JOURNAL OF AIRCRAFT
Vol. 46, No. 4, July–August 2009

Composite Materials Strength Determination Within the Current Certification Methodology for Aircraft Structures

Paolo Feraboli*
University of Washington, Seattle, Washington 98195-2400

DOI: 10.2514/1.41286

Although several composite failure criteria have been proposed over the years to predict the static strength of polymer composites, including micromechanical and first-ply failure theories, none of them has demonstrated an ability to predict the onset of damage across the wide range of load conditions and structural configurations. For that reason, current aerospace practice uses empirically derived laminate-based allowables to demonstrate compliance with regulatory requirements for static strength. Some examples of certification by analysis supported by test evidence as well as certification by test are given, and the fundamental differences are highlighted in the context of the building block philosophy. The lack of accepted failure criteria and general material-degradation models for the prediction of damage initiation and propagation provides the background for the current methodology, which relies either on testing alone or on analysis supported by extensive test evidence.

Introduction

IT IS difficult to summarize, under a unifying discussion, the certification approach for composite structures of the commercial aviation industry as a whole. Large commercial transport manufacturers have a much broader set of tools available to them for the design, analysis, and testing of composite structures than the other airframe manufacturers, such as General Aviation. The discussion that follows will focus on the certification approach specific to large commercial transports. Several design criteria exist for composite structures, which include design loads and static strength, durability and damage tolerance, crashworthiness, and discrete threats (Fig. 1). Within a large airplane corporation, a specialized set of skills is required for the people that perform analysis other than static strength. These include specialized test and finite element analysis methods that span from hail strike, bird strike, uncontained fan blade events, large notch damage propagation, and crashworthiness. Margin-of-safety (MOS) calculations, which are performed to size the vast majority of the airframe, are aimed at verifying the integrity of the structure under static strength requirements. These calculations are performed using a different set of tools, which are typically not commercially available but include internally developed semiempirical or closed-form solution tools and, to a limited extent, custom finite element models.

In the following discussion, current practices for evaluating the MOS for laminated polymer composite structures, and hence providing the foundation for static strength substantiation in the certification process, are reviewed. The importance of laminate-based allowable strengths for both unnotched and notched orthogonal laminates is reviewed, and the means by which allowables are integrated in the certification process for static strength analysis is discussed. The building block approach, which relies on analysis supported by test evidence, has served the commercial aviation industry for decades and currently provides, with few exceptions, the principal certification methodology for composite structures. A case study is discussed, featuring a three-dimensional fitting secondary structure, which emphasizes the differences between the processes of certification by test and certification by analysis.

The paper emphasizes the simplicity and effectiveness of the current certification methodologies for the static strength of composite structures. Although these approaches rely extensively on testing, whether at the basic coupon level, at several intermediate levels, or at the full-scale level, they ensure the highest probability of safety and success by minimizing risk. At the same time, they hinder the ability to introduce new materials in a timely fashion and tie airplane programs to previously specified material systems. The inability of the several failure criteria and damage models proposed in the research world to capture the whole spectrum of physical behaviors encountered in composite structures prevents them from permeating the design and certification process.

Lamina-Based Versus Laminate-Based Strength

Classical laminated-plate theory has been used successfully to calculate the elastic engineering constants of multidirectional composite laminates based on unidirectional lamina properties. The anisotropic elastic behavior, a characteristic trait of composite materials, is a relatively-well-understood phenomenon, and accepted tools are available to provide satisfactory analytical predictions of the elastic properties of composite laminates. On the other hand, the nonhomogenous nature of composite materials has been a far more challenging problem for the composites community. The development of physics-based predictive capabilities for assessing the postelastic behavior of composite materials has eluded the composites community for decades, and the root of the problem can be attributed to the nonhomogeneous, rather than the anisotropic, nature of these materials. Many failure theories have been proposed over the last four decades for predicting the strength of composites laminates. This can be achieved at the constituent, at the lamina, or at the laminate level [1] (Chapter 5).

At the constituent level, there is a potential to capture the true physical nature of damage by using micromechanics. However, these calculations require the knowledge of several physical quantities that cannot be easily measured using current test methodologies, such as the in situ strength of the cured matrix or the strength of the fiber–matrix interface. Furthermore, the inclusion of the effects of defects such as voids or fiber wrinkling (extrapolated from micrographic inspections, for example), the challenges arising from the stochastic nature of defect distribution, and the assessment of characteristic length scales make the task of determining the onset of failure even

Received 28 September 2008; revision received 12 February 2009; accepted for publication 25 February 2009.

*Assistant Professor, Department of Aeronautics and Astronautics, Box 352400 Guggenheim Hall; feraboli@u.washington.edu (Corresponding Author).

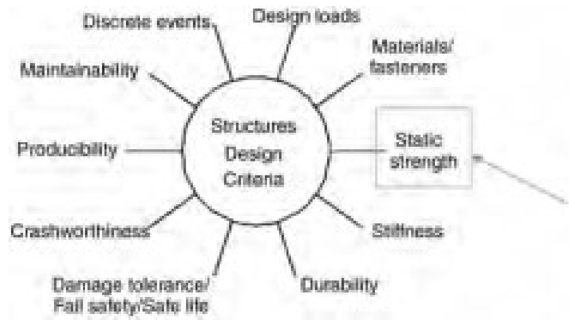

Fig. 1 Design drivers for composite structures, highlighting the focus of the paper on static strength requirements.

more difficult. Finally, the computational effort required for scaling these micromechanical calculations from a small representative volume element, typically employed for these analyses, to an actual structure can be overwhelming for today's supercomputers. For these reasons, *micromechanical approaches are not used for the certification of current aircraft structures*.

At the lamina level, failure criteria rely on experimental data obtained from unidirectional lamina or unidirectional laminate. These failure criteria may be grouped into two broad categories: mode-based and purely empirical. Mode-based criteria, such as Hashin's or maximum strain (Fig. 2, top), separately treat each identifiable physical failure mode, such as fiber-direction failure and matrix-dominated transverse failure. On the other hand, purely empirical criteria generally consist of a polynomial combination of the stress or strain components in a ply. Such criteria attempt to combine the effects of several different failure mechanisms into complex interaction formulations, such as Tsai–Wu. The average stresses in a given ply may be used to calculate the onset of damage, which is frequently called *first-ply failure*, and subsequent damages leading to laminate failure are part of the *progressive ply failure* methodology. When the first-ply failure is the result of fiber breakage, it is reasonable to consider that first-ply failure is equivalent to laminate failure. A different criterion exists when the first-ply failure results from matrix damage, in which it is reasonable to consider that the load-carrying capacity of the ply is still substantial.

It has been shown that these lamina-based failure criteria do not provide the necessary degree of generality and robustness to predict the behavior of multidirectional laminates for certification purposes. This is partly due to the fact that they neglect residual thermal stresses, which are difficult to measure and isolate, and because experience has shown that there are complex interactions between the laminas within a multidirectional laminate. In the analysis and certification methodology for large commercial transport aircraft, *lamina-based strength criteria (including the often-mentioned Tsai–Wu failure criterion), are currently not accepted*, and only laminate-based noninteractive criteria are employed for MOS calculations of composite structures.

