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ONRL Report

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Workshop Proceedings on Composite Aircraft Certification and Airworthiness

Dennis R. Sadowski

6 October 1988

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U.S. Office of Naval Research, London

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19. ABSTRACT (Continue on reverse if necessary and identify by block number) This report contains a summary of the workshop, a list of attendees, a prepaper entitled "Some Areas for Discussion Suggested by RAE," and the presentations on Composite Aircraft Structures, Civil Aviation Concerns, Impact Damage, RAE Composite Certification, and the Effect of Observed Climatic Conditions on the Moisture Equilibrium Level of Fibre-Reinforced Plastics. <i>Keywords:</i>			
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Contents

Foreword	1
Workshop Summary	1
List of Attendees	2
Prepaper: Some areas for Discussion Suggested by RAE	5
Opening Remarks Mr. Tom Hess	11
Composite Aircraft Certification Mr. K.B. Sanger	13
Certification of Primary Composite Aircraft Structures . . . Mr. R.S Whitehead . . .	19
Civil Aviation Concerns Dr. John Bristow	53
Impact Damage Professor G. Davies	59
RAE Composite Certification Dr. A.W. Cardrick	63
The Effect of Observed Climatic Conditions on the Moisture Equilibrium Level of Fibre- Reinforced Plastics T.A. Collings	69

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Workshop Proceedings on Composite Aircraft Certification and Airworthiness

Foreword

The Office of Naval Research Branch Office, London (ONRL) sponsored a 1-day informal Workshop on Composite Aircraft Certification and Airworthiness on 16 July 1987. The workshop was held at the ONRL Conference Room at 223 Old Marylebone Road, London NW1. It preceded the Sixth International Conference on Composite Materials/Second European Conference on Composite Materials, which began on 20 July 1987.

The purpose of this workshop was to discuss the issues and the philosophy concerning the testing and certification of aircraft structures constructed from a combination of advanced composite materials and conventional metals. The increasing use of composite materials in new aircraft made this an important and timely topic. Composite materials possess excellent fatigue resistance but considerable scatter in fatigue properties. This together with their susceptibility to property degradation in hot/wet conditions raises several issues with regard to how full-scale airframes should be certified for service use.

Representation was from the US and several European countries. Government, industry, and university specialists attended. A summary of the workshop discussions follows in this document. The six presentations and discussions were intended to be informal. Notes precede each presentation where only vugraphs were provided.

Any questions regarding this workshop may be directed to either of the following:

- Mr. Thomas E. Hess, Advanced Structures Technology Branch, Code 6043, Naval Air Development Center, Warminster, PA 18974-5000, Tel: 215-441-1463
- Dr. A.W. Cardrick, Materials and Structures Dept, X32 Bldg, Royal Aircraft Establishment, Farnborough Hants GU14 6TD, Tel: (0252) 24461 X5026

Additional copies can be obtained by contacting:

- CDR Dennis R. Sadowski, USN Office of Naval Research Branch Office, London, London NW1 5TH, Tel: (01) 409-4413.

Workshop Summary

This workshop provided a good forum for discussing a number of issues associated with the certification and airworthiness of new aircraft which will be constructed of

a combination of metals and advanced composites. The following topics were discussed:

- Certification procedures
- Tolerance to impact
- Fatigue
- Moisture effects
- Failure criteria
- Dwell at load in static tests.

Certification Procedures. Basic approaches were presented by the US (K. Sanger, R. Whitehead) and the UK (A. Cardrick) that rely on both small- and full-scale specimen tests – the so-called building block approach. There was no general agreement on the importance of fatigue to composites or the need to do full-scale fatigue tests. The current US Navy practice of doing full-scale fatigue tests to two lifetimes of a severe spectrum was questioned with regard to how that compares to the use of an average use spectrum for a greater number of lifetimes.

An interesting point made by Whitehead was that there never has been a "hot-spot" in a composite structure that wasn't seen in room-temperature static testing. This may indicate that a quick room-temperature dry static test is a valuable early design tool, even though the structure may be more critical in the elevated-temperature wet condition. Other opinions expressed were that we need agreement on a realistic impact threat (low velocity) and we need verified analytical methods for predicting damage and residual strength. G. Davies raised the question of what failure criteria should be used – stress or fracture. He also pointed out that three-dimensional analytical methods are available and that design engineers may need to become proficient in using them.

Tolerance to Impact. Discussions on this topic were by Davies and Cardrick. It was suggested that if the impact threat could be standardized, there would be more opportunity for international pooling of data. Cardrick presented two impactor configurations as candidates. Although Davies found APC-2 thermoplastic not too good above 20-m/sec impacts, Whitehead reported that his results show it far superior to epoxy for low-velocity impact and hydraulic ram – i.e., less damage and more residual strength.

Fatigue. Cardrick was the principle speaker on this issue. He presented an approach for a quick-look assessment of whether fatigue is more or less critical than static loading. It would be fair to say that there was not general agreement on the fatigue issue and, in particular, whether it does or does not have to be a concern for composites.

Coordination for this report, introduction/summary, and introductory notes by CDR Dennis R. Sadowski. CDR Sadowski, USN, is ONRL's Aerospace Systems Officer.

Moisture. This issue was discussed by T. Collings. This is another issue for which there is not universal agreement on criteria or worst-case environment. It is, however, a definite concern in the design process.

Failure Criteria. Mr. R. Potter, who addressed this topic, said that we need better failure criteria – an observation affirmed by Hess, who pointed out that we need a better definition of failure. Potter asked whether anyone has ever predicted a failure location, failure mode, and failure load.

Dwell at Load in Static Tests. Dr. Cardrick spoke on this issue. The concern is what happens when loads are held for a matter of minutes. Indications are that strains aren't affected, but, the question is, what happens for structures which are in a degraded condition?

List of Attendees

Dr. John Bristow
Civil Aviation Authority
UK

Mr. R. Cansdale
Royal Aircraft Establishment
UK

Dr. A. Cardrick
Royal Aircraft Establishment
UK

Mr. T. Collings
Royal Aircraft Establishment
UK

Professor Glyn Davies
Imperial College
UK

Mr. J.B. DeJong
National Aerospace Laboratory
The Netherlands

Dr. E. Demuts
Air Force Aeronautical Lab
US

Sqn Ldr C.A. Elkins
Royal Air Force
UK

Mr. J. Fray
British Aerospace
UK

Herr Ambrose Gollner
IABG-TFS
West Germany

Herr Dr. A. Habel
BWB-ML 2-51
West Germany

Mr. Allen Hall
Westland Helicopters
UK

LTC J. Hansen
EOARD
US

Dr. W. T. Hart
National Aerospace Laboratory
The Netherlands

Mr. Thomas Hess
Naval Air Development Center
US

Dr. Lars Jarfall
SAAB-Scania Aircraft
Sweden

Dr. David James
Civil Aviation Authority
UK

Mr. R. Kite
Royal Aircraft Establishment
UK

Mr. C. Luffman
Airship Industries
UK

Dr. J. Moon
Royal Aircraft Establishment
UK

Wg Cdr P. Perry
Royal Air Force
UK

Mr. R. Potter
Royal Aircraft Establishment
UK

Mr. C. Robinson
British Aerospace
UK

Commander Dennis Sadowski
Office of Naval Research
US

Mr. Michael Sancho
Service Technique Des Programmes Aeronautiques
France

Mr. K. Sanger
McDonnell Aircraft Co
US

Mr. Alex Segal
Israel Aircraft Industries
Israel

Mr. Kieth Fitz-Simons
Westland Helicopters
UK

Dr. D. Stone
Royal Aircraft Establishment
UK

Mr. J. Stott
Airship Industries
UK

Mr. Westwood
Airship Industries
UK

Mr. R. Whitehead
Northrop Aircraft Corporation
US

Dr. N. Wilson
Royal Aircraft Establishment
UK

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SOME AREAS FOR DISCUSSION SUGGESTED BY RAE

1 TOLERANCE TO IMPACT

1.1 Design and certification criteria

Aircraft structures are subject to impacts from a variety of sources and so reasonable robustness is needed in order to avoid the need for appreciable maintenance and repair. The impact conditions vary widely and may be quantified in terms of the hardness and sharpness of the impactor, the incident velocity and the incident energy. The effect of the impact on the structure depends upon the hardness of its surface, its stiffness and, if the energy cannot be absorbed elastically, its capacity to absorb energy by yield and/or disruption.

In general, any impact damage that is significant enough to affect the static strength of a metal structure will be evident and any damage that might not be evident but could affect fatigue will be found before any appreciable loss of strength has occurred. For fibre composite parts, however, some types of impact can produce appreciable damage of which there is no trace at the surface. Notably, a blow from a blunt object could produce delamination of the plies within the thickness of a carbon fibre composite part. In any subsequent application of a compressive load, the fibres would then be locally unsupported and if the laminate were split at the mid thickness the local compressive strength would be reduced to only about one quarter of its original value. In general, for the first generation of composite structures the design strains were low enough for non-evident impact-induced delaminations not to propagate under normal service loads. However, the use of higher allowable strains may lower the threshold at which damage propagates.

RAE have proposed that two types of impactor should be specified to represent the extremes of 'sharp' and 'blunt' conditions that can be expected to occur in service. It is proposed that for each type of impactor the structure should satisfy two criteria:

- a. It must be sufficiently robust to avoid the need for appreciable maintenance and repair after the majority of impacts that can be expected to occur in service, and
- b. It must exhibit clearly-evident damage after impacts of higher severity or, if any damage is not clearly evident, the structure must suffer no associated loss of integrity at the time of impact or later in the service life.

For each type of impactor, the energy and velocity levels needed to represent the majority of service impacts must be specified according to the zone of the structure and the expected source of damage. RAE are currently conducting experiments on composite and metal structures using two tentative standard impactors. The results should be available shortly and will guide the levels which must be specified for design. Multiple impacts, such as those arising from hailstones, must also be considered.

1.2 Dynamic analysis aspects of design

In designing to meet requirements of the type envisaged above it will be necessary to predict the form and extent of impact damage in various types of structure.

It is believed that the principal parameters to be considered are as follows:

- a. Prediction of the nature and extent of damage in composite structures:
 - i Structural form
 - ii Elastic energy absorption
 - iii Paint schemes (on visibility)
 - iv EMC protection schemes
 - v Airstream
 - vi Loading
- b. Prediction of residual strength:
 - i Load re-distribution
 - ii Damage propagation/arrest in complex structures

The prediction of structural response to impact damage might be achieved by the following approach.

- a. Analysis of the dynamic response of the structure to impact, treating it as an elastic problem involving homogeneous anisotropic plates and beams and calculating the local forces and bending moments which result. (It would be assumed for this analysis that there would be no coupling between the dynamic behaviour of the structure and the generation of localised damage.)
- b. Prediction, from the local forces and bending moments derived in (a) above, of the form and extent of damage. Initially this would probably have to be done by comparison with experimentally derived data. The effects of, say, paint schemes, EMC protection schemes, airstream and loading could be included at this stage.
- c. Prediction of residual strength of the damaged structure, allowing for phenomena such as load re-distribution by using locally reduced strength and stiffness properties in the analysis.

Clearly, as yet, such an approach would be too complex for design but, if shown to give adequately reliable results, it would permit the parametric investigation of structural forms from which design guidelines could be generated, an investigation which would be prohibitively expensive to carry out experimentally.

It is considered that the first stage in the successful development of the above approach is the development of the dynamic analysis and its validation using specific full-scale structures.

2 MOISTURE UPTAKE OF RESIN-MATRIX COMPOSITES

2.1 Design and certification criteria

The process of moisture adsorption and desorption in resin matrices is characterised by cyclic changes in the surface layers, which are in keeping with daily or even hourly changes in the ambient moisture and temperature, and by relatively small long-term fluctuations in the moisture content of the deeper layers. The ingress of moisture causes an increase in volume, some softening and a reduction in the glass transition temperature. Thus, in service, relatively thin composite panels that are exposed to cycles of moisture and sunlight will experience short term changes

in mechanical properties that can be correlated with the changes in environment. Thicker panels, or those sheltered from direct exposure, will experience less marked changes in properties governed more by seasonal variations, whilst thick sections buried deep in the structure will exhibit a gradual drift down in properties until the moisture content reaches an equilibrium governed by the typical relative humidity characterising many years of operation in the part of the world in which the aircraft is based.

Typically, military aircraft spend 90% or so of their time on the ground. In consequence, a wing panel on an aircraft operated in winter from one of the wetter areas of Europe, without the benefit of hangar storage, could reach an equilibrium moisture content corresponding to a relative humidity of 85% or so. Panels of less than 1 mm thick could conceivably reach an equilibrium corresponding to 95% relative humidity in the same period. Consequently, for the design and certification of military aircraft that will be operated under these conditions it is necessary to base the design of most parts on an equilibrium moisture content corresponding to 85% relative humidity, and to consider higher and lower values for thinner and thicker parts of the structure respectively.

2.2 Moisture management for testing purposes

When attempting to represent the effects of in-service environmental exposure in the airworthiness structural testing of fibre reinforced composite components it is important that the problems and the limitations associated with moisture conditioning and accelerated ageing are both understood and allowed for.

The natural process of moisture absorption in resin matrices is normally very slow, and this makes it very difficult to reach an adequate degree of degradation in a structural test element in a reasonable time. Various techniques have been suggested for accelerated conditioning and recent studies at RAE show how these might be improved.

Additionally, during the accelerated conditioning of a structure it is necessary to have some measure of the moisture uptake and its distribution through the thickness. Normally this is achieved by the use of travellers which are weighed at frequent and regular time intervals. Recent work, however, has shown that several important features need to be included and certain dimensional limits need to be imposed, before a traveller can be accepted as an adequate model of the moisture content of the item it is intended to represent.

3 FATIGUE - DO WE HAVE A PROBLEM ?

There is a strong body of opinion that if a composite structure passes a static test (even if it is undegraded) then there will be no fatigue problems. However, experience with certain types of bolted joints, bonded joints and ply drop-offs indicates that these will need to be sized to fatigue allowables if they are to have a satisfactory safe life under wing-bending spectra typical of advanced combat aeroplanes. Early consideration must be given to the characterisation of

a. the constant amplitude fatigue performance of such structural features, including environmental effects if these are significant, and

b. the fatigue performance ($\Sigma \frac{n}{N}$) and scatter under realistic spectrum loading, again including environmental effects if these are significant at the strain levels envisaged for design.

New UK requirements for the fatigue certification of the primary structure of military aircraft are about to be published (see JAC paper 1076). The main aims of the requirements are to provide a good safe life under the anticipated usage and to provide tolerance to damage caused by increases in the severity of loading with the changes in usage occurring during many years of service. In general, those details which are sensitive to increases in spectrum severity, and are difficult or uneconomic to inspect and modify or replace, must have their stresses reduced. Exceptionally, however, if the economic and operational consequences of inspection and modification or replacement are acceptable then an inspection-dependent approach may be followed.

In principle, composite patches could be used to reduce the stresses in fatigue-sensitive regions of structural details and so produce an increased safe life (giving greater resistance to increases in spectrum severity) and an increased inspection period for inspection-dependent details. In practice, however, there remain appreciable uncertainties regarding the consistency with which patches can be applied (both in manufacture and in service) and their long-term sensitivity to the cumulative effects of cycles of moisture, temperature and mechanical loading. These uncertainties must be faced in the certification of fatigue-sensitive items of composite structure and there is no technical reason why composite patches should not be substantiated in the same way.

However, the testing needed to establish an improved safe life or a longer inspection period for a composite patch would be both complex and expensive - at least initially - in order to obtain sufficient knowledge of the scatter and distribution of life to enable the performance of the weakest patches to be estimated. In the region of at least 15 tests would be needed under the actual service loading and any significant environmental cycles to measure the scatter alone, and at least four times this number if the distribution could not be assumed to be approximately log-normal. If the actual service loading and any significant environmental cycles could not be applied, then further testing would be needed under constant amplitude loading (say 15 specimens minimum possibly with environmental conditioning) in order to calibrate Miner's rule and so allow the test results to be adjusted with some confidence to the actual service conditions. Once an appreciable amount of data had been obtained, however, the amount of testing needed would be markedly reduced and the use of composite patches to enhance safe life or inspection periods would become more attractive.

By contrast, the use of 'NDE-transparent' composite patches to slow down the growth of inspectable cracks, whilst retaining the 'unpatched' safe inspection period, is not inhibited by the need to provide such extensive data. The question in these circumstances is largely one of economics rather than safety. It may well be cost-effective to apply a composite patch in the knowledge that it is only 'likely' to retard crack growth. The only certification restriction in this case is the need to show that the effect of the patch (in its application and subsequent action) is in no way detrimental - not a difficult task in most circumstances.

Thus, the general requirements governing the fatigue certification of structural details will specify the stringent conditions that must be satisfied if a patch is to be used in support of an enhanced safe life or an increased inspection period. There are no such restrictions on the use of 'NDE-transparent' composite patches to retard the growth of inspectable cracks provided the 'unpatched' safe inspection periods are retained and acceptable evidence is provided to show that neither the patching process nor the presence of the patch have any detrimental effect on crack growth.

5 EFFECTS OF PERIODS OF DWELL UNDER LOAD DURING STATIC TESTS ON COMPOSITE STRUCTURES

It is anticipated that those details of composite structure which exhibit resin-dependent failure modes could exhibit sensitivity to dwell under load during static tests under degraded conditions.

An exploratory research programme is planned to assess whether there is likely to be a problem under the strain levels envisaged for design.

6 ON THE USE OF 'B' VALUES FOR STATIC STRENGTH CERTIFICATION

With the introduction of advanced composite materials came the realisation that conventional airframe static tests alone could no longer be relied upon to reveal shortcomings in static design. In an undegraded test, those details with low environmental degradation have low reserves of strength and fail before the higher reserves needed in details with higher degradation can be demonstrated. For combat aircraft, at least, it is impractical to test a complete airframe under the most adverse combinations of moisture content and temperature and so increased reliance must be placed on showing that allowable values of stress, or strain, are not exceeded.

During the course of the interim update of AvP 970, some four years ago, the opportunity was taken to harmonise the method of deriving allowable stresses for composite parts, castings and forgings. It was found that the established test factors used for castings (Chapter 406) and composites (Chapter 408) corresponded closely to 'B' values, whilst higher allowables were permitted for forgings. By contrast, wrought materials, which had traditionally been sized to minimum specification ('S') values had changed to a dual standard (with its origin in Mil Specs and civil requirements) permitting 'B' values (generally higher than 'S' values) to be used except in single-load-path items, where 'A' values (generally similar to 'S' values) were required. It is not clear why the number of load paths was considered to have any bearing on static design. The load previously carried by a failed path could not conceivably be accommodated by the remaining path(s) unless the failure occurred at a low load due to weakening caused by fatigue or corrosion. Fatigue is the subject of separate design requirements and cannot be addressed in an effective way in static design. Corrosion, also is treated separately and it is in the corrosion requirements that any necessary reserves of static strength should be addressed. Neither fatigue allowables nor susceptibility to corrosion bear any logical relationship to variability in static strength which is the sole parameter governing the separation between 'A' and 'B' values.

