made computationally feasible by embedding a refined local model needed for detailed stress analysis into a coarser global model that can be used to determine the nonlinear shell response.[30] Detailed three-dimensional stresses at local discontinuities can be calculated (see Fig. 19)[31] and

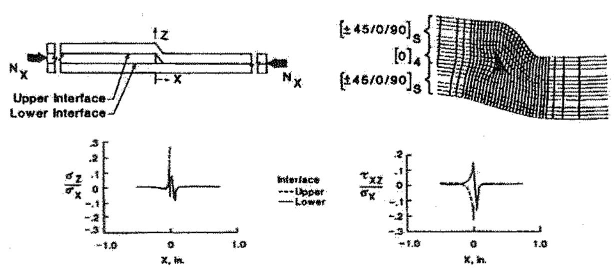

Fig. 19 Interlaminar stress gradients near a dropped ply in a compression loaded composite plate.

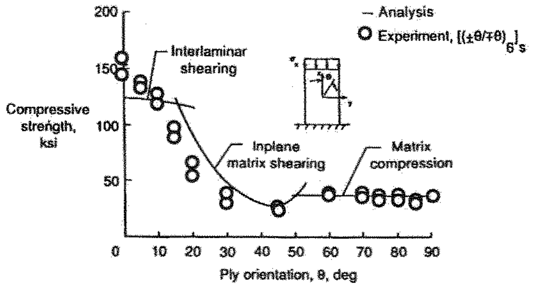

Fig. 20 Effect of fiber orientation on laminate compression strength.

accurate failure predictions can be made based on analytical models of critical failure modes (see Fig. 20).[32] Once the fundamental behavior of composite structures is understood well enough to develop verified analytical models, scaling laws for composite structures can be developed and damage tolerance and life prediction methodology can be formulated analytically. Analytical models will be developed for new and innovative structural concepts that take advantage of cost-effective fabrication processes. Structural sizing procedures will be developed that provide cost-effective and structurally-efficient structural designs that satisfy all design constraints including any that are required to represent any restrictions associated by a cost-effective fabrication procedure.

Manufacturing Methods

The current challenge is to develop and verify manufacturing processes and methods that will yield both cost effective and reliable structures. Clearly, the D/MI approach will be required to achieve this goal. Numerous lessons have been learned in the last 20 years about the costs of fabricating and nondestructively inspecting composite structures for civil and military aircraft. An excellent discussion of cost estimating techniques and the influence of manufacturing processes and equipment on cost for composite structure was recently published.[33] There is no universally accepted method or process that will lower manufacturing costs and most likely a number of processes and equipment advances will be required to achieve cost-effective structures. Developments that are believed to be critical include: lower cost starting materials that do not add to downstream manufacturing cost, significantly more automation which would reduce labor cost and eliminate scrap due to human error, simple designs that exploit automated processes, elimination of the autoclave consolidation and assembly processes that require shimming. Specific processes and methods that are being investigated under the ACT program include: textile preforms, resin transfer molding, advanced fiber placement and braiding (see Fig. 21). The textile preform efforts include: through-the-thickness stitching of uniweave tape, plus two- and three-dimensional woven or braided structural shapes. The rate of depositing material, limits of applicability and quality of the advanced fiber placement fabrication method will be

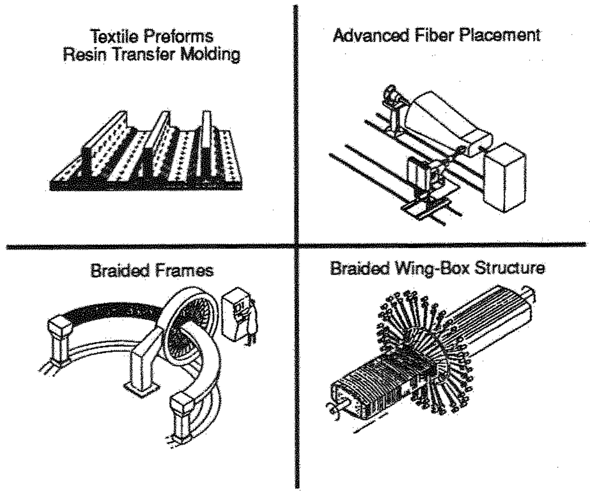

Fig. 21 Integrated material, design and manufacturing concepts.

assessed by building and testing representative wing and fuselage structure. Braiding will be evaluated for building structural elements such as frames, spars and longerons and for fabricating large components such as a wing box.