At the laminate level, failure criteria rely on experimental data obtained from laminate families that encompass the entire design space for the laminate layups to be used in the actual airframe. A common practice in the aerospace industry is to use a modification of the maximum strain failure criterion (Fig. 2, bottom). This criterion compares the applied strains against the laminate-based allowable strains in each of the four orthogonal directions to calculate the MOS for each ply. The method appears to be the same as a lamina-based maximum strain failure criterion, but in the latter, the applied strain is compared in each ply with the strain-to-failure of the unidirectional lamina in tension, compression, and shear in the 0 and 90 deg directions. A critical assumption in this criterion is that the laminate behavior is fiber-dominated, meaning that the criterion is based only upon fiber strain allowables, for which fiber failure in any lamina is considered to be laminate ultimate failure. This condition is verified by ensuring that there are fibers in sufficient multiple directions such that strains in the matrix are limited by the presence of the fibers. This can be translated in a set of documented guidelines [1] (Chapter 5). The motivation for preferring fiber-dominated laminates is that for the fiber polymer composites typically in use on commercial transports, matrix damage has not been found to cause reductions in the static strength of laminates, particularly if the operating strain level has been restricted by the presence of bolt holes for damage tolerance and repairs, as well as other important structural details. Furthermore, it has been shown that the variability associated with matrix-dominated failures can easily be in the range of 11–18%, whereas for fiber-dominated failures, it is contained in 6–10%. The associated B-basis allowables are therefore much higher in the latter case.

To date, laminate design space in the commercial transport world is restricted to the four principal orthogonal orientations: 0, ± 45, and 90 deg. Allowable strain values for laminate-based strength criteria are generated from coupon tests in the fiber directions for each laminate family. These families are typically specified with the layup percentage notation, which describe the laminate by the amount of 0, ± 45, and 90 deg fibers in the layup, separated by a /. It is usually assumed that the percentage of $+45$ deg fibers is equal to the percentage of -45 deg fibers. At first, it may seem that confusion may arise by using this compact notation, but if laminate stacking-sequence design guidelines are followed, leading to fiber-dominated layups, the difference between layups with identical percentage notation but different explicit notation will not influence the in-plane behavior {Table 5.3.3.2 (b) in [1] (Chapter 5)}.

An example of typical analysis is given to explain the fundamentals of this approach. The layup $[+45/0_2/-45/0_2/90]_s$ conforms to all eight guidelines reported in [1] (Chapter 5), and its layup percentage notation can be written as (57/29/14). A typical analysis procedure would compare the applied strains with the allowable strain in each ply in the 0, ± 45, and 90 deg directions for the given layup (Fig. 3). This in turn means that three analysis runs are performed for each ply. In the first run, 0 deg direction, the allowable strain is selected for the (57/29/14) laminate. In the second run, 90 deg direction, the allowable strain is selected for the (14/29/57) laminate. In the third run, ± 45 deg direction, the allowable strain is identified for the (14.5/71/14.5) laminate. Determination of the allowable strains for these three different laminates or laminate families is done with the aid of carpet plots such as the one in

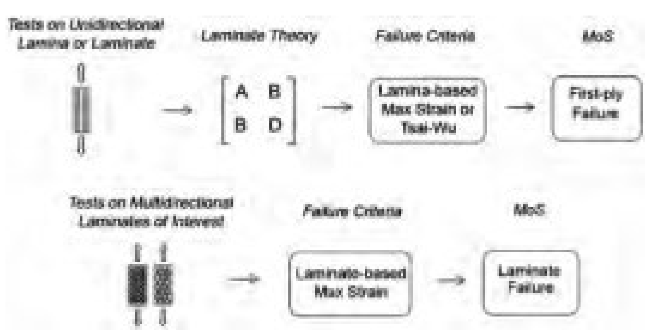

Fig. 2 Schematic explaining the maximum strain failure criterion applied at the lamina (top) vs the laminate level (bottom).

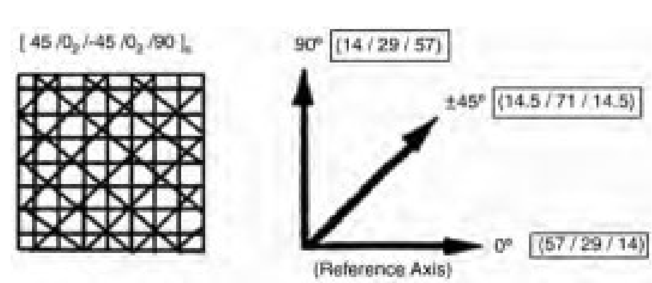

Fig. 3 Laminate layup varies for each of the four principal orthogonal directions employed for strength analysis.

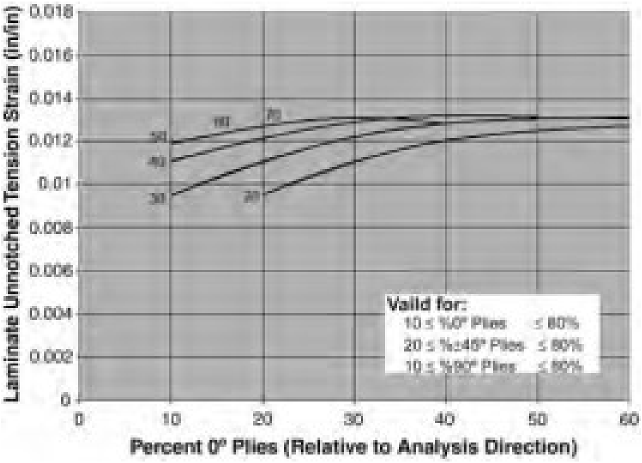

Fig. 4 Carpet plot showing B-basis allowable strains for unnotched tension at room temperature ambient (UNT RTA).

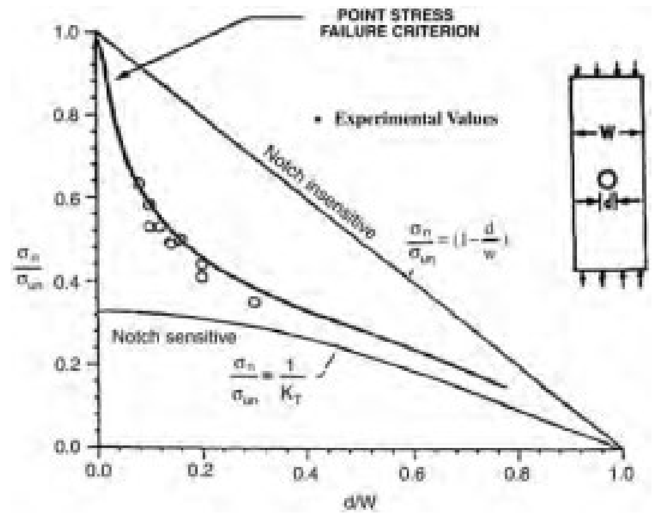

Fig. 5 Notch behavior of continuous-fiber, quasi-isotropic composite laminates compared against that of a purely notch-sensitive and a purely notch-insensitive material [1] (Chapters 5 and 6).

