13 Airworthiness Considerations For Airframe Structures

13.1 Overview

Airworthiness is often thought to mean little more than how safe an aircraft is to fly, rather than a more appropriate and specific definition such as: "safe to meet the operational needs of the ... user."[1] Attaining satisfactory airworthiness requires that conflicting design characteristics be balanced to enhance the level of safety inherent in an aircraft type and is a key factor in design. In the past, aircraft development tended to focus only on developing adequate aerodynamic performance and structural strength, with the by-product hopefully being a "safe" aircraft. Contemporary design practices, however, place airworthiness considerations alongside performance, operational effectiveness, reliability, maintainability, and economy—all are important in determining the success of an in-service product.[2]

One of the earliest examples of airworthiness regulation was the 1916 publication of a pamphlet detailing design requirements for military aircraft during World War I by the Royal Aircraft Factory. And although this publication was little more than a list of strength requirements for early aircraft, it became the forerunner of the first British airworthiness regulations. In February 1940, the British Air Ministry formed the Joint Airworthiness Committee to collect all technical design instructions (including the derivations of the original 1916 document) into a single publication. This new document, rather than being prescriptive in its requirements, was to provide an unambiguous statement of the aim of each regulation rather than the detail of how the aim should be achieved. This approach of providing airworthiness regulations that state a required outcome, rather than the method by which the outcome is to be achieved, remains in practice today in both military[3,4] and commercial airworthiness authorities.[5]

Preceding chapters have addressed the technical concerns raised by the use of composite materials in aircraft structures. The critical step between these concerns and delivery of the final product is the demonstration of an acceptable safety performance. This issue has, in the case of metallic structures, been addressed by certification procedures developed as much by experience as science. An example of the combination of science and experience in the development of airworthiness regulation can be found in the derivation of the safety factor applied to full-scale fatigue test results by the U.S. Navy (USN).[6]

A purely analytical approach to the problem of reducing test-demonstrated-lives to in-service-lives with an acceptable low likelihood of failure would lead to a factor approaching three. But through examination of decades of aircraft usage data and a measure of engineering judgement, the USN rationalized the use, to a factor of two. Because advanced composite structures are relatively new to the aviation industry, the development of certification procedures must necessarily rely less on experience and more on the mechanics of the composite.

Initial attempts to certify composite structures simply adopted those requirements already existing for metals without recognizing the inherent differences between the two materials, even though these differences can significantly affect airworthiness considerations. For example, under static loading, composites typically exhibit linear elastic behavior to failure and are extremely sensitive to stress concentrations. In contrast, metals, with a few rare exceptions, exhibit plastic behavior above a yield stress and are not notch sensitive under static conditions. Another example of where significant differences exist between composite and metallic structures is in their damage tolerance under compressive loading. Advanced composite structures are much more sensitive to damage, and for this reason there has been an increased requirement on toughness in newly developed composite systems. Typically, certification guidelines deal with the issue of damage-tolerance in composites by requiring new designs to be based on the assumption that damage at the inspection threshold is initially present in the material.

Yet another critical difference includes damage growth due to fatigue. This often represents a critical design condition in metals, whereas composites typically show excellent resistance to such loading. The stress levels associated with design critical load cases in composite materials, such as compression in the presence of impact damage, have traditionally been low enough to ensure that the damage does not grow due to fatigue. Thus, designs in composite materials have typically been determined by static considerations rather than by fatigue.[2] As designers strive to fully use the specific strength and stiffness advantages of composites, the stress levels within components will increase, and fatigue issues must necessarily be given greater consideration in the airworthiness of future aircraft.

Perhaps the most critical difference between composites and metals is in their varying performance under different operational environments. The degradation of composite structures under certain environmental conditions has led to a number of standard certification approaches.[7] Essentially, it is necessary to establish the critical material properties after exposure to the extreme thermal and moisture environments to which the structure will be subjected. In addition, it must be demonstrated that there would be no degradation after exposure to chemicals that can be present (e.g., hydraulic fluid, fuel, and de-icing fluids).