Thus it was found that:

- a. the use of 'B' values was in effect already established for castings and composites (higher values having been used for forgings),
- b. Mil Specs and civil requirements permitted the use of 'B' values except for single-load-path items, and
- c. the case for using 'A' values for single-load-path items rested on providing extra protection against fatigue and/or corrosion, both of which are the subject of separate requirements and should not be addressed in static design.

In consequence, 'B' values were specified in Chapter 200 of Def Stan 00-970 for static strength certification and no special requirements were made for single-load-path items.

It must be said that the much larger amount of testing needed to establish an 'A' value, and the penalty in allowable stress, both mitigate against the use of 'A' values for structural details. To obtain an 'A' value using order statistics (ranking values in order) is said to require 296 test results, whereas just 30 are sufficient for a 'B' value (even smaller samples are permissible using Table 1 of Chapter 200). The penalty in allowable stress for wrought materials is of the order of 5%, whereas it rises to nearer 15% for composites. It is sometimes argued that 'B' values cause an unacceptable risk of static failure, but it is surprising to find that the risk of failure at 90% of the design ultimate load (factor of safety 1.5) is generally between 1 in 100 and 1 in 2000 according to the variability of the item in question.

Notes on Opening Remarks by Mr. Tom Hess

"The importance of establishing an approach to certification and airworthiness of composite aircraft."

US Navy Interest:

- Aircraft with composites material are now in service in increasing numbers
- AV-8B – 6-percent composite (Vugraph 1)
- F/A-18 – 10-percent composite (Vugraph 2) but almost 50 percent of external surfaces are composite material
- V-22 – over 50-percent by weight projected to be of composite material.

The traditional US Navy certification is:

- Build two aircraft

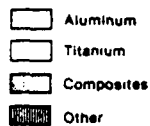
- Test one statically
- Test one to two lifetimes of fatigue.

The traditional certification is questionable for composite because:

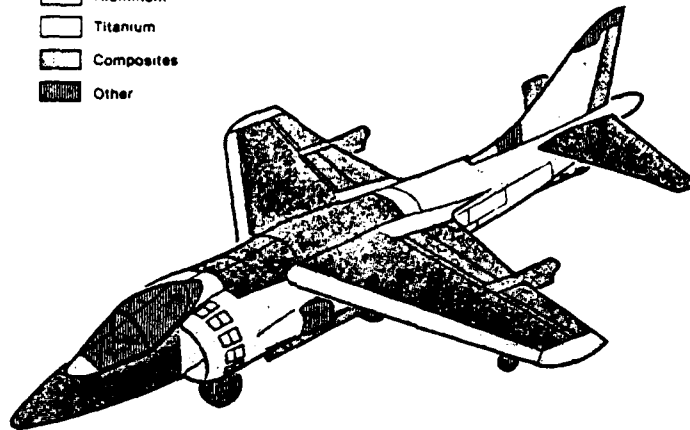
- Great tolerance
- Great scatter of properties

Issues:

- Fatigue life may require testing to 100 lifetime for confidence, however this is not economically practicable.
- Elevated moisture/temperature conditions of composites and effects on static testing need to be examined.
- Hybrid aircraft must be tested under these conditions.
- The US Navy started two programs to address an approach to composite certification.
- The US Air Force work in damage tolerance is worthwhile.



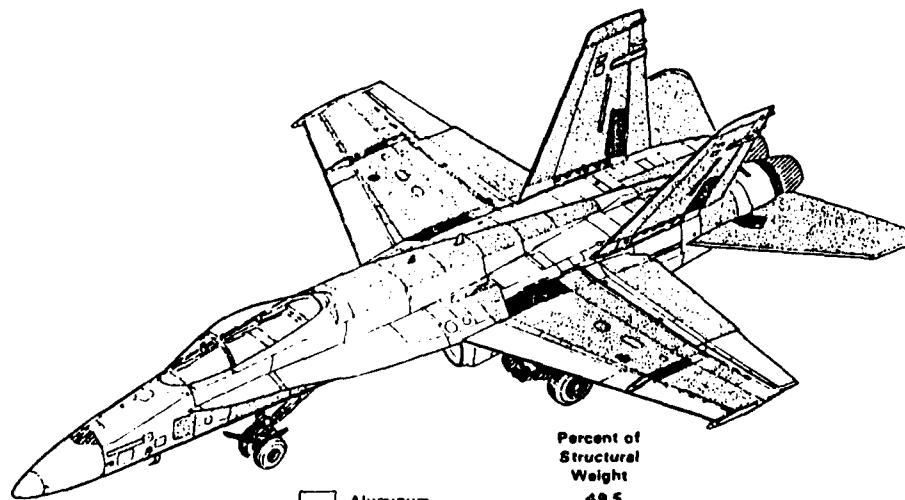
Vugraph 1








AV-8B COMPOSITE APPLICATIONS

F/A-18A MATERIALS DISTRIBUTION

Vugraph 2



	Percent of Structural Weight
 Aluminum	49.5
 Steel	16.8
 Titanium	12.9
 Graphite/Epoxy	9.9
 Other	10.9
	100.0

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Notes of presentation on "Composite Aircraft Certification" by Mr. K.B. Sanger:

- Mac Air looked at approaches to evaluating static strength and fatigue based on the Navy program and then developed a testing methodology.
- The advent of resin matrix composite caused a formal recognition of the importance and dependence of the certification procedure on the performance of the coupon design and element design allowables test. Upon performance of these tests the contractor tested specimens of larger size and complexity and related these results back to the design allowables properties. This is the general approach used today.
- The motive behind the building block approach is to account for the sensitivity exhibited by composites to the environment, and also to account for large data scatter.
- The graph of epoxy tension compression strength relative to temperature shows greatest loss under hot humid conditions.
- Composites in use are almost insensitive to fatigue, probably due to stress/strain levels of operation. With reference to F/A-18 and AV-8, McDonnell Aircraft Company is operating around 20-40 ksi gross stress which shows significant life-cycles. The main concern is with the evidence of fatigue scatter.
- An approach for certification which requires conditioning of the test article is not very practical from an economic standpoint nor to the limits of the program schedule.

An alternative approach for a structure that is primarily composites is to apply a single factor on the load to compensative for environment sensitivity. However, for a mixed metal and composite this may not be possible due to the differences of sensitivity; the McDonnell approach was to correlate measured strength during static tests with environmental allowables properties.

"COMPOSITE AIRCRAFT CERTIFICATION"

K.B. SANGER

MCDONNELL AIRCRAFT COMPANY

STATE-OF-THE-ART

- EVOLUTION OF THE CERTIFICATION PROCESS
 - BUILDING BLOCK APPROACH
- U.S. NAVY PROGRAM, "CERTIFICATION TESTING METHODOLOGY FOR "COMPOSITE STRUCTURES" (MAY 1984 - JANUARY 1986)
 - MIXED COMPOSITE/METAL STRUCTURE
 - STATIC STRENGTH APPROACHES
 - FATIGUE APPROACHES
- NEEDS
 - TRANSLAMINAR (OUT-OF-PLANE) ANALYSIS
 - LVID REQUIREMENTS

Vugraph 1

CERTIFICATION OF METAL STRUCTURE

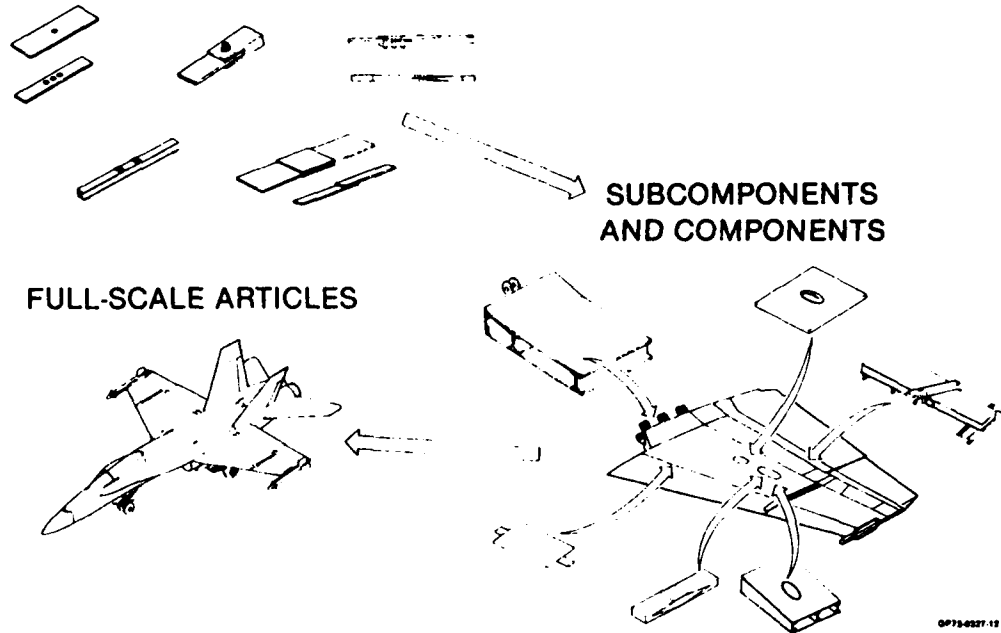
- FULL-SCALE STATIC TEST
 - 150 PERCENT DLL
 - RTA ENVIRONMENT
- FULL-SCALE FATIGUE TEST
 - 2x DESIGN SERVICE LIFE
 - RTA ENVIRONMENT

Vugraph 2

F/A-18 DEVELOPMENT/CERTIFICATION TESTS

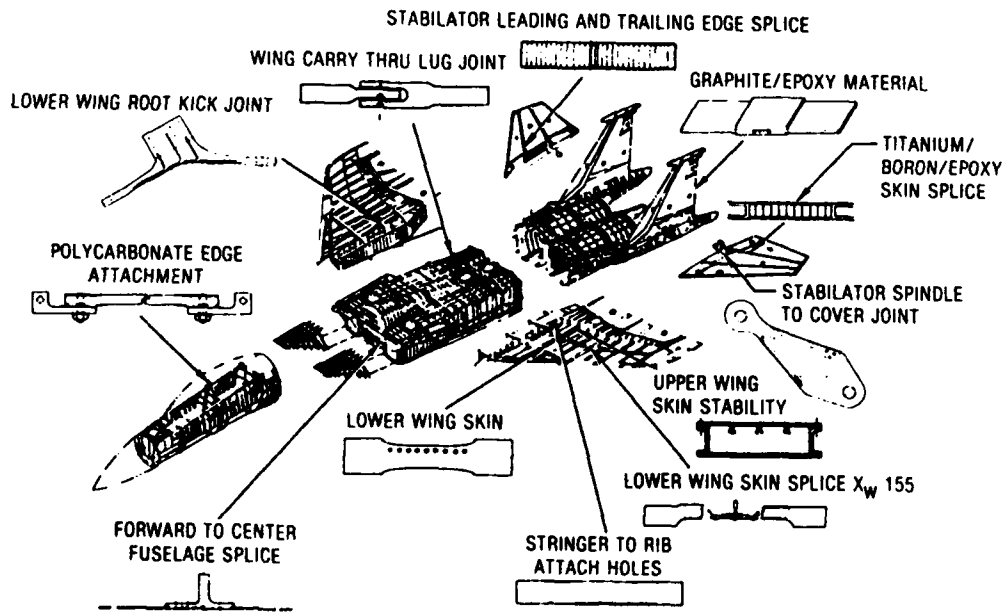
COUPONS AND ELEMENTS

Vugraph 3



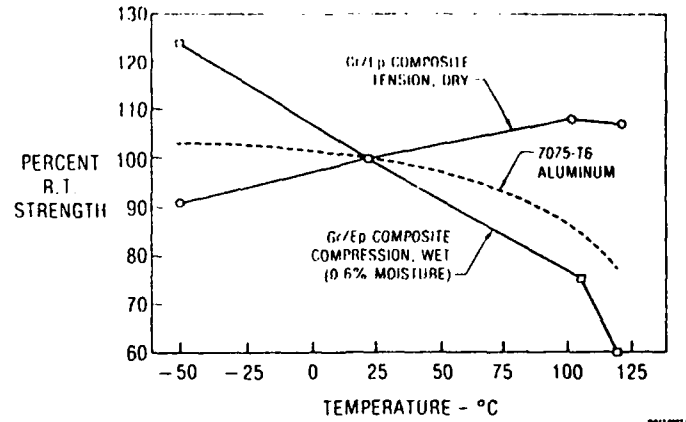
F-15 DEVELOPMENT TESTS

Vugraph 4

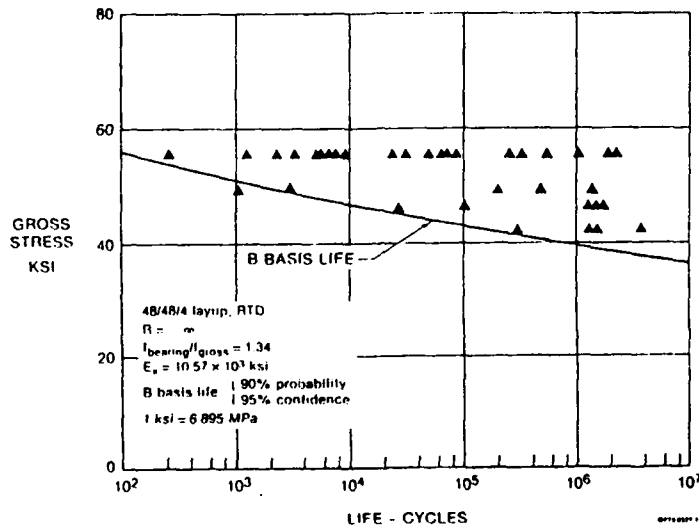


ENVIRONMENTAL SENSITIVITY

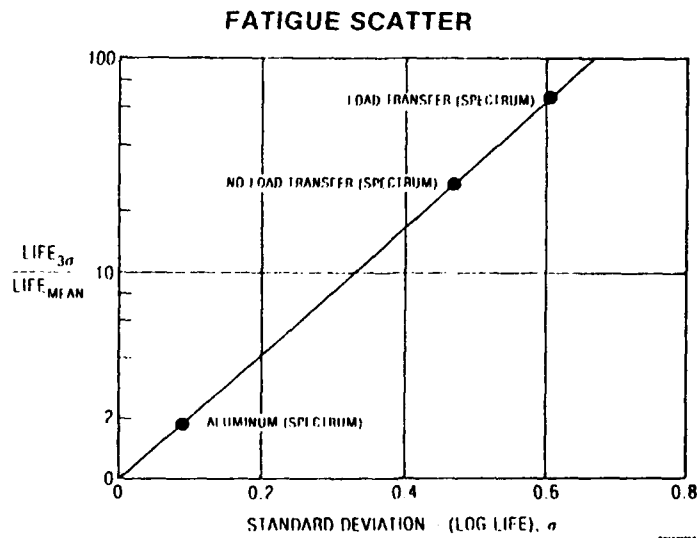
Vugraph 5



Vugraph 6

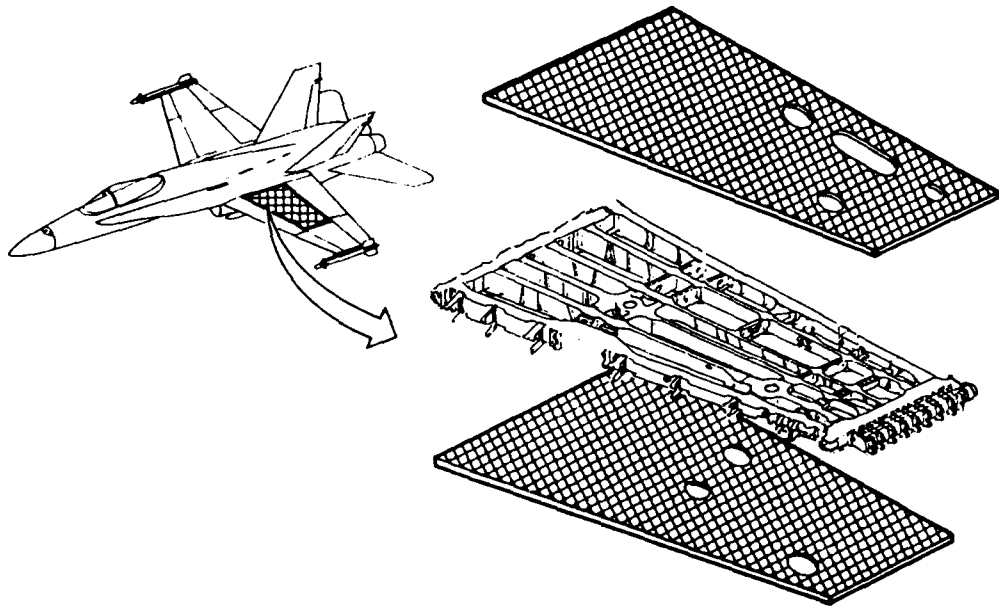


Vugraph 7



MIXED COMPOSITE/METAL STRUCTURE

Vugraph 8



STATIC STRENGTH CERTIFICATION APPROACHES:

Vugraph 9

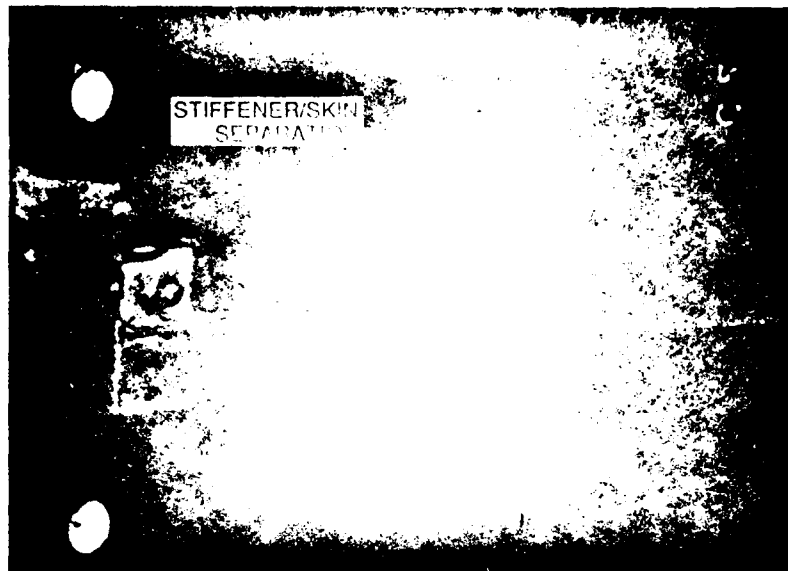
- CONDITION TEST ARTICLE
 - TEST CTD FOR TENSION LOADS
 - TEST ETW FOR COMPRESSION LOADS
- TEST UNDER RTA CONDITIONS
 - INCREASE LOADS TO COMPENSATE FOR ENVIRONMENT
 - CORRELATE MEASURED STRAINS WITH ENVIRONMENTAL ELEMENT DATA