Figure 4. It is possible to observe that these three laminates have values of 0.0136, 0.0102, and 0.0125 in./in., respectively. Carpet plots are generated by testing thousands of coupons, using different layup configurations, as discussed in the allowables section.

It should be emphasized that these observations apply only to unnotched-laminate static strength calculation. However, in practice, unnotched strength values are never used to size any polymer composite structure. Because of the notch-sensitive, quasi-brittle nature of polymer composites, which will be discussed in the following sections, unnotched properties are too conservative and are not used for MOS calculations, because they do not account for the effect of notches and other defects.

Stress Concentrations

The presence of a hole or other discontinuity in a structure introduces local stress concentrations, which can result in localized failure. A balanced symmetric laminate may be regarded, for the purpose of structural analysis, as a homogeneous orthotropic plate. Well-known formulas exist for the elastic stress concentration factor K_t^∞ for an isolated hole in an infinitely thin orthotropic laminate plate subjected to uniaxial in-plane loading [1] (Chapters 5 and 6). The laminate layup influences both the magnitude and the shape of the stress variation near the hole.

When approaching failure, experience has shown that carbon-fiber-reinforced polymers (CFRPs) exhibit a behavior that is neither notch-insensitive, as in the case of metallic materials, nor purely notch-sensitive, as in the case of brittle materials (Fig. 5). Various matrix and fiber damage effects are expected to occur at the maximum stress locations. Certain combinations of localized damage that occurs at the tip of a notch (such as ply splitting, delaminations, and weak fiber failures) can enhance residual strength by effectively reducing the stress concentration. Among the methods that have been proposed to account for this reduction in the stress concentration, the *point stress theory* (or Whitney–Nuismer criterion) is the most commonly used in commercial aviation.

Such theory proposes that the elastic stress distribution for K_t^∞ be used, but that the stress concentration be evaluated at a specific distance d_0 from the edge of the hole. This distance is known as characteristic dimension and must be evaluated experimentally. Failure is postulated to occur when the stress at a distance d_0 ahead of the notch is equal to the unnotched-laminate strength (Fig. 6). This analysis method is restricted to laminates exhibiting a stacking sequence that is very close to that of a quasi-isotropic. Typical characteristic dimensions are a function of many variables and are not true material constants. Each laminate stacking sequence and loading case, within the same material type and form, require separate characteristic dimensions to be calculated by testing notched and unnotched specimens. For the same family of laminates as defined in the previous section, the characteristic dimension is calculated from uniaxial tension and compression tests on coupons with varying hole diameters and diameter-to-width ratios. Within the same family of laminates, different d_0 are calculated for open-hole tension (OHT), open-hole compression (OHC), filled-hole tension (FHT), and filled-hole compression (FHC). During each analysis run, the applied strain is compared against the respective allowable strain for each of the preceding cases at the appropriate d_0 distance from the notch tip and in each of the four principal directions (Fig. 7).

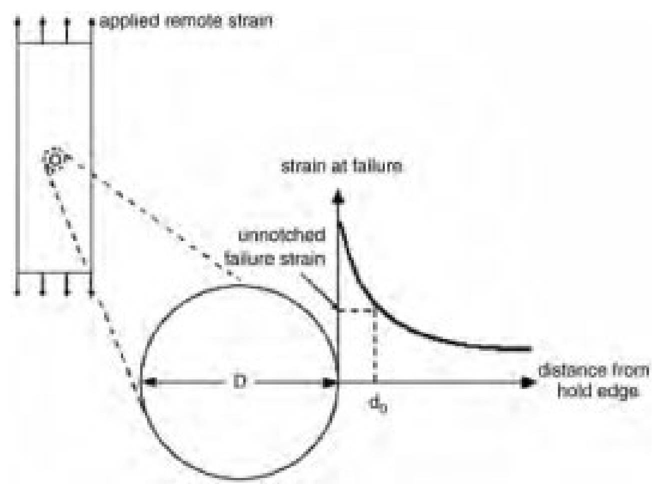

Fig. 6 Characteristic dimension d_0 for the point stress criterion.

Although purely semi-empirical, the advantage of this approach is that it relies on the same allowables-based methodology discussed in the previous section on unnotched-laminate strength for calculating the MOS. However, it imposes limitations on the selection of laminate families. For unnotched laminates, the main criterion for selecting a laminate stacking sequence is that it is fiber-dominated, leading to the laminate design guidelines contained in [1] (Chapter 5). For notched laminates, the design space is further restricted to a small area centered on the quasi-isotropic pattern in Fig. 8 [1] (Chapter 6).

Allowables

Allowables are a statistically significant material strength (although typically strain) against which an applied value can be compared in MOS calculations. The allowable values are minimum values established on a defined basis, which is an indication of assurance that the minimum property will be exceeded. A commonly

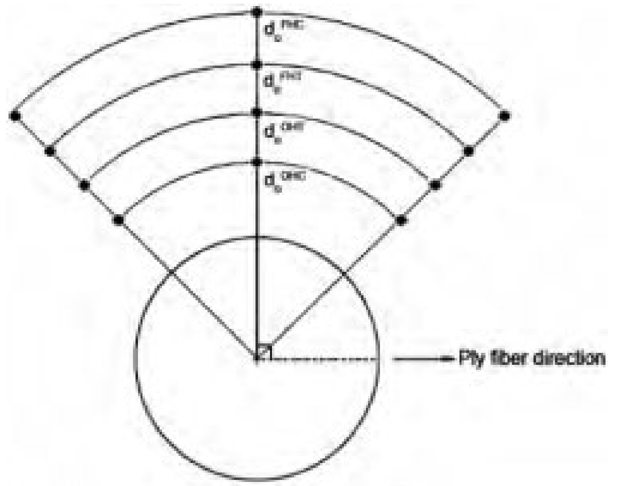

Fig. 7 Typical evaluation of the margin of safety at each characteristic dimension d_0 for the same laminate layup under different loading conditions.

used B-basis allowable imposes a 95% lower confidence bound on the upper 90% of a population of measurements. With a handful of exceptions, design values are normally given as B-allowables, not as average values (also known as typicals).

The effects of material and process variability, operating environments, and undetectable damage needs to be accounted during the analysis and hence needs to be included in the determination of the allowables (Fig. 9). Baseline material allowables are given for the room temperature ambient (RTA) conditions, typically 70°F (21°C). Knockdown factors are provided to account for service environment effects such as temperature and moisture [2] (Chapter 2). Elevated temperature wet (ETW) is typically performed at 160°F (71°C) and at moisture saturation, whereas cold temperature dry (CTD) is typically at −75°F (−60°C) and after moisture removal. Measured allowables need to be representative of the actual material and processing specifications used in service. In addition to material traits that are characteristic of the process, it is practice to establish acceptable manufacturing defects that may occur, such as foreign-object impact, embedded delamination, and surface or edge cuts. Because of the inability of current analysis methods to predict the onset of failure, the presence of preexisting damage during the certification process is assumed by inserting these embedded flaws or other forms of damage. Critical damage locations are determined based on the specific structural detail considering stress levels and exposure to likely damage threats. Damage scenarios considered need to be consistent with inspection and maintenance procedures employed during manufacture and in-service operations.