Composites based on thermosetting matrices generally demonstrate significant sensitivity to absorbed moisture (the level of which in a thermosetting resin is proportional to atmospheric humidity) and temperature. Most engineering composites (typically epoxy based) can absorb up to one percent by weight of moisture in a normal aircraft-operational environment. The moisture softens the matrix resin, reducing those composite properties dependent on the resin, such as shear and compression stiffness and strength. This effect is particularly apparent at elevated temperatures, when the matrix is additionally softened. The simultaneous inclusion of environmental effects with structural testing of full-scale components has been demonstrated previously,[8] although such testing has generally been seen as prohibitively expensive. An alternative technique to account for the decreased performance of composites at elevated temperature and humidity is to increase the applied loads in structural tests conducted at room temperature and humidity. Unfortunately, this often leads to additional problems and typically prevents the clearance of hybrid metallic and composite constructions because the stress factoring necessary to evaluate the composite structure increases the risk of premature and unrepresentative failure in the metallic components.[9,10]

The U.S. Federal Aviation Authority (FAA) has developed a document, FAA AC 20-107A[7] that describes an acceptable means of demonstrating compliance to the FAA airworthiness certification requirements for composite structures. The document describes the additional considerations that must be given specifically to the certification of composite structures and includes topics such as:

- Effects of the operational environment on material properties and fabrication techniques
- Static strength with consideration of operational environments, repeated loading, impact damage, and material variability
- Fatigue (safe-life) and damage tolerance (fail-safe) evaluation

There are also several additional considerations including flutter, flammability, lightning protection, quality control, maintenance, and repair.

In AC 20-107A, the crucial issue of adequately considering environmental effects is addressed by allowing either full-scale testing under environmental conditions or testing through a "building-block" approach. The latter method is by far the most commonly used, because of its lower cost, but still involves extensive design development testing to 1) establish environmental and scatter knockdown (or load enhancement) factors for strength-critical failure modes, and 2) validate critical design features.

Full-scale testing will also be required, but under ambient temperature/dry conditions.

The major issues affecting the airworthiness of composite structures, are the static and fatigue strengths, effect of environmental exposure, damage tolerance, and flammability.

In this chapter, typical certification procedures for metallic airframe structures are briefly outlined. This is followed by a consideration of the significant differences between metals and composites for aircraft applications and the resulting requirement for the modified procedures. Methods of extracting design allowables from test data are then discussed and procedures for certificating composite structures are outlined.

13.2 Certification of Airframe Structures

The following fundamental requirements have been developed around the experiences of metallic airframes and remain the basis for the certification of composite airframe structures. These require that the structure (by test and/or analysis) demonstrate the following capabilities:

- **Static strength:**
  - Design limit load (DLL), no failure or unacceptable deformations. DLL is normally the maximum load anticipated to be placed on the structure in its lifetime.
  - Design ultimate load (DUL), no failure, although limited permanent deformation is acceptable; DUL = DLL × 1.5 (generally).
- **Fatigue strength:**
  - ***Safe life approach:*** No significant cracking that could lead to failure should occur in the life of the airframe. This approach was used in design of most of the metallic structure in the older fighter aircraft, and is still used for USN fighter aircraft, such as the F-18, and generally in helicopters.

    or
  - ***Fail-safe approaches:***
    > ***A. Alternate load path:*** The structure is damage tolerant in that cracking may occur but will not reduce strength below an acceptable level before being detected. This requirement is generally met by multi-load-path design where, should one load path fail, the remaining load paths can continue to provide the required level of residual strength until the damage is detected. This approach is generally used in the structure of large military transport and civil aircraft.