Technology Verification

Until the data base for composite structures reaches a level of maturity comparable to that for current metallic structures, fabrication and testing of full-scale components will be required to demonstrate performance and cost goals. Prior NASA contracts with three transport aircraft manufacturers to develop composite empennage components demonstrated that industry and government laboratories were not able to predict accurately failure or cost of the components.[4] In most instances the failure loads were lower than predicted and the costs were greater than predicted. Failure loads for the first three full-scale empennage components ranged from 67 to 92 percent of design ultimate load. Valuable lessons were learned from these tests and, through minor redesign, followed by fabrication and testing of additional subcomponents or a second full-scale component, the capability to support design ultimate load was demonstrated. Similar experiences, particularly with failure prediction, have been noted by other types of aircraft manufactures.[34] The foremost lesson is that secondary loads produce through-the-thickness forces which may lead to a serious weakness in an otherwise sound structure. Also, the brittle response of composites is such that the load redistribution around cutouts, etc., often achieved in metallic structures is not obtained (see Fig. 22).

Based on experience to date, and the realization that an aircraft manufacturer is risking the company assets on each new aircraft that is introduced, full scale testing will be required to convince the company officers that the critical technology and cost issues have been solved. In addition, the certification agency and the customer will most likely demand full-scale tests for the near future. Before the commitment to develop and build a composite

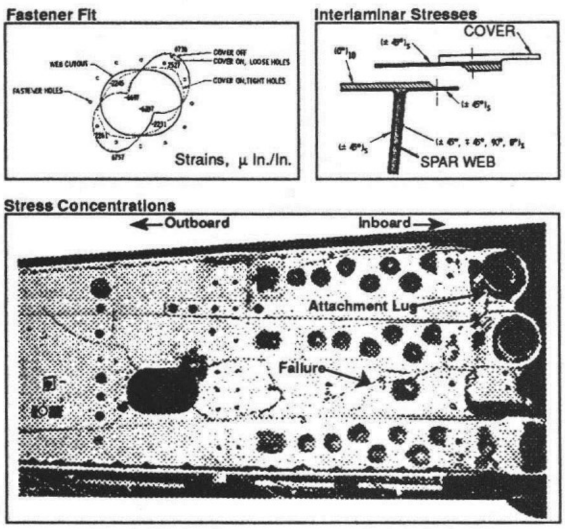

Fig. 22 Failure modes that must be included in composite designs.

primary airframe is made, full-scale fabrication and testing of a wing box, a fuselage barrel and possibly the wing-fuselage attachment component will be required to demonstrate that an "affordable" data base exists (see Fig. 23).

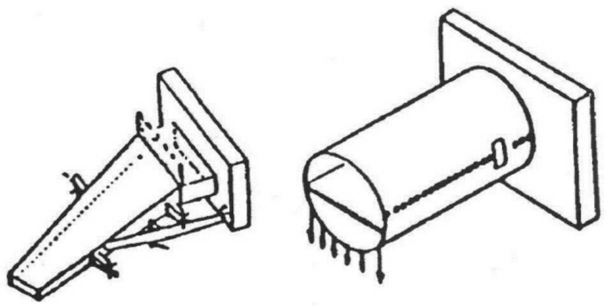

Fig. 23 Full-scale test components required for technology verification.

Summary

Composite secondary and empennage structures are in production on several transport aircraft. The weight reduction potential of composite structures is well documented. However, the cost to develop and produce composite structures remains the major barrier to increased application of this technology to transport aircraft. Materials, design concepts, structural mechanics methodology and manufacturing processes and equipment are under development or are emerging that are expected to lead to an integrated "affordable" data base. Technology verification for the next decade is expected to require fabrication and testing of full-scale wing-box and fuselage-section components before certification can occur and production commitments can be made.

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