To that extent, it is important to emphasize that determining the size and location of an equivalent flaw is a difficult task. For example, it is now a well-accepted fact that foreign-object impact damage assumes the form of a pseudocircular or elliptical area during ultrasonic inspection of traditional aircraft laminates [1] (Chapter 7). This through-thickness projected damage area does not reflect the internal state of damage, which is known to be composed of complex "trees" of delaminations, longitudinal and transverse cracks, and fiber breakage, Fig. 10. Developing a damage metric that can be determined nondestructively but, at the same time, captures the actual damage state is therefore not a trivial task and is one that has eluded the composites community for decades [1] (Chapter 7) and [3,4].

In commercial aviation, baseline material allowables are multiplied by the appropriate correction factors, or knockdown factors, when calculating margins of safety for the actual design conditions. Unnotched allowables are primarily used as screening values to compare materials and in test correlation analyses, but are not used in design, as they do not account for manufacturing anomalies, process variations, holes, or damage. Typical designs are sized based on notched allowables, such as open- or filled-hole, and are generally evaluated in the worst combination of temperature and moisture. By the time a given material system is sufficiently mature to be employed in the design of a component, its allowable strength is a small fraction of the pristine strength, as declared by the material manufacturer (Fig. 9).

Certification Methodologies and the Building Block Approach

In the general case, certification of CFRP structural components can occur by testing or by analysis.

Certification by analysis is composed of a complex mix of testing and analysis to substantiate the structural performance and durability of composite components. This process is known as the building block approach (Figs. 11a and 11b) and leads to what is commonly referred to as "certification by analysis supported by test evidence" [1] (Chapter 4). Analysis techniques alone are not sufficiently

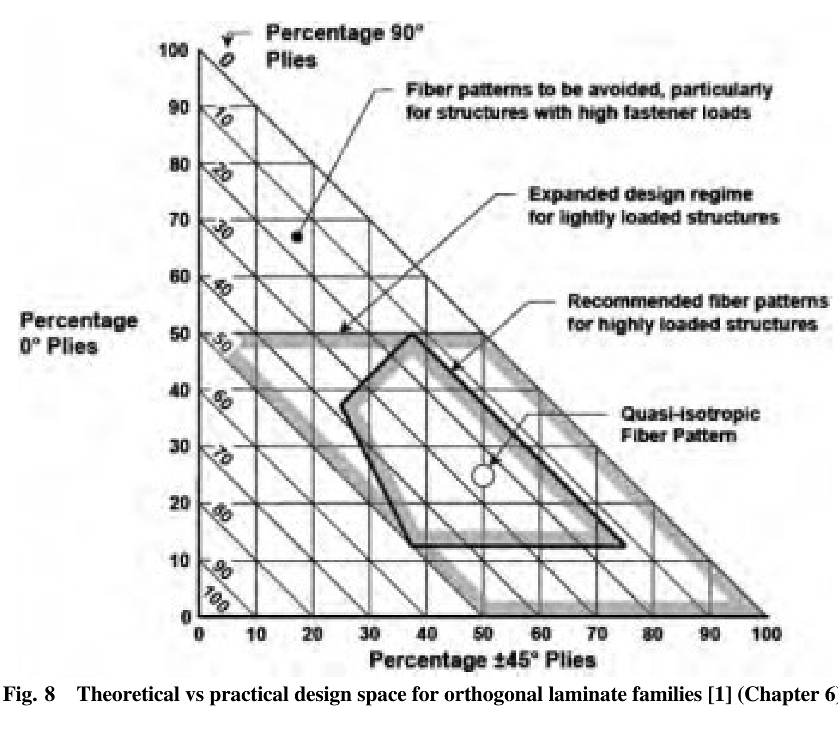

Fig. 8 Theoretical vs practical design space for orthogonal laminate families [1] (Chapter 6).

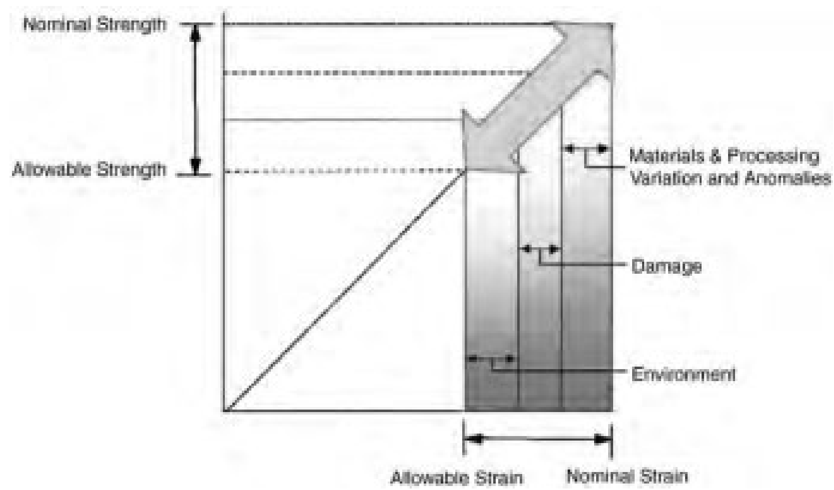

Fig. 9 Allowables account for processing anomalies, damage and environment knockdown factors.

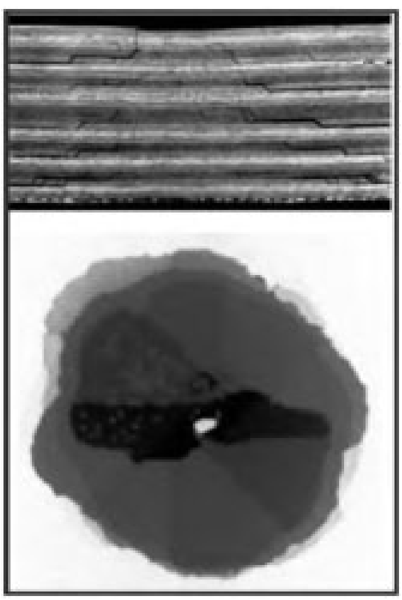

Fig. 10 Micrographic picture of the cross section of an impacted laminate, and associated ultrasonic image of projected damage area [3].

predictive to adequately predict results under every set of conditions. By combining testing and analysis, analytical predictions are verified by test, test plans are guided by analysis, and the cost of the overall effort is reduced, while the degree of confidence and safety is increased. This methodology relies on the development of material allowables, which in turn are used to feed the appropriate analysis methods for the calculation of the MOS (Fig. 12). For this reason, it is also sometimes referred to as *allowables-based* design or certification. The development of allowables is a process that usually requires several years of lead time, and its huge costs are only justified when a decision is made at the airplane-program level or even enterprise level to invest in a specific material system. For these reasons, it is often difficult to change the material system from one airplane program to the next, as it would require the generation of a full new set of allowables. For the design of primary structures, such as the general design of the fuselage and wing acreage, design and certification occurs only by means of analysis supported by test evidence. However, for the design and certification of secondary structures, such as fittings, certification may occur by testing alone.