FATIGUE CERTIFICATION APPROACHES

Vugraph 10

- APPLY A FACTOR TO FULL-SCALE ARTICLE TEST LIFE
- APPLY A FACTOR TO FULL-SCALE ARTICLE TEST LOADS
- INCREASE SEVERITY OF LOADS SPECTRUM
- CORRELATE STATIC STRAIN MEASUREMENTS WITH ELEMENT FATIGUE DATA/PERFORM 2X DESIGN LIFE FATIGUE TEST FOR METAL STRUCTURE

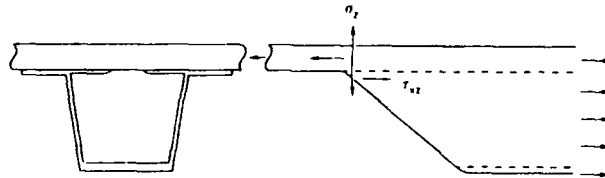
INDUCED CRIPPLING FAILURE DUE TO OUT-OF-PLANE LOADS



Vugraph 11

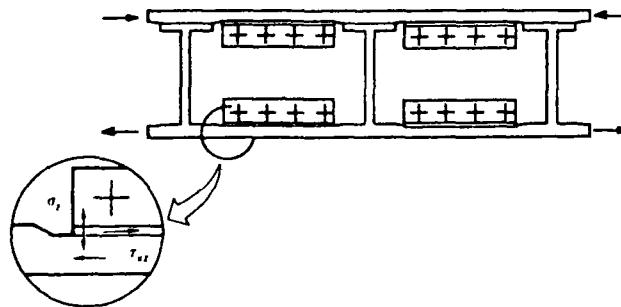
INDUCED OUT-OF-PLANE STRESSES DUE TO STRINGER RUNOUT AND STRUCTURAL DISCONTINUITY

a) INDUCED STRESSES DUE TO STRINGER RUNOUT



Vugraph 12

b) INDUCED STRESSES DUE TO STRUCTURAL DISCONTINUITY



CERTIFICATION REQUIREMENTS FOR LVID COMPOSITE STRUCTURE

INDUSTRY NEEDS

- AGREEMENT ON REALISTIC IMPACT DAMAGE THREAT
- VERIFIED DAMAGE/RESIDUAL STRENGTH METHODOLOGY
- TEST PROCEDURE
 - ALLOWABLES
 - NUMBER, TYPE, AND COMPLEXITY OF SPECIMENS
 - IMPACT AND TEST CONDITIONS

Vugraph 13

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CERTIFICATION OF PRIMARY COMPOSITE AIRCRAFT
STRUCTURES

R.S. Whitehead*

An overview of the extensive experience, lessons learned, and recommended certification procedures from two major USAF composite R&D programs is presented. Subject areas discussed in detail are static strength, fatigue/durability, and damage tolerance.

INTRODUCTION

The increased application of advanced composite materials in aircraft structures requires a critical assessment of the adequacy and applicability of existing metallic oriented certification specifications to this emerging class of materials. To do this, it is necessary to recognize the inherent differences between metals and composites. These inherent property differences led to an ad hoc qualification approach for early production hardware. This individual requirement development, or pay-as-you-go approach, while satisfying the immediate need at a "single copy" price, limited generic application and prolonged airframe development. Thus, in the long run, this approach was more expensive and time-consuming than a subscription price approach, which repeatedly uses established standardized specifications. A need exists, therefore, for an orderly, unified, consistent, and verified approach for designing, certifying, and force managing composite structures. This need has been addressed in two Air Force sponsored R&D programs. The purpose of these programs was to develop an extensive test data base on specimens ranging in complexity from coupons through elements, element

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NEW MATERIALS AND FATIGUE RESISTANT AIRCRAFT DESIGN

combinations, subcomponents, and full-scale wing and fuselage structures. This data base was then used to develop draft certification specifications for static strength, durability, and damage tolerance. In addition to the specifications, certification compliance procedures were also developed. Details of this work were presented previously in the open literature in references 1-8. This paper discusses experience, lessons learned, and recommended certification procedures for static strength, fatigue/durability, and damage tolerance of composite structures.

STATIC STRENGTH

Experience

Static strength testing of composites has shown that several inherent differences exist between composites and metals. These inherent differences are summarized in Figures 1-4.

Figure 1 compares the static strength notch sensitivity of composites and metals to stress concentrations such as fastener holes. The notched strength of metals follows the net section strength reduction line. In contrast, composites are very notch sensitive to fastener holes under both tension and compression loading. In fact, this behavior is similar to the linear elastic response of metals in the presence of fatigue cracks. The static strength notch sensitivity of composites is caused by their essentially linear elastic load-strain response. The sensitivity of static strength to loading direction is also different for composites and metals. Figure 2 shows this comparison for longitudinal (L), transverse (T), and out-of-plane (S) tension loading. Aluminum static strength is relatively insensitive to loading direction. In contrast, graphite/epoxy static strength is significantly influenced by loading direction. This is caused by the anisotropic characteristics of composites. The differences in L and T direction strength are simply a function of the percent 0° , $\pm 45^\circ$ and 90° plies used in the layup. However, strength in the S direction is controlled by the interlaminar tension strength between the plies, which is very low and is in the 3-4 ksi range.

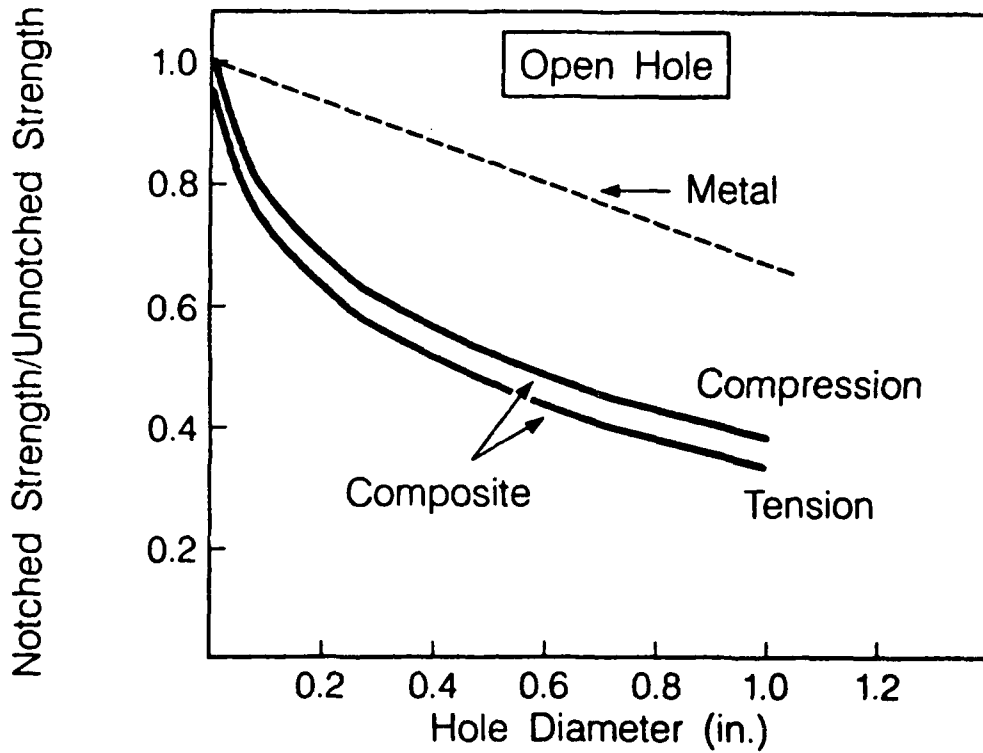


Figure 1 Static notch sensitivity comparison of graphite/epoxy and aluminum to fastener holes

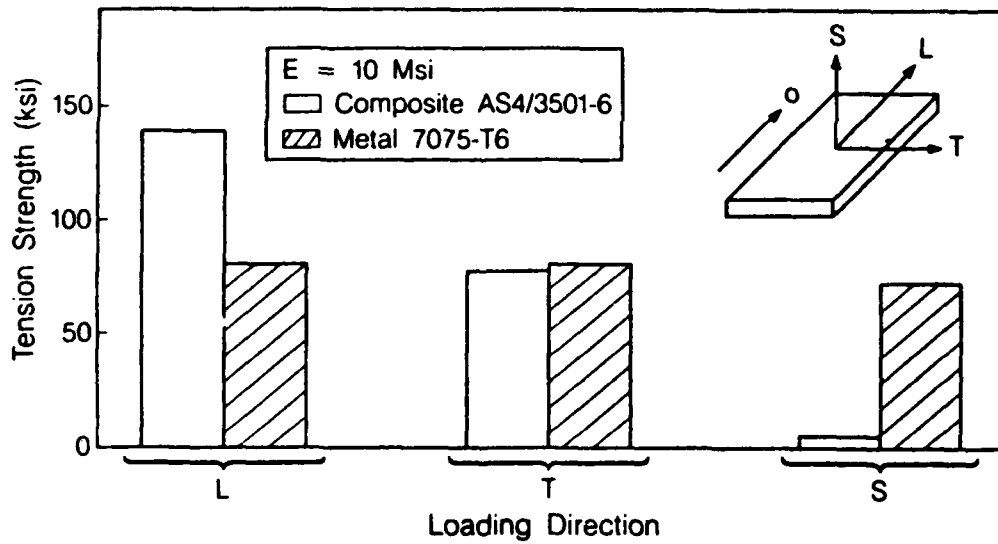


Figure 2 Influence of loading direction on graphite/epoxy and aluminum static strength

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Composites, which exhibit matrix controlled failure modes (e.g., compression), are sensitive to the aircraft hygrothermal environment. In particular, the effects of temperature and moisture have a synergistic effect. Therefore, the strength degradation of composites in hot/wet environments controls their maximum service usage temperature. Figure 3 shows the influence of temperature and moisture content on composite compression static strength. These data are for the 350°F cure system AS4/3501-6. Figure 3 shows that as the test temperature is increased above 220°F, strength loss (relative to ambient) is 15 percent, while at 250° strength loss is more than doubled to 33 percent. This large strength loss is due to rapid degeneration in resin properties (e.g., shear stiffness), which is caused by the resin approaching its glass transition temperature. In contrast, Figure 3 shows that aluminum strength is much less sensitive to temperature. Figure 4 compares the static strength variability of composites and metals. Because of their anisotropic heterogeneous characteristics, composites exhibit higher variability for laminate failure modes. For cocured composite-to-composite failure modes, even higher variability (10 percent coefficient of variation) is observed. This causes the ratio of the B-basics design allowable to mean value to be lower for composites compared to metals.

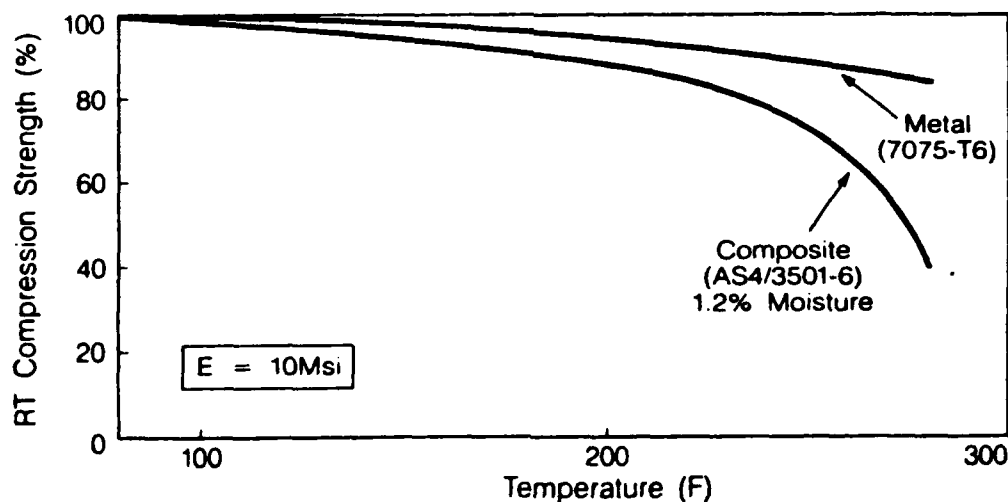


Figure 3 Influence of environment on compression static strength retention

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Material	Static Strength Variability		Design Allowable/ Mean
	α	C.V. (%)	
Aluminum	35	3.5	0.95
Composite – Laminate	20	6.5	0.89
Composite – Bonded/Cocured	12	10.0	0.84

Figure 4 Comparison of static strength variability of composites and metals

These property differences between composites and metals (notch sensitivity, weak transverse properties, matrix-dominated failures, higher variability, hygrothermal sensitivity) must be addressed in static strength structural certification. It is emphasized that these properties do not negate the weight efficiency of composite structures, just that different parameters (from metals) are important in composite certification.

The historical approach to design, analysis, and certification of composite structures has been similar throughout the industry. Composite design methods have been tailored to recognize the unique composite properties shown in Figures 1-4. In addition, conservative strain allowables (3,000-5,000 μ in/in at design ultimate load) coupled with semi-empirical analysis methods have been used for flight hardware. Design ultimate load had also been maintained at 1.5 times design limit load. Structural certification approaches have mainly been based on metals experience. This overall design, analysis, and certification approach has led to the following composite hardware experience:

- Significant weight savings compared to metal structure
- Mixed certification success
- Successful in-service application.

Problems associated with the static strength certification of composite structures are discussed below.

Figure 5 shows an outer wing box subcomponent tested in reference (1). The box consists of three bays with cocured intermediate spar to lower skin joints and an upper skin access hole. Fifteen wing boxes were tested as follows: 1) three room temperature ambient (RTA) static tests; 2) three RTA residual static strength tests after two lifetimes of severe fatigue loading; 3) three 250°F/1.3 percent moisture (ETW) static tests and six ETW residual static strength tests after two lifetimes of severe environmental fatigue loading. No fatigue failures occurred. The results are presented in Figure 6. The predicted failure mode for all the RTA tests was a lower skin failure mode, which was observed in five of the six tests. The failure mode in the sixth test was an unanticipated separation of the cocured intermediate spar/lower skin joints. This failure mode was not expected because the joint had a margin of safety of 1.35. Post-test failure analysis led to the following scenario for this failure mode. In addition to the shear flow in the joint, stress analysis of the steel shear clips, which were used to transfer shear load through the pylon rib, showed that the clips had a low torsional stiffness. The load carrying capability of the bonded joint was reduced due to bending moments induced by the relatively long length, L , of the shear clip. This is shown in Figure 7. Flexibility of the clip, where it attaches to the rib, caused the moment M to be small relative to VL , with the result that the moment must be reacted through the skin/spar joint. It was this flatwise tension load combined with the shear flow which led to the failure in the cocured joint. To inhibit this failure mode in the full-scale wing box, the shear clip was redesigned with enhanced torsional stiffness.

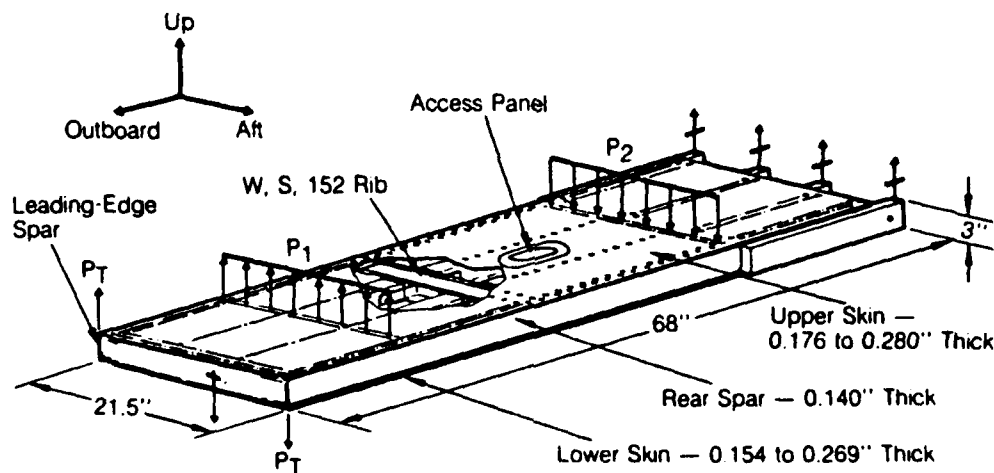


Figure 5 Wing box subcomponent tested in Reference 1

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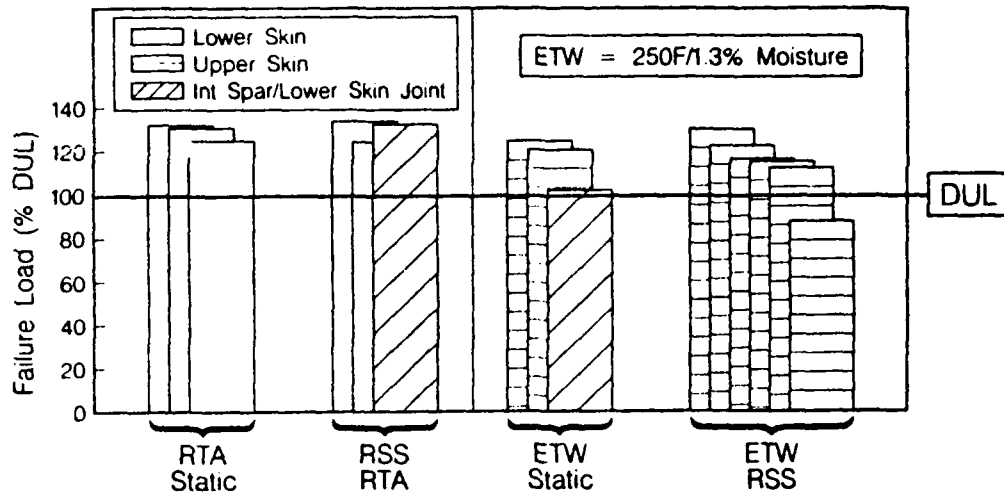


Figure 6 Summary of wing box subcomponent failure loads and modes

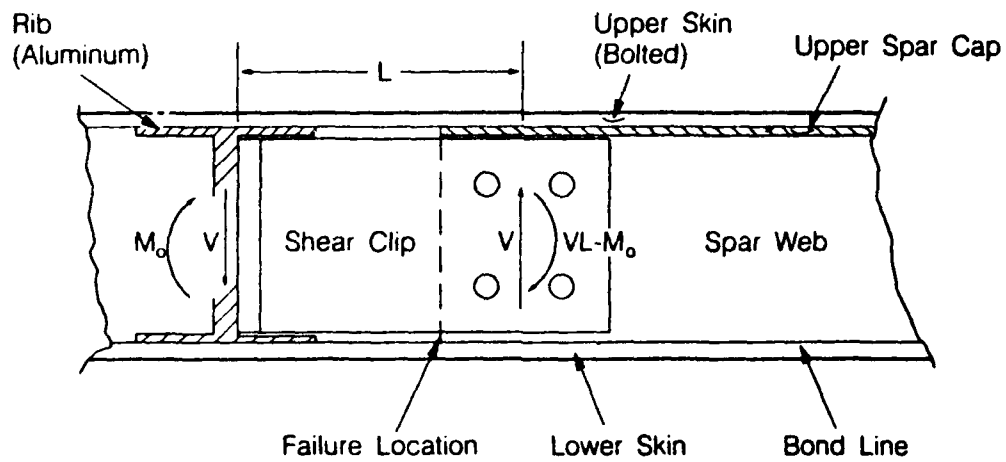


Figure 7 Wing box subcomponent lower skin/intermediate spar joint shear clip loading

The predicted failure mode for all the ETW tests was upper skin failure at the access hole, which was observed in eight of the nine tests. However, considerable strength scatter was observed (87 percent to 132 percent DUL). The ninth specimen failed by the same unanticipated intermediate spar/lower skin cocured joint failure mode previously observed under RTA conditions.