    or

    > ***B. Slow crack growth approach:*** The structure is damage tolerant in that cracking may occur, but cracks will grow slowly and will not cause failure for the full life of the structure *or* will not cause failure before detection by planned inspection (safety by inspection). This approach can be applied to single-load-path structure, where failure would be catastrophic. Damage-tolerant design for single-load-path structure is based on the assumed presence of flaws at critical locations. This is the design approach adopted for modern U.S. Air Force fighter aircraft, such as F-16.
- **Damage tolerance general requirement:**
  - The strength will not fall below an acceptable level (typically 1.2 × DLL) due to representative damage to the structure (e.g., caused by fatigue cracking, corrosion, or accidental mechanical contact) before being detected. Critical damage must be of a size that can be detected with a high degree of probability.
- **Durability/economic requirement:**
  - For the life of the airframe, damage requiring costly repairs will not occur, for example, due to fatigue or corrosion. Note that this is not a regulated

airworthiness requirement mandated by the airworthiness authority; however, it is often prescribed by the customer of military aircraft and is an important consideration in the choice of fleet by a civil airline.

Although composite structures are required to demonstrate the same level of safety as a metallic structure, the means of compliance has to take into account the differences in material behavior discussed in preceding chapters.

Design of the airframe for static strength first involves a detailed structural analysis, usually using a structural finite-element (F-E) model, and second involves a test program on specimens of increasing complexity, from simple coupons to structural elements and full-scale structures, as illustrated in Figure 13.1. Coupon and structural element tests are used to obtain material and structural allowables for design and must therefore interrogate all critical loading

- Coupons: $L \approx 0.1$m ; $N \approx$ (test number) 400
  - generate generic materials database
  - e.g., materials A and B allowables
- Structural Elements: $L \approx 0.2$m ; $N \approx 50$
  - generate generic feature data base
  - represent all potential failure modes
  - check calculation rules
- Structural Details: $L \approx 1$m ; $N \approx 10$
  - generate non-generic design values
  - check damage tolerance
- Sub-component: $L \approx 3$ m ; $N \approx 4$
  - check for unexpected failure modes, e.g., through-thickness
  - compare with detail and element tests
- Component: $L \approx 15$ m ; $N \approx 2$
  - check strain levels against failure strains
  - demonstrate airworthiness compliance
- Aircraft: $L \approx 30$ m ; $N \approx 1$
  - check strain levels
  - demonstrate airworthiness compliance

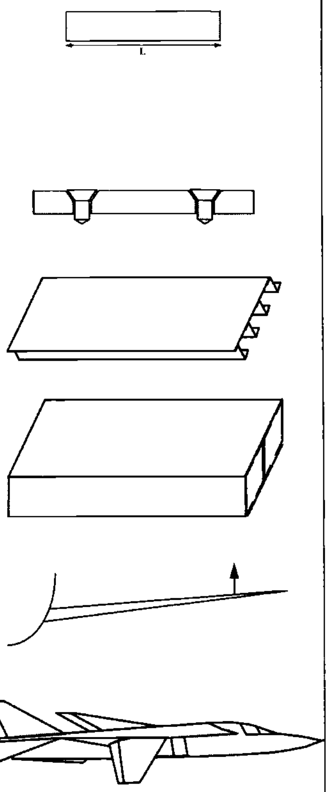

Fig. 13.1. Outline of the range of tests of increasing complexity used to certify an airframe structure made predominantly from composite material.

conditions and potential failure modes; the other tests are essentially for proof of structure. These tests are outlined in the following sections.

13.3 The Development of Design Allowables

Design allowables have to be established at the most critical environmental conditions. "Hot/wet" and "cold/dry" extremes are the most critical. Because coupons and structural details are generally small, there are no major difficulties in moisture-conditioning them. To ensure conservativeness, airworthiness authorities generally require full moisture-saturation at the highest anticipated operating temperature (typically of the order of a 1% weight increase). Special attention must be given to matrix-dominated failure modes, because these are most prone to degradation.

Sufficient coupon tests are conducted on the main laminate patterns to establish the allowable values for the critical temperature/moisture combinations in addition to the room temperature values. As well as establishing design allowables, these tests also provide knockdown factors (i.e., reduction factors). A comparison of, say, the room temperature/dry value and the hot/wet value of an allowable provides an environmental knockdown factor. Similarly, comparison of any mean value with its associated allowable value provides a variability knockdown factor.