Although testing alone can be prohibitively expensive because of the number of specimens needed to verify every geometry, loading, environment, and failure mode, situations may arise in which the time to develop a full set of material allowables for a new material system, or to calibrate the analysis methods for such material system used in a specific structural design detail, is not sufficient. In those situations, certification by test becomes an alternative to meet program deadlines. Certification by test is composed of tests performed to verify the suitability of the component to meet design requirements. No analysis methods are used in conjunction with this type of certification. The disadvantage of this approach is that it lacks any degree of generality, and a component composed of the same material but having different geometric or loading characteristics will have to undergo a complete experimental evaluation. For this reason, this methodology is also referred to as point-design or point-testing certification.

Although the concept of the building block approach is widely acknowledged in the composites industry, it is applied with varying degrees of rigor, details are far from universal, and not all building block approaches use the same number of complexity levels. This approach comprises analysis and associated tests at various levels of structural complexity, often beginning with small specimens and progressing through structural elements and details, subcomponents, components, and, finally, the complete full-scale product (Figs. 11a and 11b) [1] (Chapter 4) and [2] (Chapters 6 and 7). Each level builds on knowledge gained at previous, less complex, levels. Progressing upward on the pyramid, the specimen complexity increases, as well as the degree of specificity of the test, but the number of replicates decreases.

Previous publications by The Boeing Company for the 737, 767, and 777 vertical empennage and horizontal stabilizers have shown the extensive amount of testing required to support the development and certification of primary CFRP structures [5–9]. For such complex and vital components, all steps along the building block pyramid are populated by tests (Table 1). Material screening and qualification coupons, which constitute levels 1 and 2 in the building block of Fig. 11a, are a fundamental part of the design process, but are not discussed here as part of the certification process. Coupon and element tests constitute blocks 3 and 4 in Fig. 11a, and block 5 pertains to large subcomponents. The total number of tests performed at the subcomponent level (305) is much smaller than those performed at the coupon and element levels (8059). Usually, only two full-scale test articles are built and tested to validate the whole analysis process: one for static strength assessment and the other for durability and damage tolerance purposes [1] (Chapter 4), [2] (Chapters 6 and 7), and [5–12].

From Table 1, it can be seen that static laminate strength tests comprise nearly 30% of all coupon- and element-level tests for the Boeing 777 composite empennage. These data were used to generate the allowable-strength plots such as Fig. 4. Interlaminar strength tests comprise nearly 7%, whereas ply-level strength tests only comprise 2–3% of the overall test program. Tests conducted to assess the effect of temperature, moisture, and fatigue on unnotched laminates at the coupon level constitute altogether less than 10% of all subscale-level

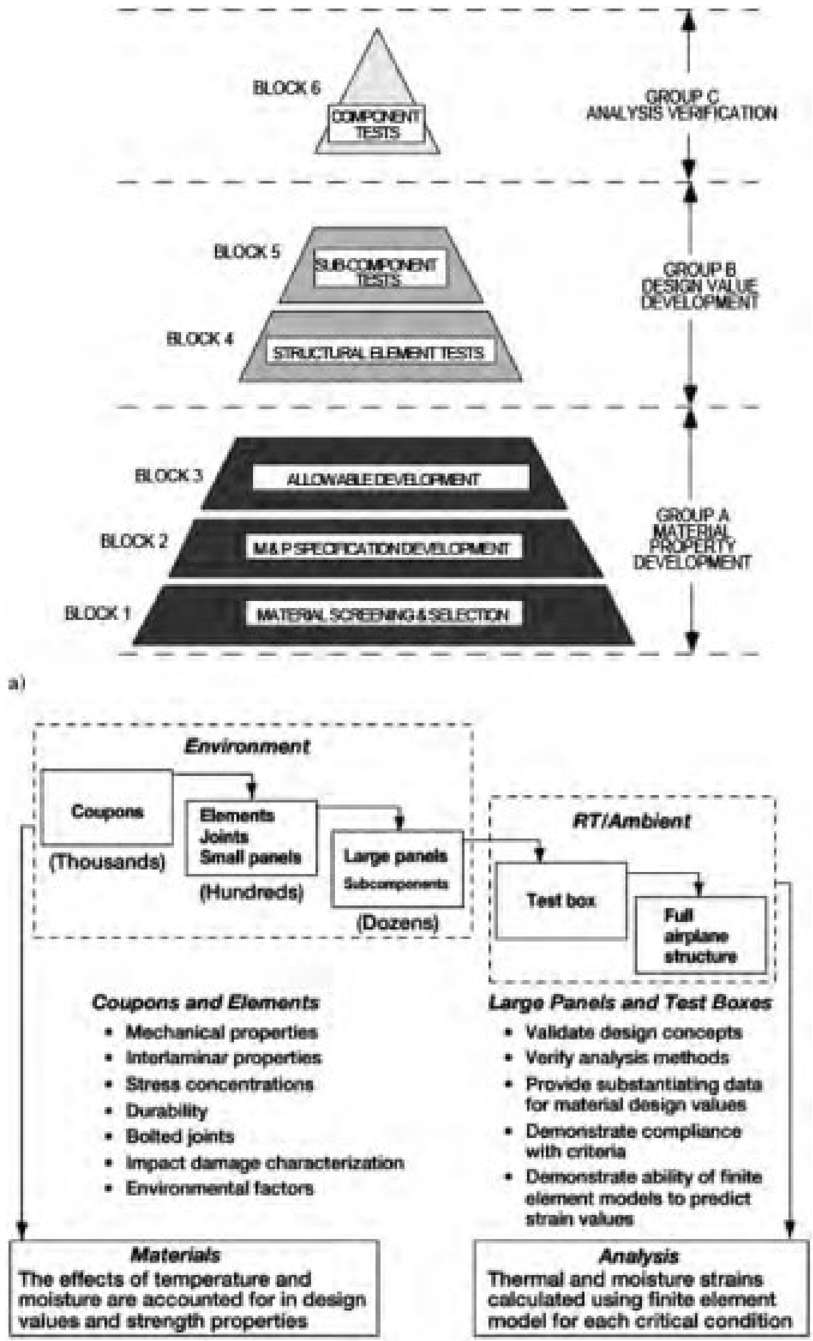

Fig. 11 Schematics of the building block approach [1] (Chapter 4).