Following testing of the wing box subcomponents, four full-scale wing boxes were tested. The failure loads and failure modes are summarized in Figure 8. The predicted failure mode for the two RTA tests was lower skin failure. However, both test failures were caused by failure of the ten intermediate spar/lower skin cocured joints, as shown in Figure 9. This was the same unexpected failure mode as that observed in the wing box subcomponent and occurred despite a careful redesign of the spar to rib shear clips. The predicted ETW failure mode was upper skin failure, which was observed in both test articles. The predicted and observed failure mode change between RTA and ETW tests was caused by the static design conditions. For ambient conditions, a subsonic high N_z pull-up maneuver was the most critical design case, whereas a supersonic moderate N_z pull-up maneuver was the most critical design case for ETW conditions.

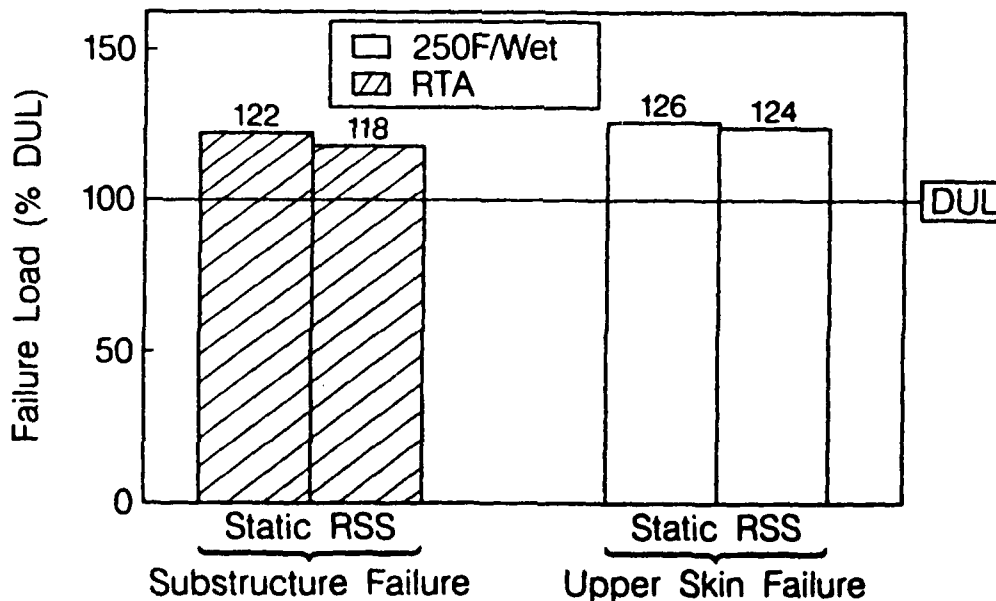


Figure 8 Full-scale wing failure modes and loads

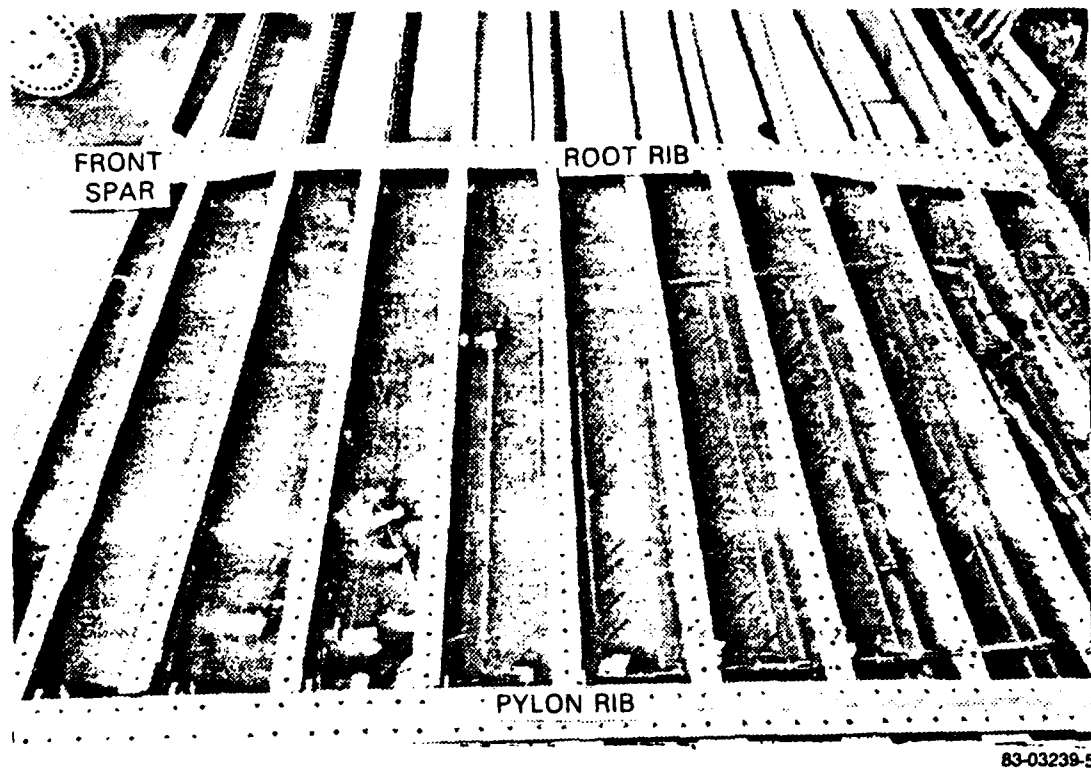


Figure 9 Full-scale wing box RTA intermediate spar/lower skin co-cured joint static failure mode

The occurrence of unexpected failure modes in full-scale static strength tests has occurred in many other hardware development programs. Unfortunately, for obvious reasons, many have not been documented in the open literature. One exception is the NASA ACEE program experience documented in reference (9). Unexpected failure modes were observed in three separate full-scale hardware tests. One example for a transport aircraft vertical stabilizer is shown in Figure 10.

The fin failed at 98 percent of design ultimate load during the planned test to 106 percent of design ultimate load in bending. Failure caused separation of the cover and front spar along the entire length of the spar as well as considerable internal damage to rib structure. After an investigation, the cause of failure was determined to be due to secondary loads, of which the principal contributor probably was local buckling of the cover near the front spar interface. While local buckling beyond limit load was allowed in the design, the influence of loads caused by buckling on the integrity of the structure was unexpected. Interlaminar tension forces caused delamination of the spar cap as shown by the insert in Figure 9 and ultimate separation along the line of fasteners.

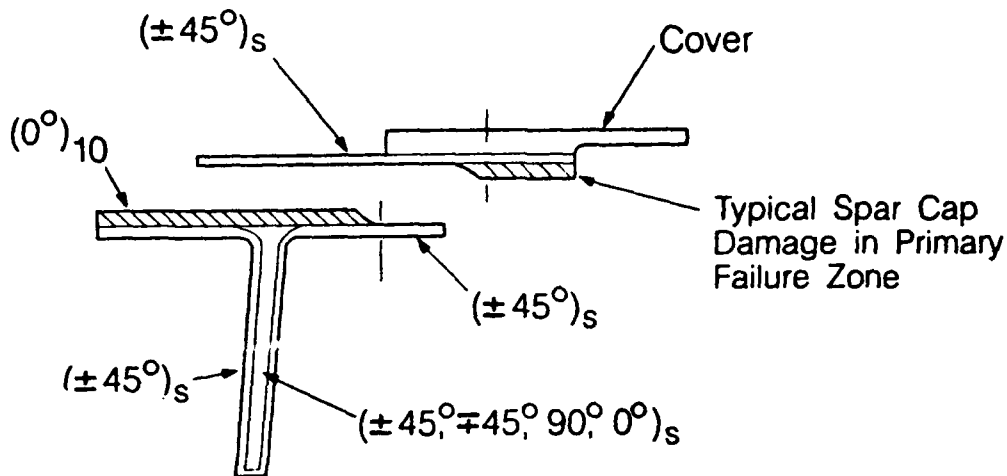


Figure 10 Front spar cap failure in a transport aircraft vertical stabilizer static test (Reference 9)

Lessons Learned

Some very important lessons have been learned from our static strength certification experience. First, the low interlaminar strength of composites makes them sensitive to out-of-plane loads. Out-of-plane loads can arise directly (e.g., from fuel pressure) or indirectly from in-plane loads. The most difficult loads to design and analyze for are those loads which arise insidiously in full-scale built-up structures. It is very important, therefore, to recognize all potential sources of out-of-plane loads and design composite structure to maximize interlaminar strength.

An example of this requirement is presented in Figure 11 for a composite torque box with cocured substructure and skins which are allowed to buckle prior to design ultimate load. The skin postbuckling produces out-of-plane flatwise tension, compression, and bending loads in the spars. The amount of postbuckling is limited by the strength of the skins, the skin/substructure interface, and the spar substructure. Figure 11 shows the possible failure modes for the all-composite torque box subjected to postbuckling loads. The cover skin can fail if the local outer fiber compression strength of the laminate is exceeded due to bending, or if the interlaminar shear strength of the laminate is exceeded. The skin/substructure attachment can fail if delamination occurs in the cocured joint, or if the transverse load exceeds the pull-through strength of the fasteners/laminate combination, or the strength of the spar caps or the transverse shear strength of the flange. In addition, the radius portion of the spar flange/web can fail if the interlaminar shear or flatwise tension strength is exceeded. Finally, compression failure of the

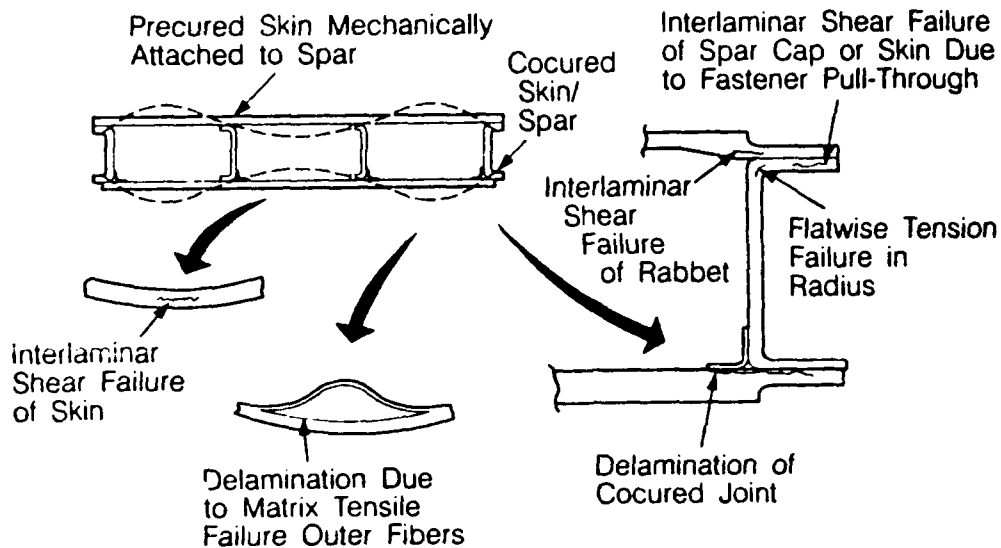


Figure 11 Potential out-of-plane failure modes in a composite torque box

spar web can occur if the web does not have enough strength to resist the crushing loads induced by buckling of the cover skins and overall stabilizer bending.

Another important conclusion from our static strength certification experience is as follows. The full-scale static strength test identifies structural "hot-spots." This is the reverse of our experience with metal structures, where "hot-spots" are generally identified in the full-scale fatigue test.

Figure 12 summarizes the lessons learned from static strength certification testing of composite structures.

Certification Recommendations

The follow static strength certification recommendations are based on the experience and lessons learned described above.

Material selection. Material selection is crucial to the successful application of composites in primary aircraft structures. Composite materials have operating limits just as aluminum does. Selection of a composite material should be based on the relationship between the aircraft hygrothermal envelope and the material operating limits (MOL). The material operating limit is reached when the synergistic effect of temperature and moisture causes severe degradation in resin mechanical properties. Good design practice dictates that composites should not be used in this regime.

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- Inherent Property Differences Exist Between Composites and Metals
- Composite Structures Are Sensitive to Out-of-Plane Loads
- Multiplicity of Potential Failure Modes
- Failure Modes of Full-Scale Structures Are Difficult To Predict
- Static-Strength Test Identifies Structural "Hot Spots"

Figure 12 Summary of lessons learned from static strength certification testing of composite structures

The concept of a MOL is shown schematically in Figure 13 for an environmentally sensitive failure mode. The decrease in design allowable strain as temperature increases is shown for a constant moisture level. Catastrophic strength loss coincides with severe degradation in resin properties as the glass transition temperature (T_g) is reached. In order to operate in a safe regime, the MOL should be reduced below the T_g by a safety factor K . This produces the shaded material service envelope shown in Figure 13.

The discussion above appears to be a statement of the obvious. However, violations of the approach shown in Figure 13 have occurred and have led to disastrous certification experiences. It should also be noted that careful compliance with the requirements in Figure 13 will minimize environmental issues in the subsequent certification test program.

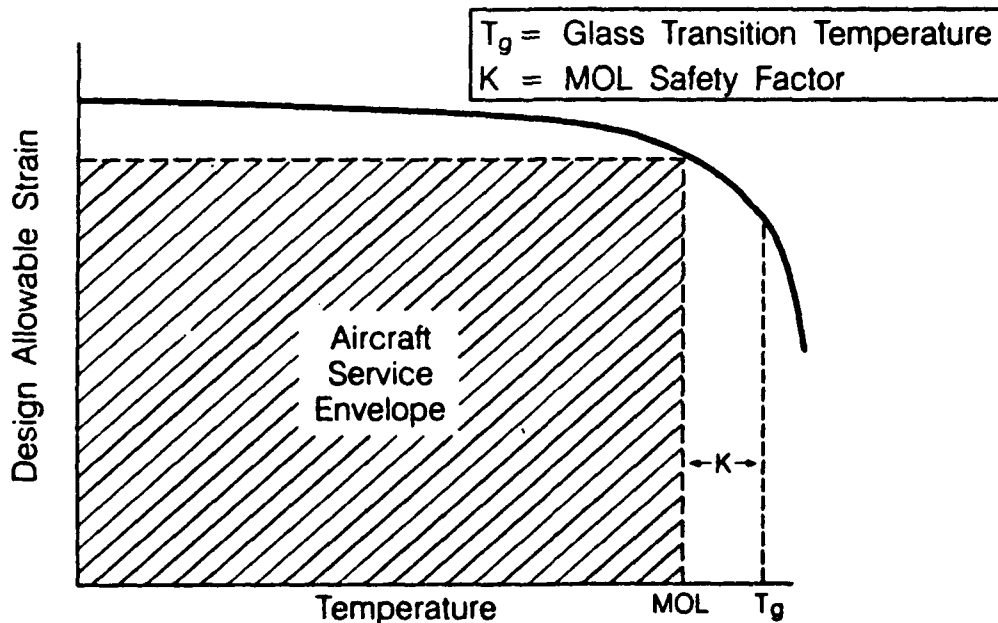


Figure 13 Material selection criterion

Design verification testing. Design development tests are conducted prior to the full-scale test. The objective of these tests is to validate the design of critical structural features.

A building block approach to design development testing is crucial for the certification of composite structures because of their sensitivity to out-of-plane loads and their multiplicity of potential failure modes. This is discussed in more detail in reference (5). The essence of the building block approach for composites is as follows. First, use the design/analysis of the aircraft structure to select critical areas for test verification. Second, determine the most strength-critical failure mode for each design. Third, select the test environment which will produce the strength critical failure mode. Special attention should be given to matrix sensitive failure modes (such as compression and bondline) and potential stress "hot spots" caused by out-of-plane loads. Following selection of the critical failure modes, a series of specimens is designed, each one to simulate a single failure mode. These specimens will generally be low complexity specimens. However, the crux of the building approach is to also design test specimens which simulate progressive design complexity. In this way, multiple potential failure modes are interrogated.

This building block method to design development testing provides a step-by-step approach to composite design development testing, which has several advantages:

- The influence of the environment on individual failure modes is determined.
- The interaction of failure modes is established from the known behavior of individual failure modes.
- Scale-up effect is determined from data on smaller-scale specimens.
- "Hot spots" induced in complex structures can be analyzed relative to the known behavior of smaller specimens.

Specimen complexity should be a function of the design feature being validated and the predicted failure mode. Special attention should be given to correct failure mode simulation, since failure modes are frequently dependent on the test environment. In particular, the influence of complex loading on the local stress at a given design feature must be evaluated. In composites, out-of-plane stresses can be detrimental to structural integrity and therefore require careful evaluation.

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An example of the building block approach for specimen complexity is given in Figure 14, which shows the approach used for the wing structure in reference (1). Here the wing structure has been broken down into critical areas. Each critical area has been simulated in a test specimen whose complexity is governed by the necessity to simulate all potential failure mode(s). Particular attention was given to matrix critical failure modes. The following recommendations are made for specimen complexity simulation in design development testing:

1. Use the design/analysis of the aircraft structure to select critical areas for test verification.
2. Specimen complexity should be controlled by the requirement to simulate the correct (full-scale structure) failure mode(s) in the specimen.
3. Special attention should be given to matrix sensitive failure modes, such as compression, bondline, and hole wear.
4. Potential "hot spots" caused by out-of-plane loads should be carefully evaluated.