A limited number of structural detail tests are made to confirm these allowables at the worst environmental conditions. These tests generally include open- and filled-hole tension and compression and bolt-bearing, including load bypass. Tests should also include details representative of areas of out-of-plane loading, such as stiffener run-outs.

The development of design allowables from coupons and structural details is a very important and costly component of the design and certification process. This is particularly the case for composites because of the high scatter in most mechanical properties.

13.3.1 Static Strength Allowables

Airframe static strength design is based on coupon and structural detail data that allows for statistical variation or scatter in strength. Two statistical levels of allowable values are possible:

- **A-allowable**—value achieved by 99% of the population at the 99% confidence level
- **B-allowable**—value achieved by 90% of the population at the 95% confidence level

To determine these allowables, the statistical model that best fits the property distribution is first determined. For metals, the distribution is generally normal or (for fatigue) log-normal. When dealing with composites, the first model evaluated is usually the two-parameter Weibull distribution (because it is a more physically realistic model for brittle materials) followed by the normal and then the log-normal.

An important economic aspect in testing is the estimation[11] of the minimum number of specimens that need to be tested to obtain acceptable allowables. This depends on the statistical parameters. Testing can be reduced significantly if the distribution parameters are already known.

The choice of which allowable to work with depends on the particular application. For materials with a low scatter, such as airframe alloys, the A-allowable strength is often used because this obviously offers the greatest margin of safety, but in cases in which scatter is large, this may impose too great a penalty on useable strength. The B-allowable strength is also appropriate for fail-safe or multiple-load-path design. For composites, the B-allowable is generally used because of the relatively high scatter on strength.

If service requirements can lead to a further reduction in static strength, the allowable static strength may be reduced by multiplying by knockdown factors. A similar procedure can be used to obtain allowables for structural details. As discussed in Chapter 8, several knockdown factors may be applied to coupon data to obtain the final design allowables.

This approach, inherently conservative, avoids exhaustive testing that would be needed to develop allowables for all conceivable conditions and designs. It is generally assumed that the scatter is unchanged from that obtained when deriving the allowables.

13.3.2 Fatigue Allowables

For development of the fatigue or durability allowables, composite coupons and structural elements are tested under constant amplitude and also under spectrum loading representative of expected service conditions. To simulate the most environmentally degraded condition, some testing will be conducted at elevated temperatures with the coupons appropriately moisturized.

The fatigue tests at constant amplitude cycling provide basic data for assessment of spectrum loading behavior and establish load discrimination levels—the load levels that can be neglected in the test spectrum. In addition, constant amplitude testing provides information on environmental effects.

The main feature of fatigue is that scatter is significantly greater than for most other mechanical properties, so many more of these time-consuming tests are required to obtain the allowable values. Fiber composites have a very flat S-N curve (stress versus number of cycles to failure) compared with metals because they are highly resistant to fatigue (see Chapter 8).

Significant damage growth generally occurs only at strain levels above 60% of the static strength. However, once growth commences, its progression is generally rapid, often catastrophic. Thus unlike metals, the slow growth option for composites is not considered possible, therefore damage growth data are rarely obtained.

A similar situation holds for bonded joints for which the rate of damage growth can also be rapid, and tests are generally made to establish the threshold for damage growth (see Chapter 9).

It is generally found that when the various knockdown factors are applied to the static strength obtained from the coupon data, the resulting design allowables are sufficient to provide an adequate margin for in-plane strength degradation under cyclic loading. However, this may not be sufficient to allow for strength degradation in some types of joint.

13.3.3 Influence of Damage on Allowables

For damage-tolerant design, the reduction in the strength allowables caused by impact damage, over the range of likely energy levels, must be quantified. As discussed in Chapter 8, the most severe loss in static strength occurs under compression loading; Chapter 7 describes the testing approach used for simple coupons.