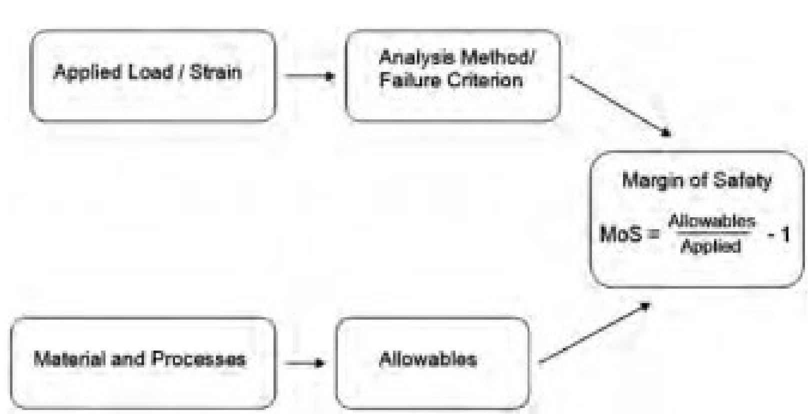

Fig. 12 Allowables and analysis methods are directly linked.

Table 1 Summary of tests for 777 empennage [1] (Chapter 4) and [7].

Test type	Number of tests
Coupon and element levels	
Ply properties	235
Long-term environmental exposure	200
Laminate strength	2334
Interlaminar strength	574
Radius details	184
Crippling	271
Stress concentrations	118
Effects of defects	494
Bolted joints	3025
Durability	385
Bonded repair	239
Total	8059
Subcomponent level	
Bolted joints (major splices)	110
Rib details	90
Spar chord crippling	50
Skin/stringer compression panels	26
Skin/stringer tension panels	4
Skin/stringer shear/compression	6
Skin/stringer repair panels	6
Skin splice panels	2
Stringer runouts	4
Spar shear beams	6
Total	305

tests. For long, slender elements such as stiffeners, crippling tests are also performed and comprise about 3% of the total. Radius details tests, such as curved beam bending and radius pull-off, comprise 2% of the tests. Bolted joints comprise 37% of coupon- and element-level tests and include the generation of the full bearing/bypass curve. This curve includes fundamental design values such as bearing strength, OHT, OHC, FHT, FHC, and high- and low-bypass configurations.

Tests at levels 3 and 4 of the building block are used to generate the data to feed the analysis methods to be used at the higher levels of the pyramid. The values generated at these levels of the building block can be employed for designing other structures in future airplane programs, provided that the same material and processing conditions, as well as similar geometric and design features are used. These include fastener types, stacking sequences, and curved details.

At the subcomponent level, tests are used almost exclusively to validate the analysis methods previously calibrated. The degree of specificity of the tests is such that future airplane programs will unlikely be able to use the same test data. In general, level 5 includes large 3- and 5-stringer panels to characterize skin/stringer interactions; major bolted-joint assemblies; and spar, stringer, and rib tests. These tests are used to minimize the risks associated with the last level of the pyramid (full-scale validation of the complete assembly): level 6. In general, a static test article as well as a fatigue test article are built and contain details of the certification procedure, including embedded artificial flaws, no-growth approach, limit-load strain surveys, and destruction at ultimate load [7].

Example Study: Certification of CFRP Fitting

The following section will provide a proposed example of certification of a secondary CFRP structure. The study is articulated in two sections. The first is certification purely by test, and the latter is certification by analysis supported by test evidence. The structure selected for this study is a three-dimensional root fitting that is used to transmit the loads from the skin and stringer to the side-of-body joint (Figs. 13 and 14). The component is subject to offaxis pull-off loads, which are a combination of tension or compression and bending. The part is composed of continuous carbon-fiber/epoxy tape, having a layup of $[+45/0_2/-45/0_2/90]_s$. Part thickness varies between a minimum of 0.080 in. (2 mm) and 0.150 in. (3.8 mm). Fastener sizes include both 0.250 in. (6.36 mm) and 0.375 in. (9.53 mm). This component is purposely selected because in the commercial aviation world, a more complex primary assembly, such as a flap or stabilizer or empennage, will likely be certified through a full-scale building block program such as the one described in the previous section.

Case A: Certification by Test

In this scenario, the full-scale part is built according to manufacturing specifications and tested in a condition representing the worst-case loading scenario. A possible test setup is shown in Fig. 15, and it shows the fitting mounted on an undeformable fixture, representing the side-of-body joint, rigidly fixed to the base of the test frame. The load is introduced via another fixture, mounted on the crosshead of the test frame, which represents the loading that the part would experience during service. The specification control document (SCD) for this part contains the load requirements that the part is expected to meet, and those are in turn generated from the loads model (or global model) of the airplane. The part is loaded to failure, usually a certain percentage over ultimate load requirements, and no attempt is made at understanding the stress or strain state in the part. No attempt is made at understanding the behavior of the material. Certification is achieved if the part exceeds the load requirement and if the failure mode is considered acceptable.

For this type of certification, the effects of materials and processing variations, environment, damage, manufacturing anomalies, and fatigue all have to be included at the part level. In order, these include the following:

1) For materials and processing variations, full-scale parts are manufactured using different fiber batches, different resin batches, and different prepreg batches. The effects of these parameters would be investigated by testing pristine parts, which means that they do not contain artificial defects, they would be tested at RTA, and would use nominal size fasteners.

2) For environmental effects, full-scale parts are tested inside an environmental chamber in which the suitable temperature and moisture conditions are reached. This includes ETW and CTD conditions. The conditioning chamber has to envelop the entire test fixture and setup and requires the design of a complex test procedure.

3) For the effect of manufacturing anomalies, full-scale parts are manufactured containing embedded flaws, such as Teflon inserts at the midplane of the radii and edge or surface cuts, representing flaws that may be present in the part as it comes out of the mold. These are placed in the most critical locations, as determined in the SCD. Testing is performed under the worst-case environmental scenario.

4) For the effect of damage, full-scale parts are inflicted impact damage, usually at a sufficient level of energy to create barely visible

Fig. 13 Skin/stringer-to-root fitting considered for this case study.

Fig. 14 Detail of the CFRP fitting considered for the case study.

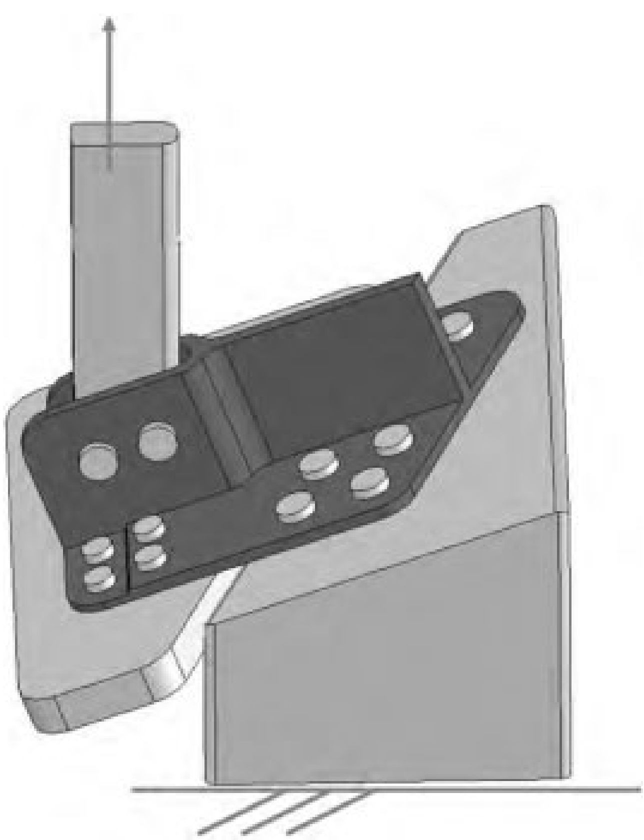

Fig. 15 CFRP fitting installed in the test fixture representing the loading scenario to be expected in service.

impact damage (BVID). This is typically achieved by impacting the part with a rigid hemispherical indenter until the damage is almost detectable, according to specific manufacturers criteria. Damage location is chosen to be the most critical for the given part, as determined by the SCD. More details are provided in MIL-17 [1] (Chapter 7). The goal is to simulate the type of damage that may be incurred during assembly or in service but may be undetected during inspection procedures. Testing is performed under the worst-case environmental scenario.