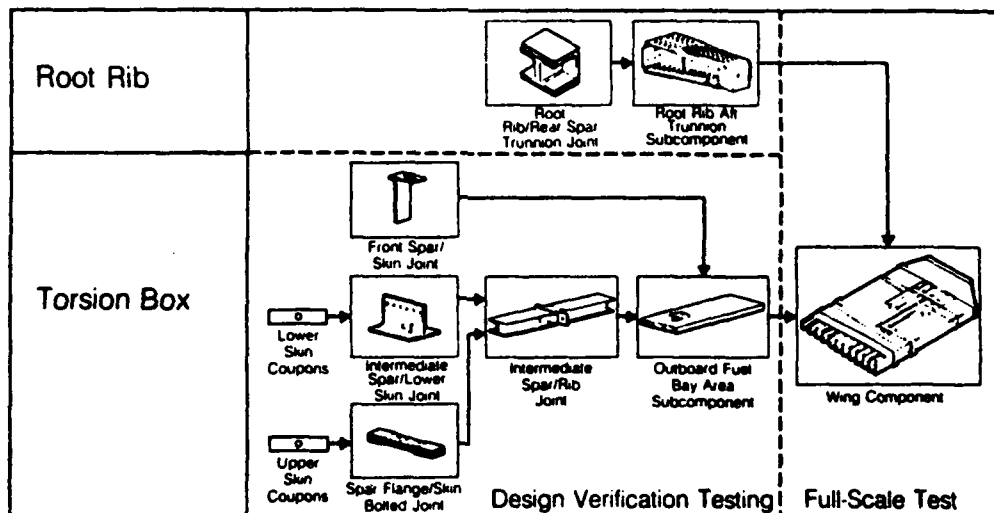


Figure 14 Building block approach used for the wing structure in Reference 1

The sensitivity of composite matrix dominated failure modes to the aircraft hygrothermal environment makes environmental test simulation a key issue. Environmental test simulation should be considered separately for static and durability testing. However, the static test philosophy will form an integral part of the overall test philosophy. The philosophy for static design development tests should be that the test environment used will be the one that produces the failure mode which gives the lowest static strength.

Full-scale test. The full-scale static test is the most crucial qualification test for composite structures for the following reasons. Secondary loads are virtually impossible to eliminate from complex built-up structures. Such loads can be produced by eccentricities, stiffness changes, discontinuities, fuel pressure loading, and loading in the post-buckled range. Some of these sources of secondary loads are represented for the first time in the full-scale structural test article. These loads are not a significant design driver in metallic structures. However, the poor interlaminar strength of composites makes them extremely susceptible to out-of-plane secondary loads. It is very important, therefore, to carefully account for these loads in the design of composite structures. Unfortunately, there is a general state of uncertainty as to the source, magnitude, and effects of secondary loads in complex built-up full-scale composite structures. This has been confirmed by several documented examples of unanticipated secondary loads leading to unexpected failure modes in full-scale composite structural static tests.

In addition, a detailed correlation in terms of measured load and strain distributions, structural analysis data, and environmental effects between the design development and full-scale test data will be necessary to provide assurance of composite static strength. Static test environmental degradation must be accounted for separately either by adverse condition testing, by additional test design factors, or by correlation with environmental design development test data.

Work in reference (1) has shown that the RT/ambient static test plays the most significant role in revealing unexpected hot spot failures from secondary out-of-plane loads. A room temperature environment is, therefore, recommended for the full-scale static test, which should be conducted to failure. This recommendation is not universally accepted by all certification agencies. Some agencies favor an environmental static test which corresponds to the temperature and absorbed moisture level of the most critical static design condition. This issue is most significant in fighter aircraft where a hot/wet failure mode is often the most critical design condition. Unfortunately, a full-scale environmental static test is very expensive and time-consuming. A possible solution to this problem, proposed by

Dr. Lincoln in reference (3), is as follows: the design philosophy would not permit any significant change in failure mode due to environment. For example, consider an aircraft structure, where the most critical design load condition is associated with a hot/wet environment. The requirement of this philosophy would be for the structure to be designed to have the same failure mode under both hot/wet and RT/ambient conditions. This approach would eliminate the need for hot/wet qualification testing. If this design requirement had been adopted for the fighter aircraft wing structure in reference (1), a weight penalty of approximately 6 percent would have been incurred in the main wing box structure.

FATIGUE/DURABILITY

Composites have superb fatigue properties. Figure 15 compares the RTA spectrum fatigue behavior of graphite/epoxy and aluminum under their respectively most sensitive fatigue loading modes. It can be seen that graphite/epoxy fatigue response is vastly superior to aluminum. This has been confirmed by the extensive environmental data base generated in reference (1) and summarized in reference (6).

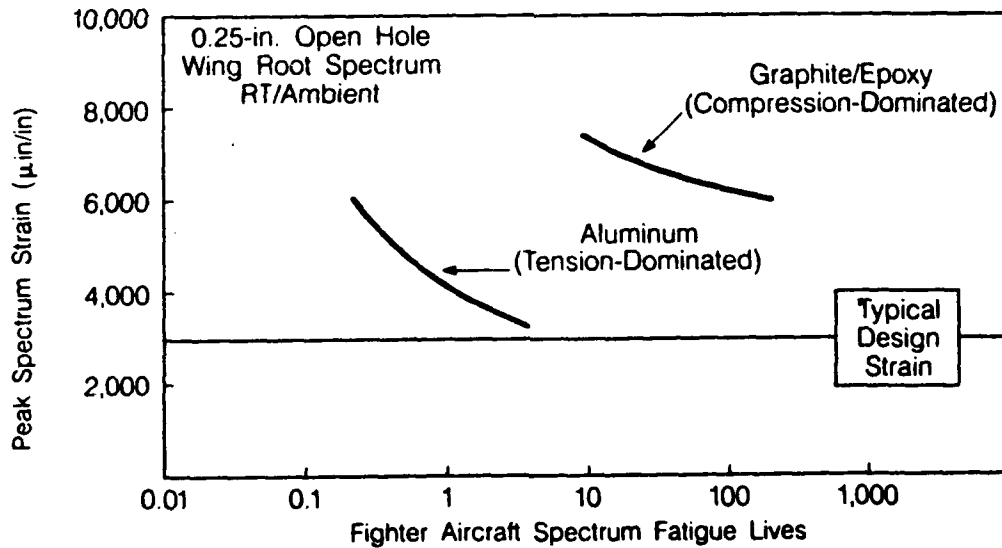


Figure 15 Comparison of graphite/epoxy and metal spectrum fatigue behavior

In this work, several hundred specimens ranging in complexity from coupons to elements to element combinations to subcomponents to full-scale wing and fuselage structures were tested under very severe environmental and fatigue loading conditions. The fatigue loading was much more severe than the design spectrum. All tests were conducted with the maximum spectrum load set at 72 percent of the previously determined average static failure load, which led to test spectra with significant load enhancement factors compared to design. In addition, severe quasi-real time environmental cycling was imposed on the test articles. This involved continuous thermal cycling, severe thermal spikes, and regular moisture absorption/deabsorption cycles. Representative results are presented in Figure 16. No fatigue failures occurred in the two lifetimes of fighter aircraft fatigue loading and all specimens were residual static strength tested at 250°F/wet environmental conditions. Figure 16 shows some variability in residual strength; however, this was determined to be due to scatter in static strength rather than fatigue degradation. It should be noted that even matrix sensitive failure modes such as compression and out-of-plane flatwise tension were not fatigue sensitive.

250F/1.2% Moisture RSS Data

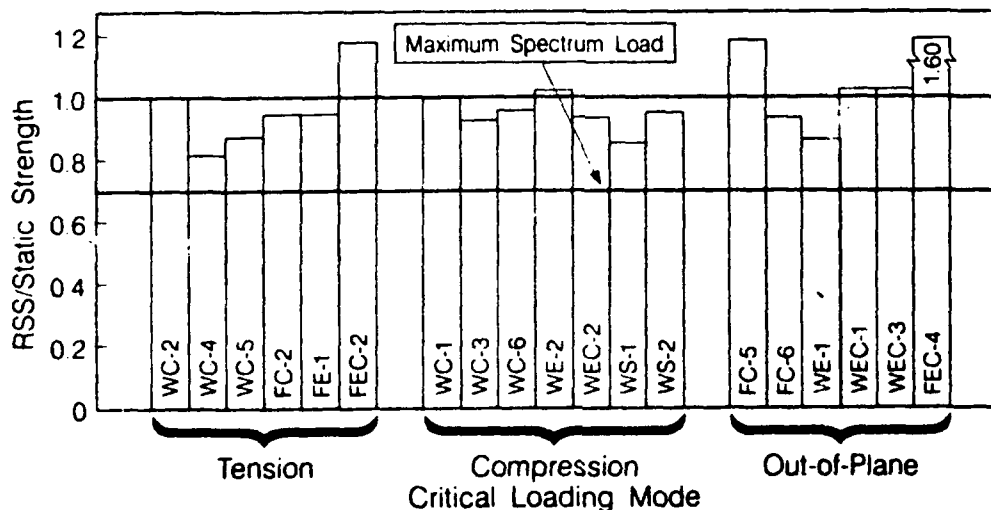


Figure 16 Environmental spectrum fatigue and residual static strength response of composite structures

Figure 17 shows a comparison of the scatter in spectrum fatigue life observed in composites and aluminum. The scatter in life is inversely proportional to the Weibull shape parameter (α). That is, the higher the value of α , the lower the scatter in fatigue life data. Figure 17 shows that composites exhibit significantly higher scatter than aluminum. This is caused by the significantly flatter S-N curves (superior fatigue resistance) observed in composites.

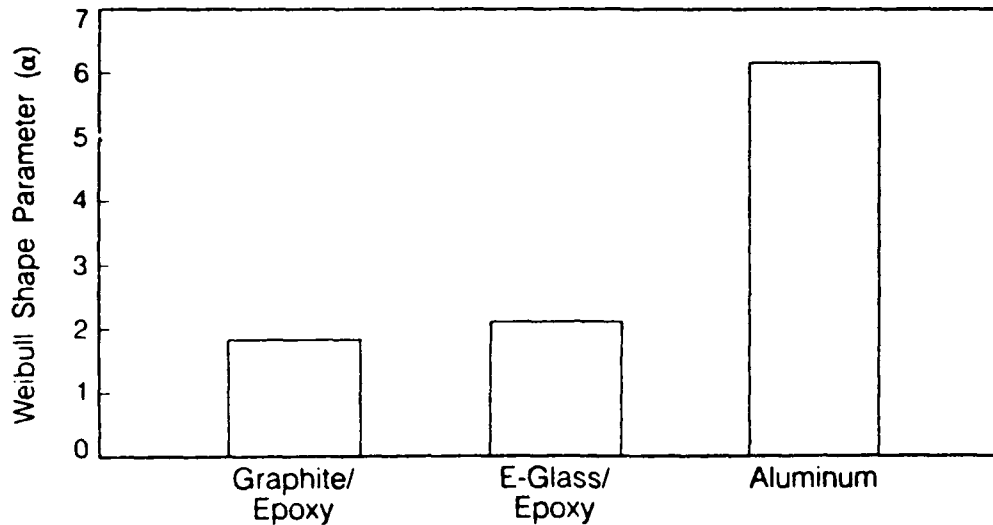


Figure 17 Comparison of the spectrum fatigue life data scatter of composites and aluminum

In metallic structures, it has been demonstrated that both fatigue initiation life and crack growth life are a function of load sequence. This load sequence dependence is caused by high loads producing residual compressive stresses, which reduce the fatigue damage accumulation rate. Historically, therefore, considerable attention has been given in metallic fatigue tests to careful simulation of the flight-by-flight loading history. In particular, the number of high loads included in the fatigue test spectrum has been a subject of concern. A common practice is to delete some high loads from the fatigue test spectrum in order to provide a conservative fatigue test, since retardation of crack initiation will be reduced by the omission of the high loads. In composite materials, no significant load sequence effect on fatigue life has been observed. However, studies on load spectrum variations have shown that composites are extremely sensitive to variations in the number of high loads in the fatigue spectrum. In contrast, truncation of low loads does not significantly affect fatigue life. These differences in load spectrum sensitivity may lead to contradictory load history requirements for a mixed composite/metal fatigue test. For example, removal of some high loads may be prudent for the metallic structure, while their removal may cause significant overestimation of composite fatigue life.

Although composites have outstanding fatigue resistance, they have exhibited some durability sensitivity. Durability is defined by the USAF as a measure of economic life. Adequate structural durability is assured by eliminating functional impairment during the life of the airframe. Functional impairment occurs when excessive repair or part replacement causes unacceptable economic burden. Thus, durability is an economic issue, not a safety issue.

The durability in-service experience with monolithic structures has been excellent. However, durability experience with thin-skinned honeycomb structure has been less satisfactory. The following problems have occurred:

- Sensitivity to low-level impacts (<10 ft-lb), causing visible skin damage, nonvisible skin or core damage, accelerated moisture intrusion, and core corrosion
- High repair frequency
- Excessive part replacement.

These problems have caused unacceptable maintainability and supportability costs.

Lessons Learned

Two major lessons have been learned from our composite fatigue/durability experience. These are:

1. Composites have outstanding fatigue resistance. For realistic structural laminates in typical design applications, composite structures can be considered to be fatigue insensitive, if they possess adequate static strength.
- 2 Maintainability and supportability of thin-skinned honeycomb structures has been poor.

Certification Recommendations

Detailed recommendations are given in reference (1); and are summarized below.

Load Spectrum Simulation. The same general guidelines established for metallic structures should be used. The following recommendations are made for load spectrum simulation in composite fatigue testing:

- High loads in the fatigue spectrum must be carefully simulated.
- Low loads (< 30 percent limit load stress) may be truncated to save test time without significantly affecting fatigue life.

Mixed composite/metal structure. Because of the superior fatigue performance of composites, a mixed composite/metal structure fatigue will essentially interrogate adequately only the metal structure. Thus, any potential "hot spots" in the composite structure may not be found.

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Because of the potential inadequacy of full-scale tests on mixed composite/metal structures and also the natural reluctance to overdesign metal parts in a full-scale test structure, it will be necessary to validate the composite structure during the design development testing phase. However, the specimen complexity should be adequate to enable the performance of the full-scale structure to be correctly simulated. Validation of the composite structure using subcomponent tests can offer the following advantages:

- The components may be chosen for test purposes to interrogate the composite structure only.
- If environmental test conditions are required, it will be easier and cheaper to achieve in a component.
- It may be possible to test more than one replicate and thus increase confidence in the data base.
- The results can be utilized in the certification of the full-scale structure.

For component tests to achieve their objective, great care must be taken in getting the boundary conditions correct. In addition, eliminating metal failure modes by overdesign or replacement must be carefully evaluated so that relative effects such as differential thermal expansion are not masked.

Environmental simulation. The environmental complexity necessary for fatigue design development testing will depend on the aircraft hygrothermal history. Three factors must be considered. These are: structural temperatures for each mission profile, the load/temperature relationship for the aircraft, and the moisture content as a function of aircraft usage and structure thickness. In order to obtain these data, it is necessary to derive real-time load-temperature profiles for each mission in the aircraft's history. These relationships will have a significant influence on the fatigue test environment, and are strongly dependent on the aircraft type, configuration, and mission requirements and must be carefully developed on a case-by-case basis. The structural material should be selected to meet these mission requirements using the criterion in Figure 13. If this is accomplished, hot/wet fatigue testing will be minimized. Material selections which lead to significant environmental fatigue test requirements should be a last resort.

Scatter. The large scatter in composite fatigue life data makes the traditional life factor approach used for metals impractical because equivalent composite life scatter factors are in the 20-70 range. Alternate approaches to account for scatter effects were evaluated in reference (10).

The first approach was use of the load enhancement factor. The objective of this approach is to increase the applied loads in the fatigue certification tests so that the same level of reliability can be achieved with a shorter test duration. A schematic showing this approach is shown in Figure 18, where the fatigue life scatter represented is typical of that observed in composites. At one fatigue lifetime, a typical residual strength distribution is shown. If the maximum applied load in the fatigue test (P_F) is increased to the mean residual strength at one lifetime (P_T), then the B-basis residual strength of the structure would be equivalent to the design maximum fatigue stress. Thus, a successful fatigue test to one lifetime at applied stress (P_T) or a fatigue test to N_F would both demonstrate B-basis reliability. In addition, combinations of the load enhancement and fatigue life factors could also be used to demonstrate B-basis life. In order to use this approach with confidence in a certification methodology, a formal relationship between the load enhancement factor (LEF) and the life factor is required. This was verified mathematically in reference (10).

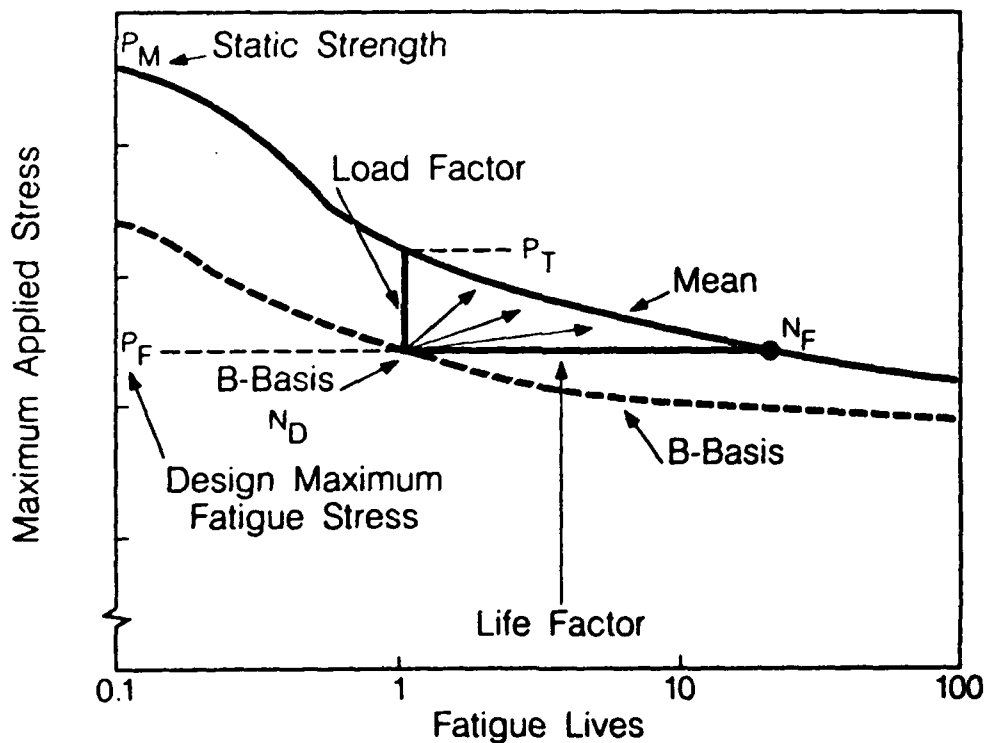


Figure 18 Load enhancement factor approach

While the evaluation of the enhanced loads approach in reference (10) has shown that it has a sound theoretical basis and can be used with confidence for certification testing, some practical limits of this approach exist. First, for asymmetric spectra, the degree of load enhancement may be limited because of a requirement not to exceed ultimate load. Second, for mixed structures, the enhanced load approach will provide an excessively severe fatigue test for the metal parts.