Cyclic loading at representative strain levels can cause further loss in allowable static compression strength as well as limited damage growth and therefore will also require evaluation.

It is important in the setting of design allowables to assess the influence of impact damage on structural details such as ply run-outs and panels with stiffeners because the location of the damage is very important. When impact damage is located between stiffeners, failure can occur in two stages if the damage exceeds a threshold,[12] the first being rapid growth of the damage to the stiffener, followed somewhat later by failure of the stiffeners. Damage inflicted at a stiffener run-out can result in early loss of the stiffener.

13.4 Demonstration of Static Strength

The same broad program for a metal structure is generally followed for a composite structure. Attention has to be paid to the significant differences, as discussed in the preceding sections. A large number of coupon and element tests are performed, as previously discussed, to provide generic design data and to assess the effect of the operational environment and damage on the materials and details used in the complete structure. They are also used to check safe strain levels in sub-components, components, and the full-scale test.

Fewer sub-component level tests (see Fig. 13.1), are used to refine the predictions made for the complete structure from the results of the element and coupon testing and to validate critical design features. Certification usually culminates in one or two room-temperature ambient, full-scale static and fatigue tests. Environmental effects are accounted for in either the testing or the analysis of the test results.[13] This building-block approach to composite certification is commonly used for both military[14] and commercial aircraft and helicopters.[10,12,15–17]

13.4.1 Structural Detail and Sub-component Tests

Structural detail tests and sub-component tests are made to develop non-generic data related to the specific design. The structural details, elements, and sub-components selected for test will initially be based on the predictions of the finite element model. They are then statistically tested to failure under the most severe environmental conditions and the failure strain measured. These tests establish the mean values of ultimate strains in the environmentally degraded condition. (Because of the size constraints of sub-components, these are usually tested at ambient conditions and knockdown factors applied from coupon tests.) It is also important to note the region and nature of the failure and to ensure that the structural detail tests relevant to the region demonstrate the same failure mode. If this is not the case, then such tests must be repeated with appropriate adjustments to the loading or constraints. Then, assuming that the scatter in the structural element and sub-component tests are the same as those in the detail tests, application of the variability knockdown factor to the mean values determined gives allowable values for the full-scale structure in the environmentally degraded condition.

An important economic issue is the time required to develop a representative moisture distribution. Depending on the thickness of the composite structure, this may take from several weeks to several months.

13.4.2 Full-Scale Tests

The full-scale test is very important to inter-rogate the effect of secondary loading caused by out-of-plane loading. Such loads arise from eccentricities, stiffness changes, discontinuities, and local buckling, which may not be fully predicted or eliminated in design nor represented by the structural detail specimen. In addition, it is also important to validate the F-E model to ensure that internal loading of the structure occurs as predicted.

The full-scale article, which may be a wing, fuselage, or full aircraft, is generally tested in the room-temperature/dry condition. The main difference between this test on a composite and a metal aircraft is that, for the composite, the structure is much more extensively strain-gauged. This test largely serves to validate the finite element model. If the strain gauge results show regions of high strain in areas where no sub-components were tested, then it is necessary that such testing be performed. Then, concentrating on the ultimate load test, the measured strains at 150% DLL are compared with the knocked down design allowables as established by the coupon and structural element tests. If the measured strain exceeds the allowable value, failure is deemed to have occurred, and some redesign is necessary. Although there are uncertainties at various stages in the above test, the general approach seems reasonable. It can be seen that, for composite aircraft, virtually all development testing (on small and large specimens) becomes an integral part of the airworthiness certification.

However, it should be pointed out that the above is not the only approach to demonstrating static strength. One alternative sometimes proposed is to carry out the test of the full-scale article under ambient conditions (as above), but with the applied loads increased to allow for environmental effects. (The amount of the load increase is determined from specimen tests, much as already described.) Another alternative, of course, is to accept the need for a full-scale environmental test.