5) For the oversize fastener, full-scale parts are built and assembled using fasteners that are larger than the nominal size, representing the ability to tolerate assembly errors or repairs. Testing is performed under the worst-case environmental scenario.

6) For durability, full-scale parts are fatigued for the adequate number of cycles in the same setup as for the static test and then tested for residual strength in the same condition. Testing is performed under the worst-case environmental scenario.

If a slightly different part were to be manufactured using different geometric details, such as another fitting in Fig. 13, a different test setup would have to be built, and the entire test matrix would have to be repeated. It is easily seen how this approach often results in really high costs and is often even difficult to implement. However, if a certain degree of similarity were to be demonstrated, showing equivalency in some key tests could reduce the overall test matrix to a more acceptable level.

A representative test matrix, including several repetitions to assess consistency in the results, is shown in Fig. 16. It can be seen that for this type of certification, all tests are performed at the full-scale level: level 6 in the pyramid. In some cases, a modified test plan may be considered acceptable, which uses a limited set of tests at the coupon level to assess the temperature and moisture effects, which are the most difficult to conduct at the full-scale level. In that case, the environmental knockdown factors would be applied to the pristine test results at the part level, but this approach would require a certain degree of confidence in understanding the interactions between embedded flaws and damage with the accelerated environmental degradation. From this test matrix, it can be seen that point-testing, or certification by test, is a complex endeavor that often results in cost- and weight-ineffective designs.

Case B: Certification by Analysis

In this scenario, the generation of allowables is required, as well as the development of analysis methods. Unless a full set of allowables is already available from previous experiences, such as in the case of the 777 empennage, a new set will require to be developed. This may occur if the material system is completely new or sufficiently different from those for which allowables already exist. In the latter case, the allowables will be developed only to the extent needed for this particular application, which in turn requires the identification of the features that are critical design drivers. This translates into saying that for the part in Fig. 14 there is no benefit in generating allowables for crippling or for radius details, because the part does not contain such features.

Key characteristics of this part include the following:

1) The stacking sequence is $[+45/0_2/-45/0_2/90]_s$.
2) Thickness is 0.080 in. (2 mm) and 0.150 (3.8 mm).
3) Fastener sizes are 0.250 in. (6.35 mm) and 0.375 in. (9.53 mm).
4) Part will be subject to fatigue loads.

For this type of certification, the effects of materials and processing variations, environment, and fatigue will be characterized

Part	Pristine Articles RTA	Damaged Articles RTA	Damaged Articles CTD	Damaged Articles ETW	Damaged Articles Oversize Fastener s	Damaged Articles Fatigue and Residual Strength	TOTAL
3D fitting	30	6	6	6	6	6	60

Fig. 16 Part-level test matrix for CFRP-fitting certification by testing: 30 implies 6 specimens and 5 batches, and 6 implies 6 specimens and 1 batch.

at the coupon level. Coupons are manufactured using different fiber batches, different resin batches, and different prepreg batches. Tests are conducted on pristine, unnotched specimens. These are conducted at RTA, ETW, and CTD. Tests are performed in tension and compression. They are performed on both laminate thicknesses and for three different layups, including (57/29/14), (14/29/57), and (15/70/15), to account for the three principal orthogonal directions. Fatigue tests are also performed at the coupon level in a fashion similar to the static strength tests. To determine the characteristic dimension d_0 and to characterize bolted joints, coupon- and element-level tests are performed to build a fully populated bearing/bypass diagram. These tests include open- and filled-hole tension and compression, as well as bearing and bearing/bypass tests. Hole diameter, fastener size, and hole diameter-to-width ratio are also varied. Laminate layup is varied similarly to the case of unnotched-laminate strength. The objective of the laminate-level test database shown in Fig. 17 is to establish a statistically significant empirical database for unnotched and notched laminates for subsequent correlation of analytical models. For the example outlined in Fig. 17, it is apparent that an allowables-based certification involves over 1000 coupon-level tests to establish the minimum confidence in the critical characteristics for the part in consideration, whereas the point-testing certification requires only 60 full-scale fitting to be tested, as shown in Fig. 16. For one-off parts, point-testing is indeed a viable option.

The effect of damage (such as BVID) and manufacturing anomalies (such as embedded defects) is only included at the highest level of the pyramid. The final part is manufactured according to specifications and then assembled onto the full-scale wing assembly. Full-scale testing of this part alone is not likely to occur, and model verification usually takes place during testing of the full assembly. This may include a strain survey at limit load at regular intervals during the spectrum fatigue loading, followed by final destruction of the full-scale test article. In some cases, however, a full-scale test article of the fitting alone may be manufactured and tested in a fashion similar to that reported in the certification by test section (Fig. 15). This part would actually include embedded flaws and impact damage and would be tested at RTA to destruction, usually a certain percentage beyond ultimate load. The effects of fatigue and environment would be accounted for by applying the appropriate knockdown factors from the coupon-level tests.

Future Challenges

The previous discussion highlights the dependence of current certification methodologies of composite structures on extensive experimental data. Whether at the coupon and element levels or at the full-scale component level, both certification approaches show the need for extensive test data to assess the complex interactions between laminate thickness, layup, stress concentrations, temperature, moisture, materials, and processing characteristics.