A second approach takes advantage of the excellent fatigue response of composites and is summarized in Figure 19. The objective of this approach is to set fatigue stress allowables below the B- (or A-) basis fatigue threshold. This is possible in practice because composites have flat curves where the fatigue threshold is a high proportion of the static strength.

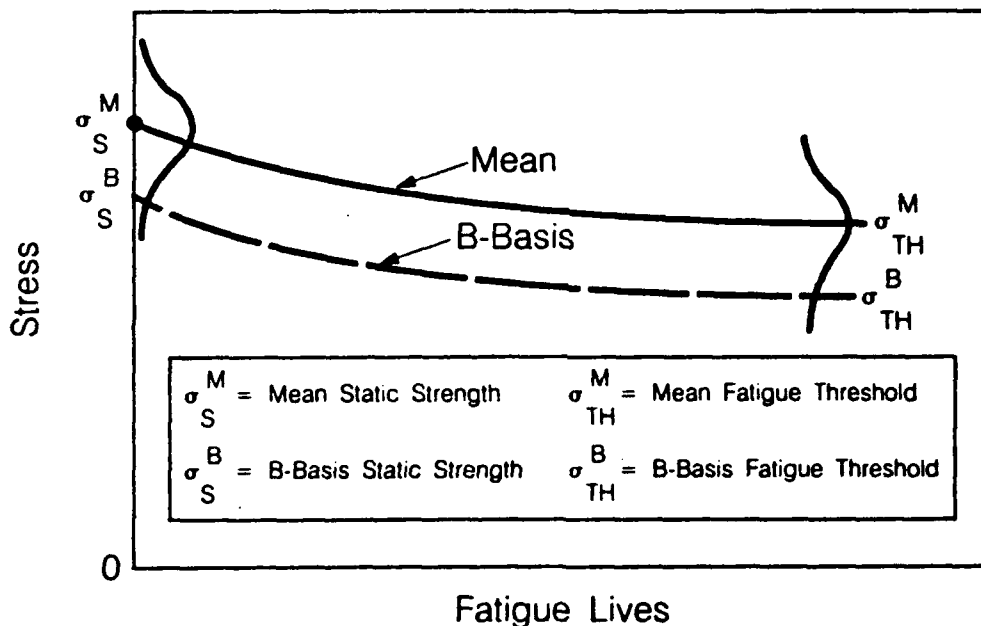


Figure 19 Fatigue life threshold approach

Durability. The poor service experience with thin-skinned honeycomb composite structures has led the USAF to introduce draft low-level impact design requirements in reference (3). These are summarized in Figure 20. The object of these requirements is to improve the maintainability of composite structures.

Full-Scale Test. The work in reference (1) and other USAF-sponsored programs have shown that composites possess excellent durability. In particular, the extensive data base developed in reference (1) showed that composite structures, which demonstrated adequate static strength, were fatigue insensitive.

NEW MATERIALS AND FATIGUE RESISTANT AIRCRAFT DESIGN

Zone	Damage Source	Damage Level	Requirements
1. High Probability of Impact	<ul style="list-style-type: none"> ● 0.5-in. Diameter Solid Impactor ● Low Velocity ● Normal to Surface 	<ul style="list-style-type: none"> ● Visible ● 6 ft-lb Max 	<ul style="list-style-type: none"> ● No Functional Impairment or Structural Repair Required for Two Design Lifetimes ● No Water Intrusion ● No Visible Damage From a Single 4 ft-lb Impact
2. Low Probability of Impact	<ul style="list-style-type: none"> ● Same as Above 	<ul style="list-style-type: none"> ● Same as Above 	After Field Repair of Visible Damage: <ul style="list-style-type: none"> ● No Functional Impairment After Two Design Lifetimes ● No Water Intrusion

Figure 20 Proposed USAF low-level impact certification requirements (Reference 3)

Thus, it is recommended that no durability full-scale test is required for all composite structures or mixed composite/metal structures with non-fatigue critical metal parts, provided the design development testing and full-scale static test are successful. For mixed structure, with fatigue critical metal parts, a two-lifetime ambient test will be required for fatigue validation of the metal parts.

DAMAGE TOLERANCE

Experience

Extensive work was conducted in reference (2) to determine the influence of defects and damage on composite static strength and fatigue life. The results are presented in reference (11), and are summarized in Figures 21-23. The data presented are representative of wing skin laminates fabricated from AS4/3501-6 and were obtained from 5-inch wide coupons.

A defect/damage severity comparison for compression static strength is presented in Figure 21. The plot relates damaged static strength to defect/damage severity. The data are also compared to the strength reduction for a 1/4-inch-filled unloaded hole. Porosity up to two percent, delaminations up to 1.5-inch diameter, and surface scratches are less than or equal to the strength reduction caused by a 1/4-inch hole. Fastener holes with delaminations around them show negligible strength loss compared to an unflawed fastener hole. In contrast, blunt impact damage causes a strength loss which significantly exceeds that of a 0.25-inch hole. Severe blunt impacts reduce strength by up to

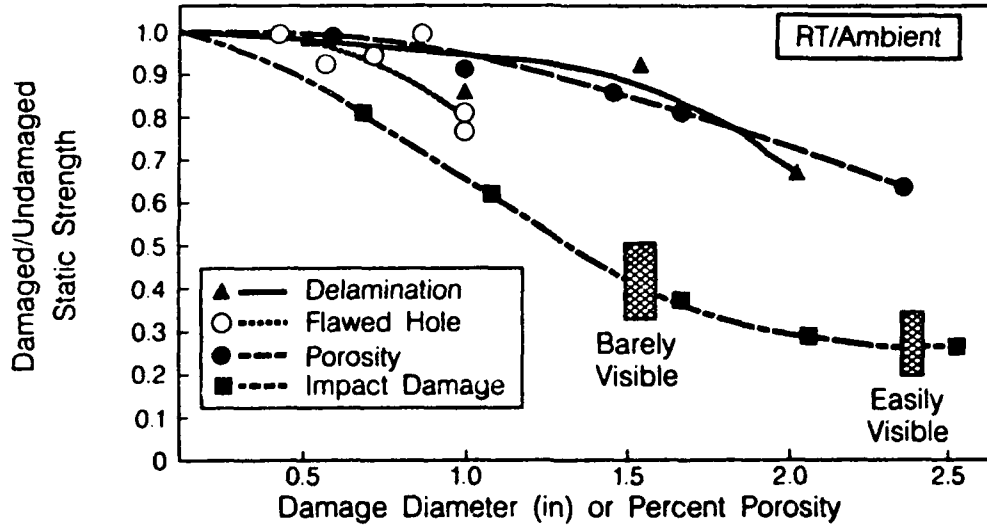


Figure 21 Compression static strength defect/damage severity comparison

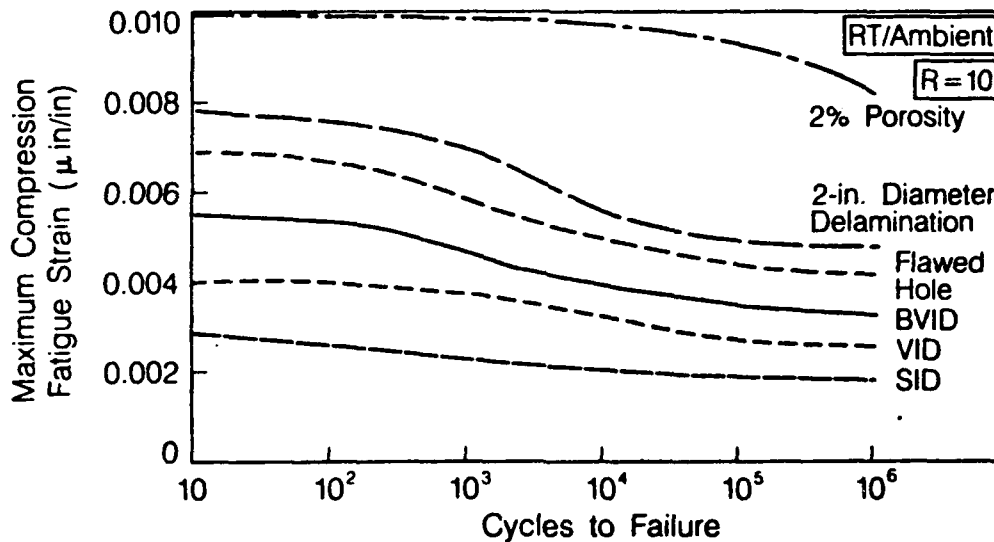


Figure 22 Compression fatigue life defect/damage severity comparison

60 percent; this is greater than the strength reduction of a 1.0-inch open hole. Clearly, therefore, impact damage is the most severe damage type for static compression strength.

A defect/damage severity comparison for compression-compression fatigue loading is shown in Figure 22. The material system is T300/5208, except for the porosity data which are the AS/3501-6 material system. The fatigue data show the same defect/damage severity trends as those observed for static

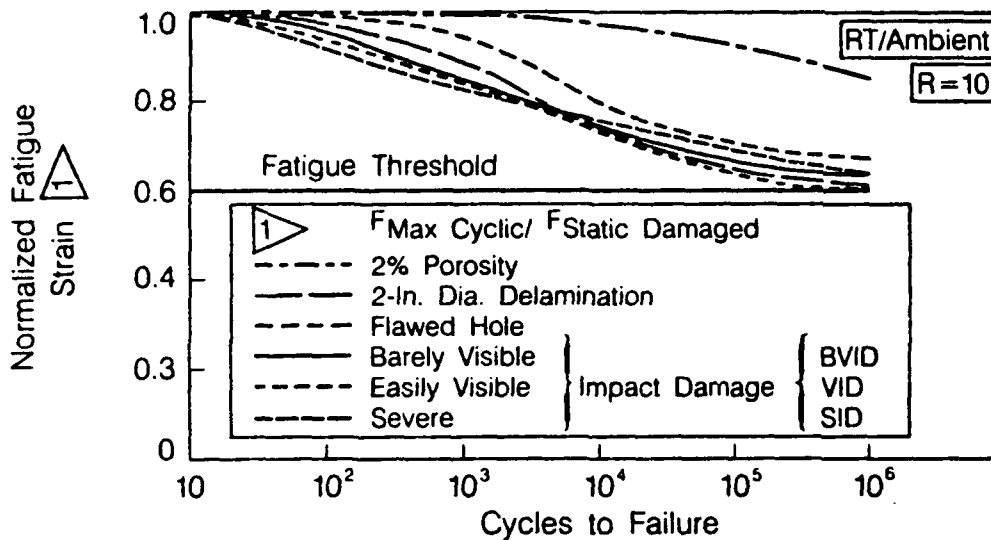


Figure 23 Normalized compression fatigue life defect/damage severity comparison

strength. Nonvisible and visible impact damage have the greatest fatigue sensitivity in terms of the fatigue strain required to give a life of 10⁶ cycles. This is caused by the greater static strength sensitivity of these damage types.

The data in Figure 22 are replotted in Figure 23 in terms of normalized fatigue strain (maximum fatigue strain ÷ damaged static failure strain). These data show that, for all of the defect/damage types, a potential fatigue threshold (10⁶ cycles) exists at 60 percent of damaged static strength for constant amplitude loading.

To check the applicability of these coupon data, extensive built-up structure damage tolerance testing was conducted in references (2) and (7). The work on 3-spar panels representative of fighter aircraft upper wing skin-to-spar attachments was summarized in references (2) and (11).

The specimen design is shown in Figure 24. The flat, stiffened panel specimen was loaded in compression through potted ends and was typical of fighter aircraft upper wing skin/spar attachment area. The skin was fabricated from 24 plies of double thickness AS4/3501-6 graphite/epoxy tape, which provides a nominal skin thickness of 0.25-inch. The channel spars were fabricated from 0.125-inch thick formed titanium. The specimen was designed to permit skin buckling to occur at approximately 5,000 μ in/in, which is typical of a fighter aircraft skin design.

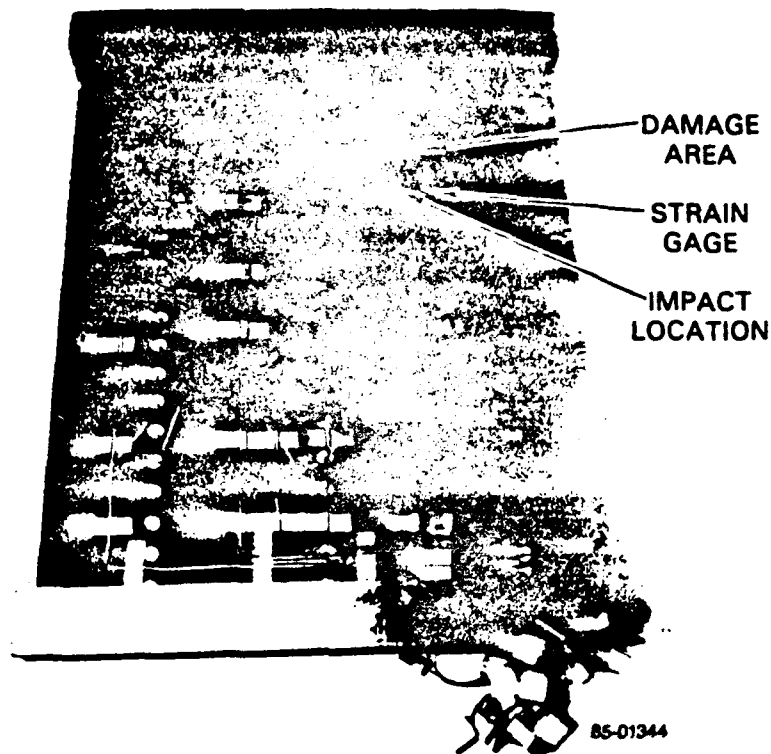


Figure 24 Impact damaged 3-spar panel - 100 ft-lb

Three spar test panels were impacted at 100 ft-lb at three skin locations, midbay, over the spar cap edge, and over the spar cap between fasteners. Maximum indentation depth on the impact surface was 0.05 inch. The influence of impact location on C-scan damage area is shown in Figure 25. Although some scatter is observed in the data, the mid-impact damage location clearly causes the largest damage area.

Figure 26 shows Panel 37-1, which was subjected to a 100 ft-lb impact prior to static compression testing. Failure sequence was as follows. At a gross applied skin compression strain equal to $2,500 \mu\text{in/in}$, the central region of the midbay impact damage area propagated rapidly to the spar attachments and arrested. Additional loading to a gross skin strain equal to $3,770 \mu\text{in/in}$ caused catastrophic panel failure through the midbay impact damage. This distinct two-stage static failure process permitted a 50 percent additional load-carrying capacity after initial failure. Further replicate tests again showed a 50 percent additional load-carrying capability after initial failure and arrest.

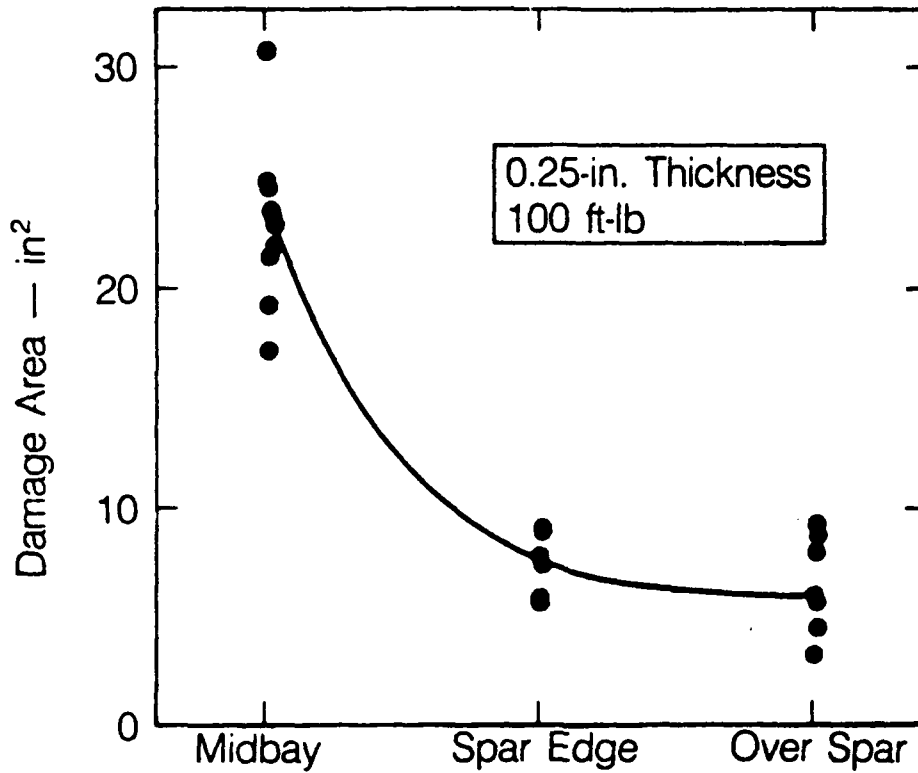


Figure 25 Influence of impact location on C-scan damage area

Failure Sequence

1. Damaged Area Failed and Arrested at Spars (2,500 $\mu\text{in/in}$)
2. Catastrophic Failure Through Impact Damage Area (3,770 $\mu\text{in/in}$)

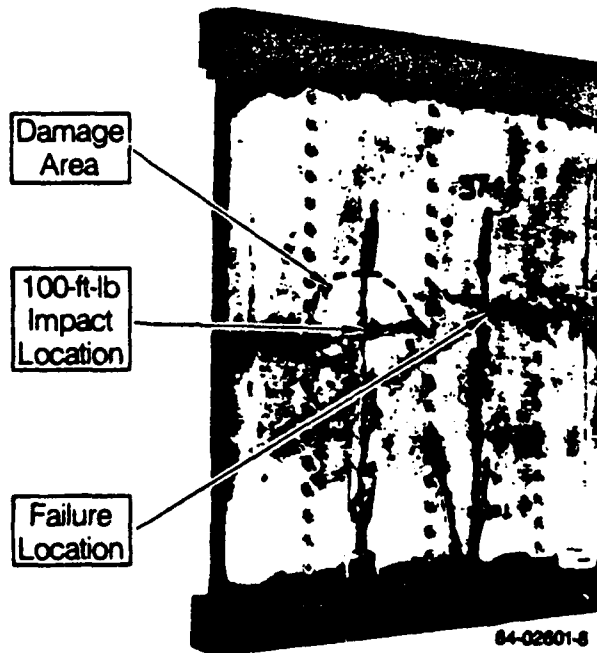


Figure 26 Static failure of impact damaged 3-spar panel - AS4/3501-6

Additional tests were conducted on panels with midbay impacts ranging from 20 to 83 ft-lb. The results are summarized in Figure 27. All test panels (except 20 ft-lb impact damage) exhibited the two-stage failure sequence. The 20 ft-lb midbay damage panel exhibited only catastrophic failure (no arrest). Figure 27 shows that a significant difference exists between initial failure and final panel failure strains. Comparison with coupon test data in Figure 28 shows that initial failure and arrest in the built-up panels correspond to catastrophic failure load in coupons. These data demonstrate significant a damage tolerance configuration scale up effect in built-up structures.