13.4.3 Proof Tests

In the past, because of certification or non-destructive inspection (NDI) concerns, some airworthiness authorities have required that every production composite component be given a proof test, generally to a load slightly in excess of the design limit load. In such cases, the components are given a thorough NDI both before and after test to check for damage that may be caused by the proof test. Because confidence in the material properties and analysis methods has improved, this approach is no longer usually insisted upon.

13.5 Demonstration of Fatigue Strength

The situation with regard to demonstrating a satisfactory fatigue performance for composite aircraft structure is far from clear. However, at least one requirement is to check for through-thickness failure modes, caused, for example, by unexpected through-thickness stresses.

The full-scale fatigue tests that have been carried out on current aircraft containing composites have generally been the same as would have been used for an all-metal aircraft, in other words, a test to N lifetimes (where N may be 2, 4, etc.) in a normal environment. (Of course, such aircraft are mainly metal.) Again, the prospect of doing a fatigue test on a full-scale aircraft with the moisture/temperature environment fully represented is daunting. One method for wings with integral fuel tanks is to fill the tanks with continually heated water; however, this convenience is not available for most other structures. Also, sufficient data are not available on the state of fatigue of large composite structures to provide any real basis for selecting a scatter factor. The full-scale test in a normal environment will certainly continue to be performed to verify the metal structure, and it may sometimes also serve to reveal unsuspected problems with the composite structures. However, it seems that the main verification of the fatigue performance of the composite structure will be based on structural detail and sub-component and possibly component testing in an appropriately humid environment, with the temperature cycling (especially the thermal spikes) accurately represented.

The apparent large scatter in composite fatigue life would suggest the need for testing composite structures for excessively long periods (e.g., 30 or more lifetimes) to satisfactorily account for the variability knockdown factor. The main approach to reduce excessive testing periods is load enhancement as described in Ref. 18. By suitable elevation of loads, it is possible to allow the test to be conducted for only one lifetime. If this is too severe on other parts of the structure, the application of somewhat reduced loads can allow the tests to be conducted to only two or three lifetimes. An important issue in mixed composite and metal structure is that the metal components may need to be strengthened (beyond that required in the actual airframe) to allow adequate testing of the composite components under enhanced loads.

It should not, however, be inferred from the above, that the fatigue of composites is necessarily a cause for major concern; it is more a matter of there being difficulties in establishing a convenient test demonstration. A more detailed discussion of the fatigue issue is given in Chapter 8.

13.6 Demonstration of Damage Tolerance

Damage tolerance is generally evaluated at all levels from the coupon to the full-scale article. Damage can be deliberately inflicted, for example, using a portable impact tester in critical regions, such as over fasteners. Damage can also include saw cuts and, in the case of sub-components, disbonds and delamination incurred during manufacture.

Demonstration of damage tolerance on full-scale articles involves residual strength checks usually conducted after several lifetimes of fatigue cycling have been performed on an undamaged article. Damage as described above is then inflicted on the article in the critical areas and limit-load-testing performed. Often, at this stage, the fatigue testing is continued for one or more further lifetimes of fatigue cycling to observe any damage growth. Provided that failure has not occurred, the next step is to repair the damage and continue fatigue cycling for a further one or two lifetimes. Finally, the article is tested to failure. A satisfactory, if quite conservative, result would be to achieve ultimate load in this test.

13.7 Assessment of the Impact Damage Threat

For damage tolerance design and evaluation, it is important to assess the threat of impact damage in relation to the airframe structure. Clearly, vertical surfaces and surfaces that are high on the aircraft are less prone to damage than horizontal surfaces. The top horizontal surface will be more prone to damage from dropped tools and hailstones, whereas the bottom surface will be more prone to damage from runway stones and burst tires.

Ref. 12 describes a comprehensive statistical study to assess impact threat. The data used are based on measurement of impact dents in U.S. military aircraft, including F-4, F-111, A-1, and F-18. The measured dent depth is converted, through calibration, into the equivalent impact energy. The threat is then quantified in terms of Weibull distribution, predicting the probability of occurrence of impacts over the range of energy levels. This information could be used to plan the level and location of damage on a test article.