Current analysis capabilities are not capable of predicting the onset of damage, neither in terms of location nor load, and special design criteria are established to include the presence of preexisting flaws in the most critical locations. This provision is also made necessary because of the limitations of current nondestructive inspection capabilities in conjunction with economics-driven inspection plans. Current analysis capabilities are also not capable of predicting the propagation of damage within the composite. Specialized tools such as the virtual crack closure technique exist for the prediction of the crack growth at a well-defined interlaminar plane or at the adhesive bond interface between stringer and skin. However, tools that can successfully predict the growth of damage in the plane of the laminate, such as in the case of a transverse notch in a tension specimen, are currently not applicable to the general case, and although semi-empirical approaches exist that employ Mar–Lin or damage-zone models, they have limited application and rely extensively on calibration testing at the subcomponent level. The use of cohesive elements to model the initiation of failure has been used in conjunction with stiffness degradation algorithms and show

Lay-up (%0, %45, %90)	Thickness [in.]	Tens. RTA	Compr. RTA	Tens. CTD	Compr. ETW	TOTALS
25/50/25	0.080	9	9	9	9	36
	0.150	9	9	9	9	36
57/29/14	0.080	9	9	9	9	36
	0.150	9	9	9	9	36
14/29/57	0.080	9	9	9	9	36
	0.150	9	9	9	9	36
14.5/71/14.5	0.080	9	9	9	9	36
	0.150	9	9	9	9	36
TOTALS		72	72	72	72	288

Lay-up (%0, %45, %90)	Thickness [in.]	Hole diameter [in.]	Specimen Width [in.]	W/D ratio	OHT RTA	OHC RTA	OHT CTD	OHC ETW	FHT RTA	FHC RTA	FHT CTD	FHC ETW	Bearing RTA	Bearing CTD	Bearing ETW	TOTALS
25/50/25	0.080	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.375	2.25	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.375	3.00	8.0	5	5	–	–	5	5	–	–	5	5	5	35
	0.150	0.375	1.50	4.0	5	5	–	–	5	5	–	–	5	5	5	35
	0.150	0.500	2.00	4.0	5	5	–	–	5	5	–	–	5	5	5	35
57/29/14	0.080	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.375	2.25	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.375	3.00	8.0	5	5	–	–	5	5	–	–	5	5	5	35
	0.150	0.375	1.50	4.0	5	5	–	–	5	5	–	–	5	5	5	35
	0.150	0.500	2.00	4.0	5	5	–	–	5	5	–	–	5	5	5	35
14/29/57	0.080	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.375	2.25	6.0	5	5	5	5	5	5	5	5	5	5	5	55
14.5/71/14.5	0.080	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.250	1.50	6.0	5	5	5	5	5	5	5	5	5	5	5	55
	0.150	0.375	2.25	6.0	5	5	5	5	5	5	5	5	5	5	5	55
TOTALS					90	90	60	60	90	90	60	60	90	90	90	870

Fig. 17 Allowables test matrix for CFRP-fitting certification by analysis: 5 implies 5 specimens and 1 batch.

promising results. However, progressive failure theories are highly dependent on large numbers of coupon, element, and even subcomponent test validation and hence are also of a highly-semi-empirical nature. Several failure theories are being developed and proposed worldwide [13], some based on improved interaction theories and others based on the physics at the microscopic level, but have not yet permeated the design, analysis, and certification process of composite structures in commercial aviation.

The need for truly predictive capabilities for damage initiation and propagation is greater now than ever, and virtual allowables generation based on a limited set of lamina-level tests represents the future for commercial aviation. Having successfully managed to capture the behavior of composites at the macroscopic (or laminate) level, the age-old question remains as to what level of detail will be required to develop physics-based predictive capabilities. Will the mesoscopic (or lamina) level provide sufficient information, or will it be necessary to investigate at the microscopic (or constituent) level, or even further down at the nanoscopic (or molecular) level?

Conclusions

The building block approach currently provides the foundation of the certification methodology for the commercial aviation industry. Although several novel failure criteria and strength reduction algorithms have been proposed in the research community, current predictive capabilities for damage initiation and propagation in composite materials are not adequate to be inserted in the broader certification process. Certification by analysis or certification by test are the current means by which airframe manufacturers can show compliance with regulatory requirements, and their application has been shown toward the certification of an example structural element.

Acknowledgments

The author would like to acknowledge his colleagues William M. Sheridan (Boeing Commercial Airplanes, Structural Methods and Allowables), Larry B. Ilcewicz (Federal Aviation Administration), Patrick B. Stickler (Boeing Commercial Airplanes, 787 Technology Integration), Alan G. Miller (Boeing Commercial Airplanes, 787 Technology Integration), John T. Quinlivan (Boeing Commercial Airplanes, Vice-President and General Manager, retired), Allen Fawcett (Boeing Commercial Airplanes, Designated Engineering Representative), and Keith T. Kedward (University of California, Santa Barbara) for providing the foundation and background for this work and for their commitment to composite safety and certification. Undergraduate student researcher Enrique Galgana is gratefully acknowledged for generating the graphics.

References

[1] *The Composite Materials Handbook—MIL 17*, Vol. 3, Rev. F, ASTM International, West Conshohocken, PA, 2002, Chaps. 4–7.

[2] *The Composite Materials Handbook—MIL 17*, Vol. 1, Rev. F, ASTM International, West Conshohocken, PA, 2002, Chaps. 2, 6, 7.

[3] Harris, C. E., Starnes, J. H., and Shuart, M. J., "Advanced Durability and Damage Tolerance Design and Analysis Methods for Composite Structures," NASA TM-2003-212420, June 2003.

[4] Kan, H. P., and Graves, M. J., "Damage Tolerance of Composites," Vol. 3, U.S. Air Force Wright Aeronautical Labs., AFWAL TR-87-3030, Dayton, OH, July 1988.

[5] McCarty, J. E., Johnson, R. W., and Wilson, D. R., "737 Graphite-Epoxy Horizontal Stabilizer Certification," AIAA Paper 82-0745, 1982.

[6] Schreiber, K. H., and Quinlivan, J. T., "The Boeing 777 Empennage," *9th International Conference for Composite Materials*, Madrid, Spain, 1993.

[7] Fawcett, A., Trostle, J., and Ward, S., "777 Empennage Certification Approach," *11th International Conference for Composite Materials, Melbourne*, Australia, 1997.

[8] Mohaghegh, M., "Validation and Certification of Aircraft Structures," 46th AIAA Structures, Dynamics and Materials Conference, Austin, TX, AIAA Paper 2005-2162, 2005.

[9] Ward, S., and Razi, H., "Principles for Achieving Damage Tolerant Primary Composite Aircraft Structures," *11th DoD/FAA/NASA Conference on Fibrous Composites in Structural Design*, Fort Worth, TX, Aug. 1996.

[10] Kan, H. P., Cordero, R., and Whitehead, R. S., "Advanced Certification Methodology for Composite Structures," U.S. Department of Transportation Rept. FAA/AR-96/111, Apr. 1997.

[11] Horton, R. E., and McCarty, J. E., *Damage Tolerance of Composites—Engineered Materials Handbook*, Vol. 1, ASM International, Materials Park, OH, 1989, pp. 259–267.

[12] Ilcewicz, L. B., and Murphy, B., "Safety and Certification Initiatives for Composite Airframe Structures," 46th AIAA Structures, Dynamics and Materials Conference, Austin, TX, AIAA Paper 2005-1877, 2005.

[13] Hinton, M. J., Kaddour, A. S., and Soden, P. D., "A Comparison of the Predictive Capabilities of Current Failure Theories for Composite Laminates, Judged Against Experimental Evidence," *Composites Science and Technology*, Vol. 62, Nos. 12–13, 2002, pp. 1725–1797. doi:10.1016/S0266-3538(02)00125-2