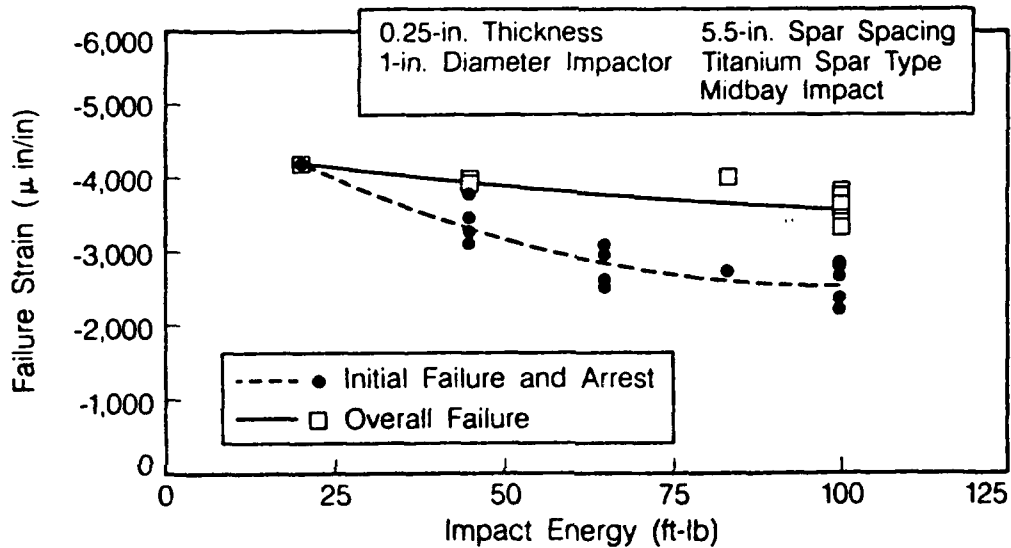


Figure 27 Summary of static strength of impact damaged 3-spar panels - AS4/3501-6

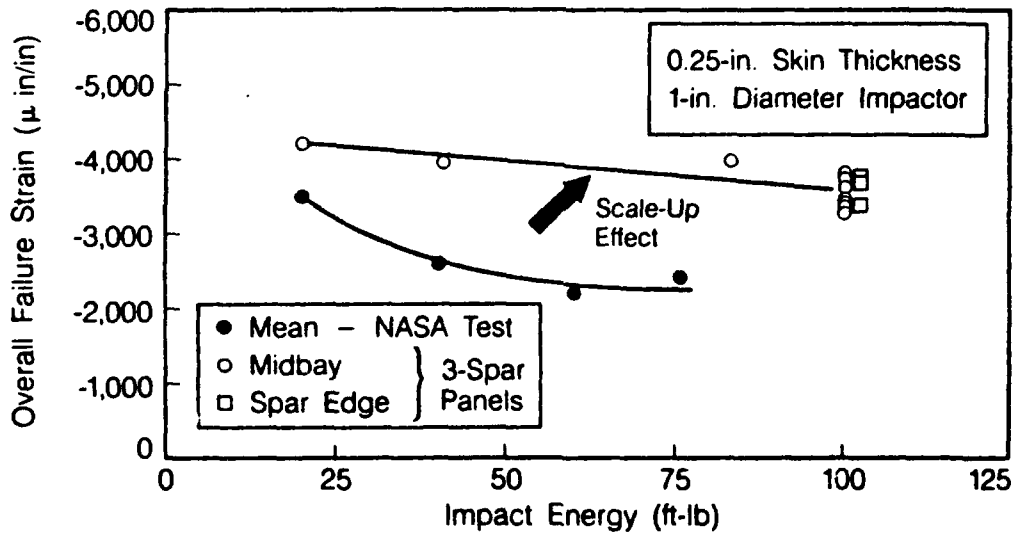


Figure 28 Impact damage tolerance scale-up effect in built-up structure - AS4/3501-6

Lessons Learned

The lessons learned for composite damage tolerance are presented in Figure 29. These lessons learned highlight a conceptual difference in damage tolerance certification for composites and metals. Figures 30 and 31 show the non-inspectable slow damage growth concept for metals and composites, respectively. For metals (Figure 30), residual strength decreases gradually over the aircraft service life as a fatigue crack initiates and grows. Thus, the exposure time where residual static strength is degraded is a small percentage of the total service life. In contrast for composites (Figure 31), residual strength degradation is not gradual, but takes place as a sudden large strength degradation. This occurs for two reasons. First, the impact event is random and can occur with equal probability on either the first or last day of the aircraft service life. Second, the impact event causes an immediate reduction of static strength. This leads to a potentially large exposure time in the degraded strength condition. Figure 32 summarizes this difference for composites and metals.

- Impact Damage Is the Most Severe Defect/Damage Type
- Impact-Damage Areas and Static Strength Are Strongly Dependent on Structural Configuration
- Failure Modes of Impact-Damaged Build-Up Structure Are Significantly Influenced by Structural Configuration
- Significant Impact-Damage Tolerance Scale-Up Effects Exist for Build-Up Structure
- Impact-Damaged Structures Are Insensitive to Fatigue Loading

Figure 29 Summary of composite damage tolerance lessons learned

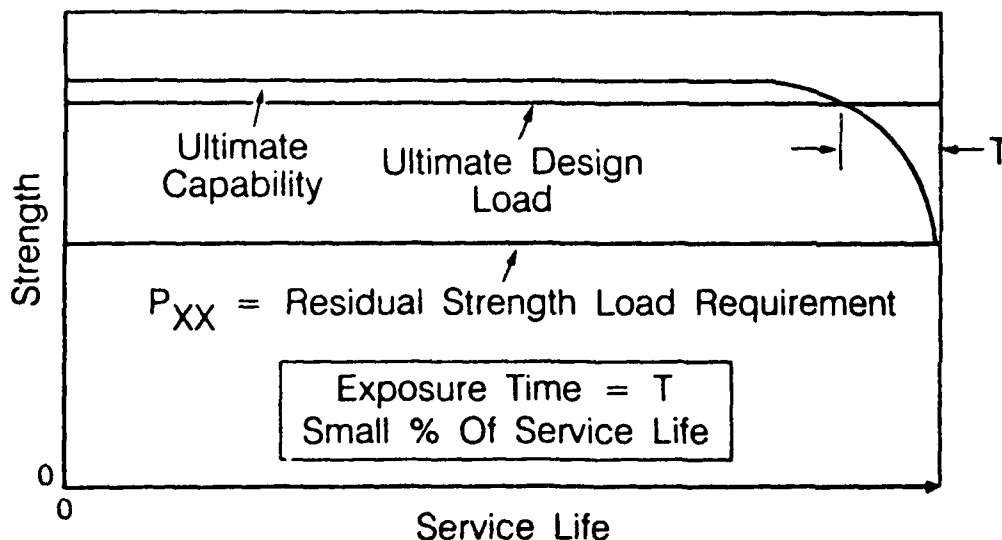


Figure 30 Metallic non-inspectable slow damage growth

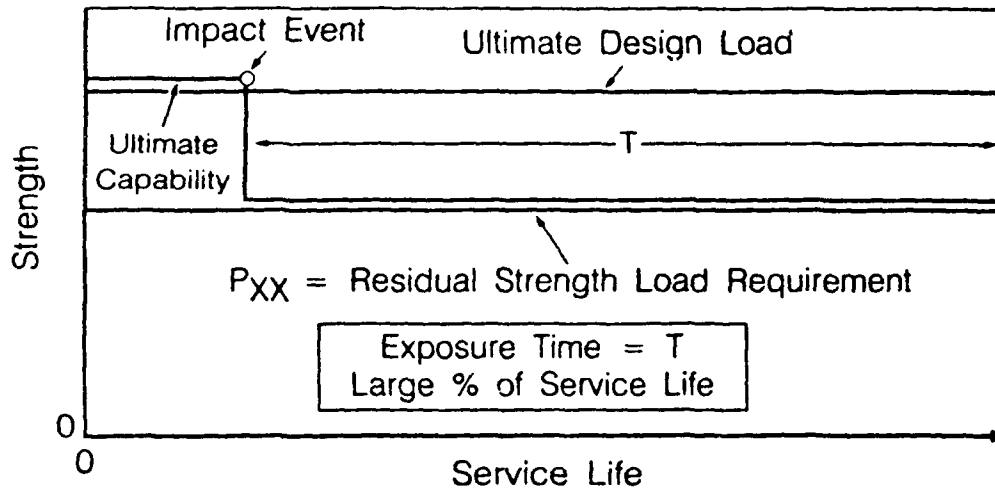


Figure 31 Composite non-inspectable slow damage growth

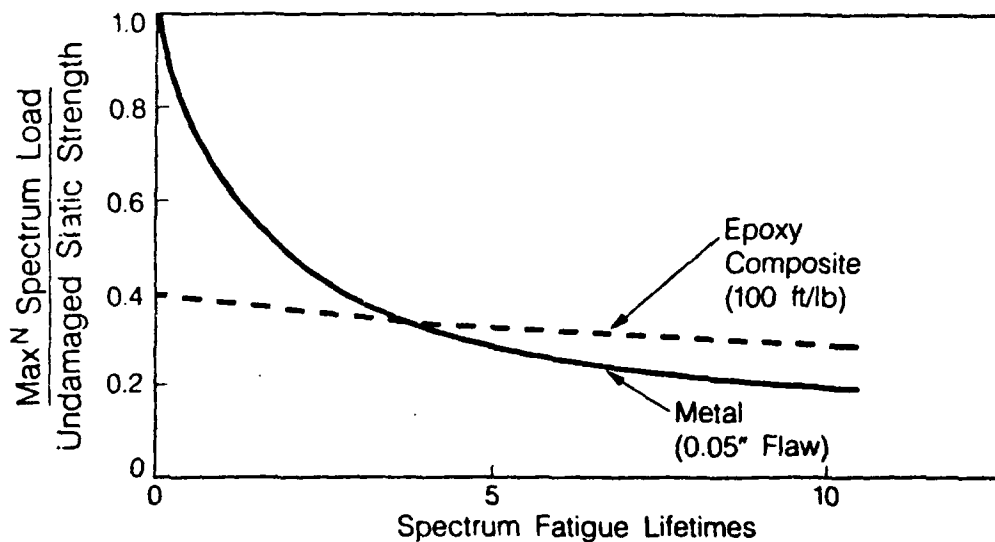


Figure 32 Comparison of composite and metal damage tolerance

Certification Recommendations

The unique features of composite damage tolerance were recognized when draft USAF damage tolerance design requirements were developed in reference (2). The highlights of the draft requirements are presented in Figure 33 and discussed in reference (3). The damage assumptions in the draft requirements are presented in Figure 34. In practice, the impact damage requirement dominates design since it is the most severe. Figure 35 summarizes schematically the impact damage

NEW MATERIALS AND FATIGUE RESISTANT AIRCRAFT DESIGN

- Conceptually Equivalent to MIL-A-83444
- MIL Prime Format per MIL-A-87221
- Recognition of the Unique Property Characteristics of Composites
- Composite Defect/Damage Assumptions Significantly Different From Metals

Figure 33 Highlights of draft USAF composite damage tolerance design requirements

Flaw/Damage Type	Flaw/Damage Size
Scratches	Assume the Presence of a Surface Scratch 4.0-Inch Long and 0.02-Inch Deep
Delamination	Assume the Presence of an Interply Delamination That Has an Area Equivalent to a 2.0-Inch-Diameter Circle With Dimensions Most Critical to Its Location
Impact Damage	Assume the Presence of Damage Caused by the Impact of a 1.0-Inch-Diameter Hemispherical Impactor With 100 ft-lb of Kinetic Energy or With That Kinetic Energy Required To Cause a Dent 0.10-Inch Deep, Whichever Is Least

Figure 34 Damage assumptions in draft USAF damage tolerance design requirements

requirements. Two cut-offs were used: first, an impact energy cut-off equal to 100 ft-lb, which represents, conceptually, a tool-box dropped on its corner from approximately three feet; and second, a visibility cut-off at 0.10-inch dent depth, which represents, conceptually, damage detectable in a visual inspection. Figure 35 shows that the requirements do not potentially cover all non-visible damage; however, the 100 ft-lb impact is considered a conservative and potentially rare event (once per lifetime per aircraft fleet).

The recommended compliance approach for the draft requirements is summarized in Figure 36. First, no significant damage growth is permitted in two design lifetimes. This is recommended because damaged composites have extremely flat S-N curves (Figure 22) and exhibit rapid unstable growth after growth initiation. Thus, it is not possible to control composite damage tolerance using the metal damage growth and inspection philosophy. An advantage of this compliance approach is that it eliminates inspection requirements.

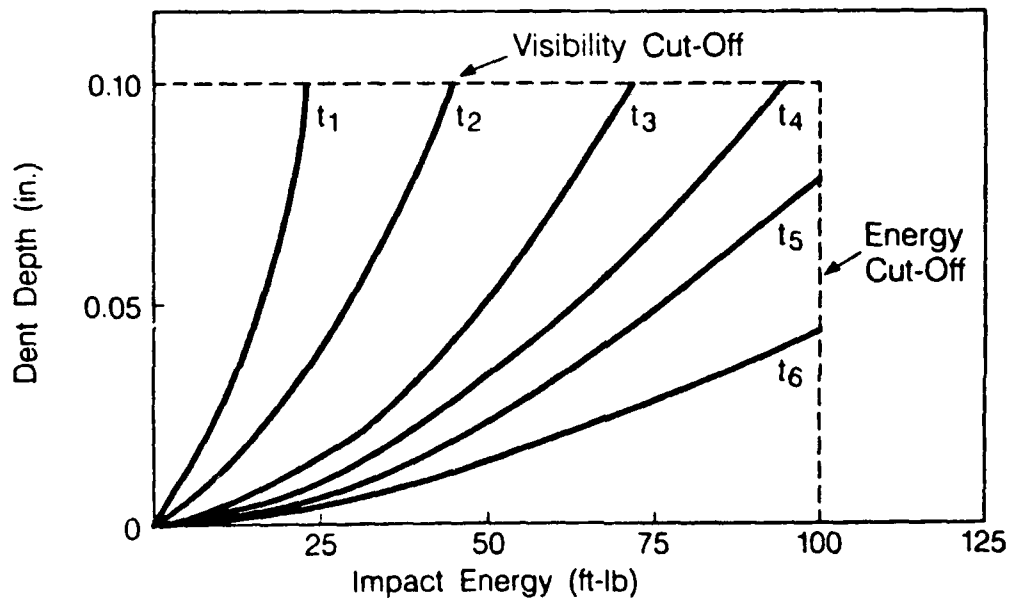


Figure 35 Summary of impact damage assumptions in draft USAF damage tolerance design requirements

- No Significant Damage Growth in Two Design Lifetimes
- No In-Service Inspections Required
- No Full-Scale Test Validation Required

Figure 36 Recommended compliance approach for the draft USAF damage tolerance requirements

Finally, no full-scale test validation is required for composite damage tolerance certification. This is recommended because extensive testing in references (2) and (7) has shown that subcomponent validation tests accurately represent full-scale composite damage tolerance behavior.

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- (2) Whitehead, R.S., and Demuts, E., "Damage Tolerance Qualification of Composite Structures," Proceedings of the USAF Aircraft Structural Integrity Conference, Dayton, OH, November 1985.
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NEW MATERIALS AND FATIGUE RESISTANT AIRCRAFT DESIGN

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- (5) Whitehead, R.S., and Deo, R.B., "A Building Block Approach to Design Verification Testing of Primary Composite Structure," Proceedings of the 24th AIAA/ASME/ASCE/AHS Structures, Structural Dynamics and Materials Conference, Lake Tahoe, NV, May 1983.
- (6) Whitehead, R.S., Ritchie, G.L., and Mullineaux, J.L., "Durability of Composites," presented at the 9th Mechanics of Composites Review, Dayton, OH, October 1983.
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Notes of Presentation on "Civil Aviation Concerns" by Dr. John Bristow

Airworthiness requires that a set of requirements be established as a standard for comparison. A common set of requirements has evolved covering aircraft developed on both sides of the Atlantic. (Vugraph 1)

However, a number of concerns have come up in working with this requirements document. (Vugraph 2)

An argument put forward that 70°C/70-RH conditioning is the same as 84-RH Equilibrium is presented in Slide 4. Moisture computer predictions are shown. Argument being you cannot get to the point in the sky at temperature unless you are in the temperature, but they look at the moisture loss after 1-hour at temperature on the ground. Claim is almost the same as that you would have in 70/70 conditions. This is a current issue for resolution.

Bristow said that the requirements must insist on 54°C testing for small aircraft, 70°C for civil transport. He said that 54°C is necessary because the room-temperature-curing glass epoxy system that they use to make these

light aircraft out of can cope with that temperature. But just above that their shear strength properties drop off markedly. Some contractors have been surprised that materials tested by room-temperature dry test with margin over 25 percent broke before ultimate load when tested in temperature. Impact civil requirement expenses in Vugraph 8. If you can not see the damage, you must be able to withstand ultimate load. If you can see damage you must withstand 2/3 of ultimate limit load.

The major problem of certification for an international project in progress in Europe is shown in Vugraph 13. The project involves nine material systems, eight construction sites in five countries, four specification authorities, and four controlling companies. Thus, there are problems in quality control and specification of materials.

Vugraph 15 shows an example where a limit load test on the foreplane showed a design fault – the spar not being extended to the wing tip. Vugraph 15 shows that there is no substitute for one or two full-scale tests in cyclic load is addressed in some way.