It is of interest to note that this approach can be used, independent of the test, to assess the probable residual strength ("structural reliability") for the various zones in the aircraft. The structural reliability is obtained by integrating the product of $P(E)$, probability of occurrence of energy level E with $p(e)$, the probability of survival of the structure at strain e at that energy level.

References

[1]Meekoms, K. J., *The Origin and Evolution of the Design Requirements for British Military Aircraft*, RAE publication, June 1983.

[2]Cardrick, A. W., and Curtis, P. T., *Certification of Composite Structures for Military Aircraft*, D/RAE(F)M&S/5/15/1, 13 Oct 1991, presented at Aerotech '92 at the NEC, 14–17 January 1992.

[3]U.K. Ministry of Defence, *Design and Airworthiness Requirements for Service Aircraft, Vol. 1: Aeroplanes*, DEF STAN 00-970, Directorate of Standardisation, U.K., May 1988, pp. 2512.

[4]U.S. Department of Defense, *Aircraft Structures; General Specifications For AFGS-87221A*, 8 June 1990.

[5]Federal Aviation Administration, *Federal Aviation Regulations, Part 25, Airworthiness Standards: Transport Category Airplanes*, U.S. Department of Transportation, Oct., 1994.

[6]Elby, D., Hoffman, P., Hoffman, M., and Polakovics, D., *Managing Safety: The Mathematical Basis and Origins of the Navy's Factor of Safety*, Naval Aviation Structural Integrity and Aging Aircraft Conference, 3–5 Feb. 1998.

[7]Federal Aviation Administration, *Composite Aircraft Structure*, Advisory Circular AC 20-107A, 25 April 1984 and companion document by the JAA, ACJ 25.603, *Composite Aircraft Structure Acceptable Means of Compliance*, 1986.

[8]Ripley, E. L., and Cardrick, A. W., *The UK Approach to the Certification of Composite Components for Military Aeroplanes and Helicopters*, AGARD Report No. 660, April 1977.

[9]Perry, F. S., "Harrier II: A Comparison of U.S. and U.K. Approaches to Fatigue Clearance," AGARD Meeting on An Assessment of Fatigue Damage and Crack Growth Prediction Techniques, Sept., 1993.

[10]Dutton, S. E., and Lofland, R. A., "Certification of Structural Composite Components Used on the MD900 Helicopter," *Proceedings of ICCM-11*, 14–18 July 1997.

[11]Park, W. J., "On Estimating Sample Size for Testing Composite Materials," *Journal of Composite Materials*, Vol. 13, 1979, pp 219–212.

[12]Khan, H. P., Cordero, R., and Whitehead, R. S., *Advanced Certification Methodology for Composite Structures*, DOT/FAA/AR-96/111, Federal Aviation Administration, 1997.

[13]Whitehead, R. S., Kan, H. P., Cordero, R., and Saether, E. S., *Certification Testing Methodology for Composite Structure: Vol. II: Methodology Development*, NADC-87042-60, Naval Air Development Center, Oct., 1986.
[14]Weinberger, R. A., Somoroff, A. R., and Riley, B. L., *U.S. Navy Certification of Composite Wings for the F-18 and the Advanced Harrier Aircraft*, AGARD Report No. 660, April 1977.
[15]McCarty, J. E., Johnson, R. W., and Wilson, D. R., "737 Graphite-Epoxy Horizontal Stabilizer Certification," AIAA 82-0745, 1982.
[16]Fawcett, A., Trostle, J., and Ward, S., "777 Empennage Certification Approach," *Proceedings of ICCM-11*, 14–18 July 1997.
[17]Whitehead, R. S., *Certification of Primary Composite Aircraft Structures*, 14th Symposium of the International Conference on Aeronautical Fatigue (ICAF), June 1987.
[18]Lameris, J., *The Use of Load Enhancement Factors in the Certification of Composite Aircraft Structures*, NLR TP 90068U, National Aerospace Laboratory, 1990.