CIVIL AIRCRAFT REQUIREMENTS

INCLUDING GUIDANCE MATERIAL

UNITED KINGDOM

BRITISH CIVIL AIRWORTHINESS REQUIREMENTS

SECTION D AEROPLANES

SECTION G ROTOCRAFT

SECTION K LIGHT AEROPLANES

ARB PAPER ON COMPOSITES

FEB 81

Vugraph 1

EUROPE

JOINT AIRWORTHINESS REQUIREMENTS

JAR 25 TRANSPORT CATEGORY AIRCRAFT

JAR DISCUSSION PAPER ON COMPOSITES

MAY 83

U.S.A.

FEDERAL AVIATION REGULATIONS

PART 23 AIRCRAFT UNDER 12000 LBS

PART 25 AIRCRAFT OVER 12000 LBS

PART 27 HELICOPTERS UNDER 6000

PART 29 HELICOPTERS OVER 6000

ADVISORY CIRCULAR ON COMPOSITES

AC 20-107 1978

RE-DRAFT MAR 83

JOINT ACTION
FAA - JAR
AGREED COMMON
TEXT DEC 84
ISSUED AS
AC20-107A
ACJ25-603

COMPOSITE CONCERNS

Vugraph 2

ENVIRONMENT
 IMPACT
 METAL/COMPOSITE HYBRIDS
 CYCLIC LOADING
 SPECIFICATIONS
 QUALITY CONTROL
 DETAIL DESIGN
 INTERNATIONAL PROJECTS

ENVIRONMENTAL CONSIDERATIONS

Vugraph 3

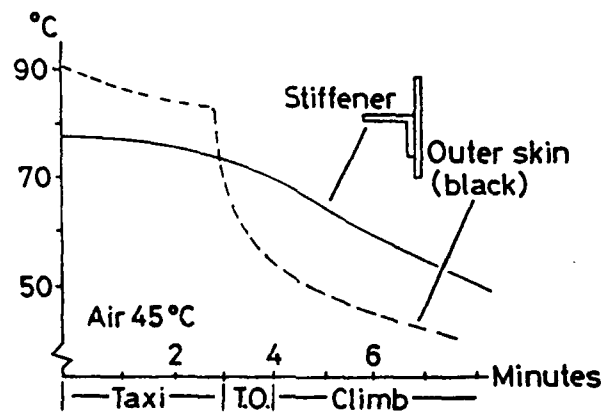
70°/70 RH
 84RH EQUILIBRIUM
 STATIC TEST v CYCLIC TEST
 54°C TESTING

Moisture level after 1 hour in 90°C

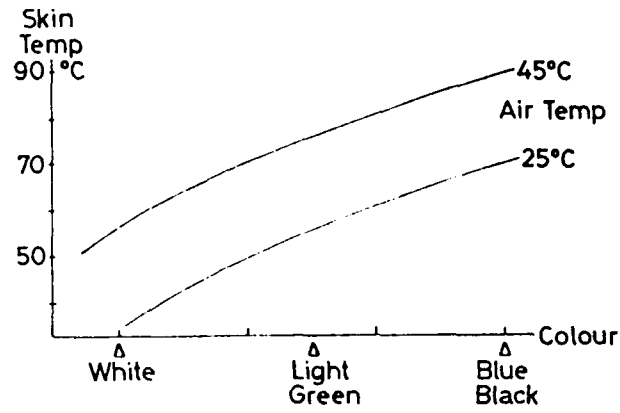
Vugraph 4

<u>120 sys:</u> 2.36mm thick	
Djakarta	1.41 %
	<u>-.15 %</u>
	1.26 %
Saturation in 70% RH	1.18 %
<u>170 sys:</u> 1.93mm thick	
Djakarta	0.800 %
	- .096 %
	0.704 %
Saturation in 70 % RH	0.680 %

Vugraph 5



Vugraph 6

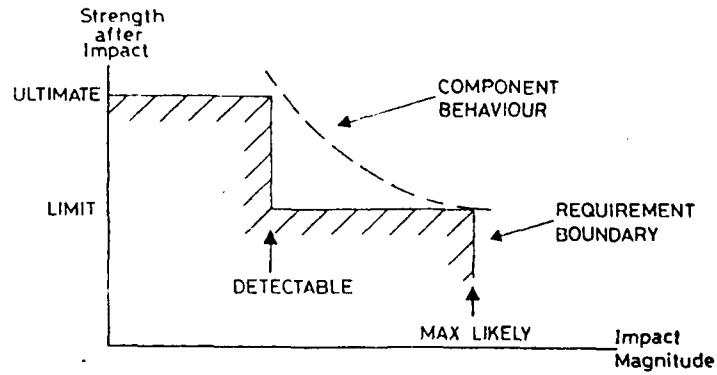


IMPACT

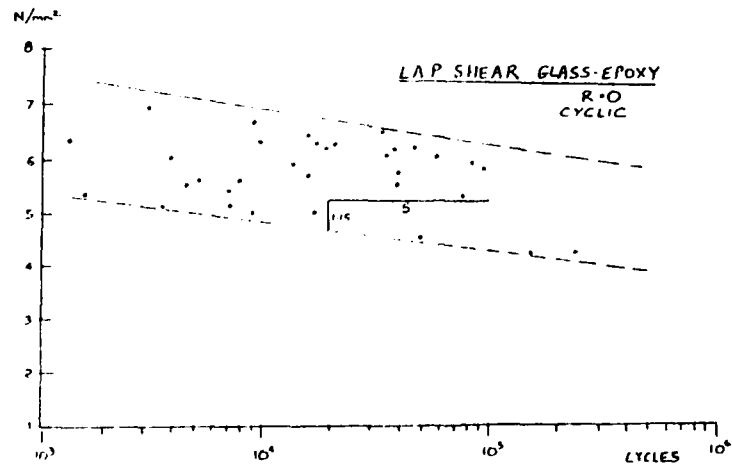
Vugraph 7

Low energy
Bird Strike
Discrete source damage
High energy (crash)

Vugraph 8



Vugraph 9



TYPICAL CONTENTS LIST FOR MATERIAL AND PROCESS SPECIFICATIONS

Vugraph 10

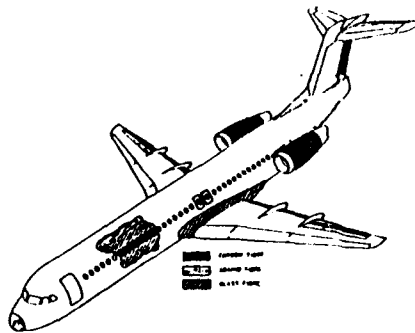
Basic Fibre Properties.
Types, numbers and frequencies of tests.
Basic Matrix Properties and Chemical Characterization.
Types, numbers and frequencies of tests.
Basic Composite Properties
Types, numbers and frequencies of tests.
Condition of Fabrication.
Method, lay-up, tooling, workshop environment.
Significant parameters of cure cycle.
Limits, ranges and recording methods.
Cured component properties.
Types, numbers and frequencies of tests.
Inspection criteria at each stage.
Storage and handling conditions throughout the process.

Proof of Compliance

Vugraph 11

	Emphasis on: →			
	Analysis	Test		
		Detail	Subcomponent	Full-scale
↑ Increasing Importance ↓				
Radome	X			
Wing/fus. fairing	X			
O. wing TR.E. panel	X			
Dorsal fin	X	X		
Floor panels	X			X
Cowlings	X	X	X	X
Rudder	X	X		X
Aileron	X	X		X
Flaps	X	X	X	X

Vugraph 12

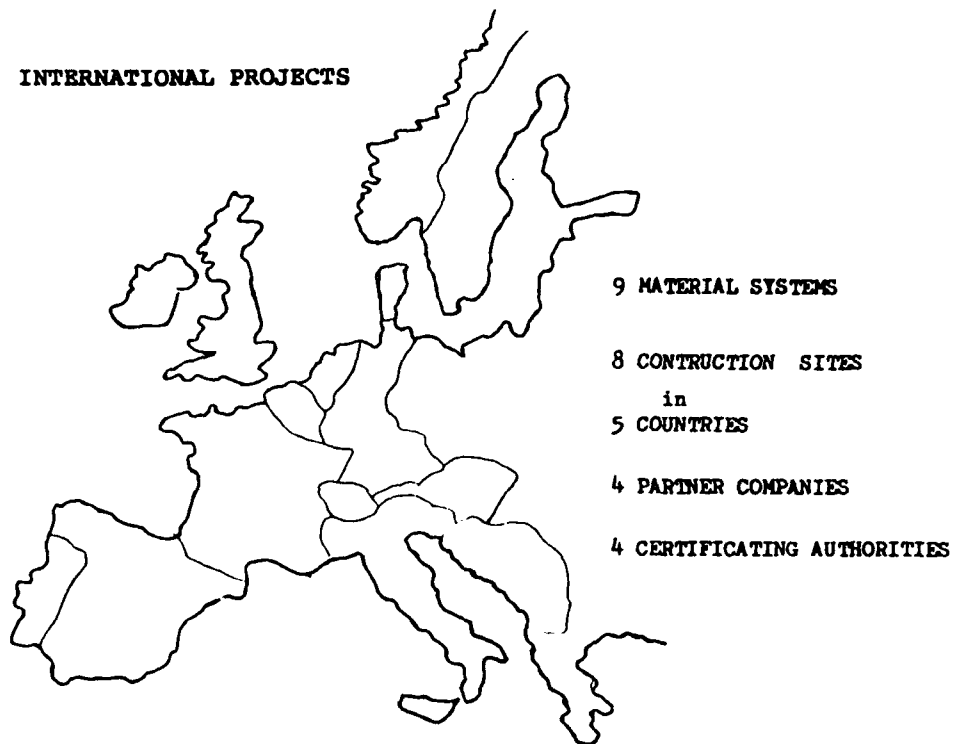


Composite Components

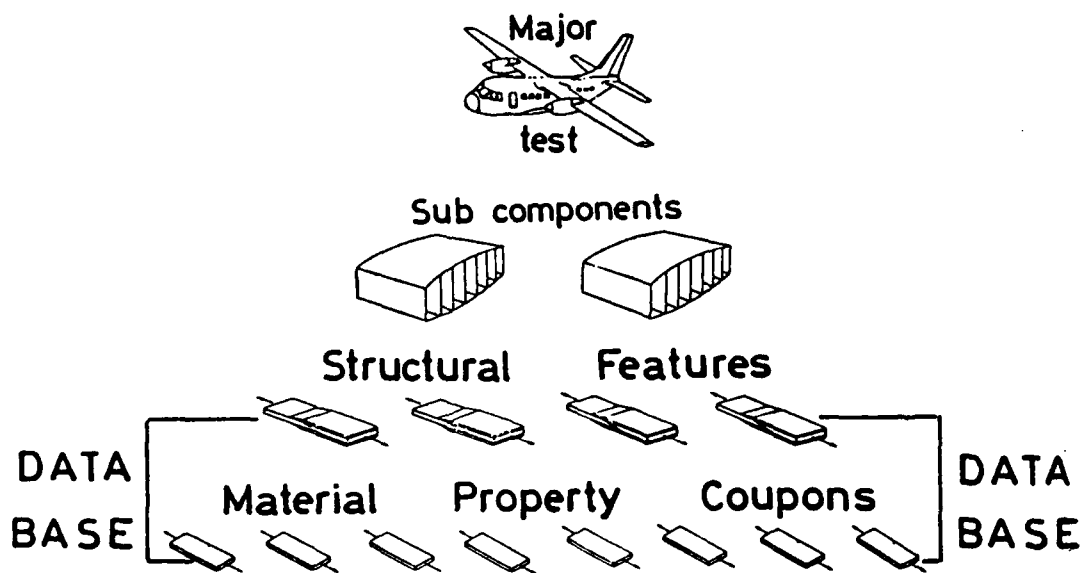
Radome	Sandwich	AFRP/NOMEX
Wing/fus. fairing	Sandwich	AFRP/GFRP/NOMEX
Dorsal fin	Sandwich/solid	AFRP/NOMEX
Tail fairing	Solid	GFRP
Rudder	Sandwich/solid	CFRP/AFRP/NOMEX
Aileron	Sandwich/solid	CFRP/NOMEX/METAL
Flaps	Sandwich/solid	CFRP/NOMEX/METAL
Wing TR.E. panels	Sandwich	AFRP/NOMEX
Cowlings	Sandwich	CFRP/NOMEX
Floor panels	Sandwich	CFRP/GFRP/NOMEX

INTERNATIONAL PROJECTS

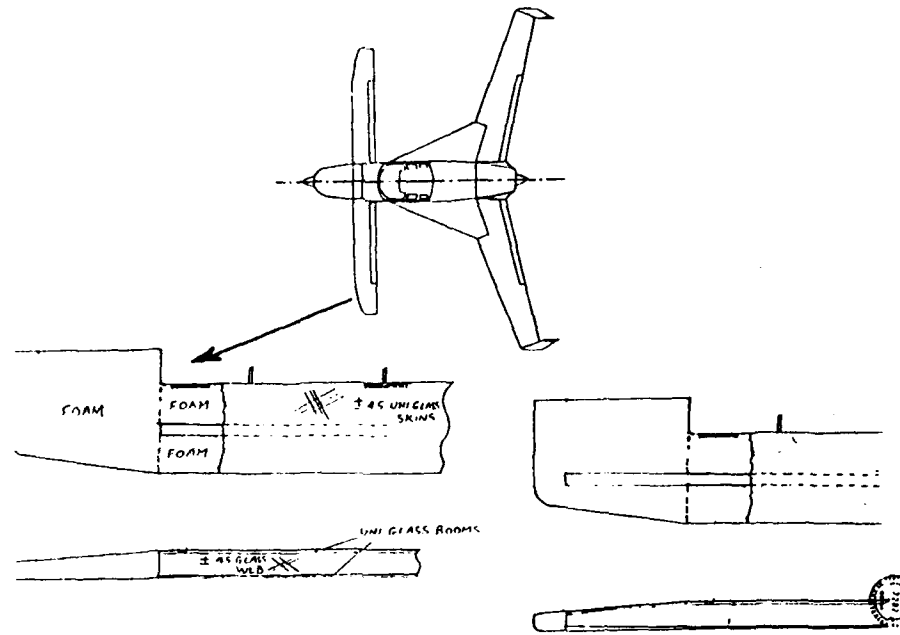
Vugraph 13



Vugraph 14



Viewgraph 15



OTHER CONSIDERATIONS

Viewgraph 16

- Repairs
- Erosion
- Stiffness
- Inspection
- Flammability, smoke and toxicity
- Lightning strikes
- Electromagnetic/electrostatic effects

Notes on Presentation on "IMPACT DAMAGE" by Professor G. Davies, Imperial College, England

- We may be making a few mistakes in dynamic response testing in that we are basing our residual strengths on coupon testing, whereas in real life the structure response locally may be quite different.
- At low velocity, dynamic response can affect the resultant damage.
- A suggested definition of low velocity is that velocity in which the strain has time to react in a fairly simple fashion, not necessarily linear. For epoxy matrix that velocity is approximately 20 m/s.

- For low-velocity analytical prediction, a local model should be incorporated in a complex model test article.
 - For low-velocity, Davies is confident that we can predict the difference between coupon testing and real structural behavior.
 - At high velocity – well above 20 m/s – we look at hydraulic shock. We have produced a numerical design tool that can predict the behavior of complex structures. Experiments show that it seems to work. What we have actually done is to produce a finite element model for liquids and structures. The designers must be careful in modeling cavitation and projectile face for entry.
- We have not found long-term drag pressure problems as some people have suggested.

Vugraph 1

IMPACT DAMAGE

Ad-hoc testing or Prediction?
Damage a function of:

- Composites component materials
- Lay-up
- Impactor mass, shape, velocity
- Dynamic response

Dynamic response even for simple plate (coupon) at low velocity:


$$\epsilon = \frac{V}{C} \cdot \sqrt{\frac{M}{M'}} \cdot \frac{Y}{K} \cdot f\left(t, \frac{M}{M'}\right)$$

Conclusion: Test data base too large.

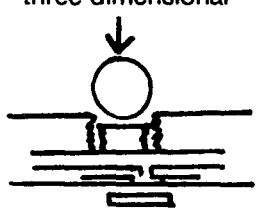
Vugraph 2

LOW VELOCITY or HIGH VELOCITY?

two-dimensional



three-dimensional

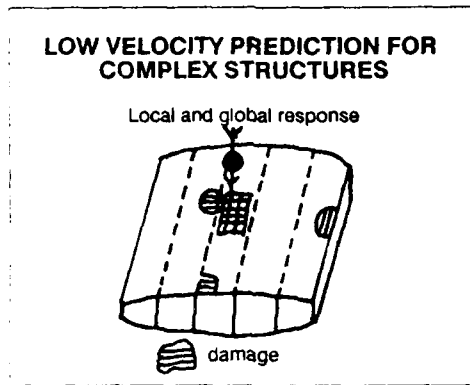


$\epsilon \approx \frac{V}{C}$

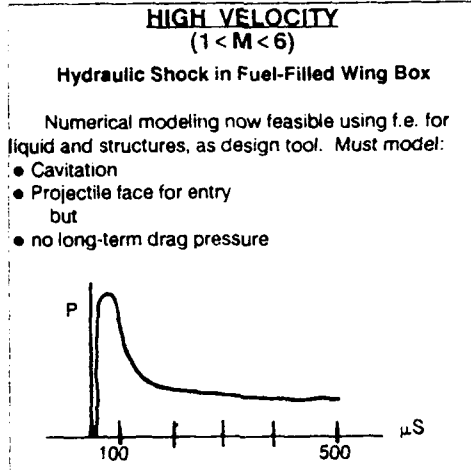
- for epoxy matrix $V = \epsilon C \approx 20\text{m/sec}$
- No strain rate effects.

$\epsilon = f(\text{configuration})$
Sensitivity of thermoplastics to ϵ

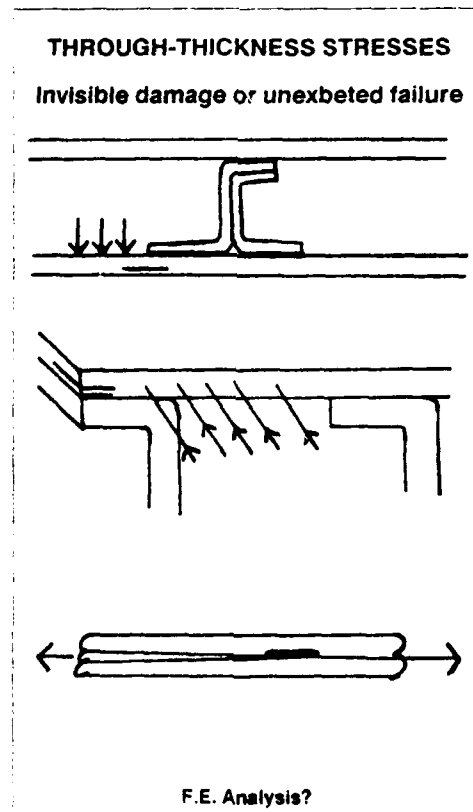
Vugraph 3



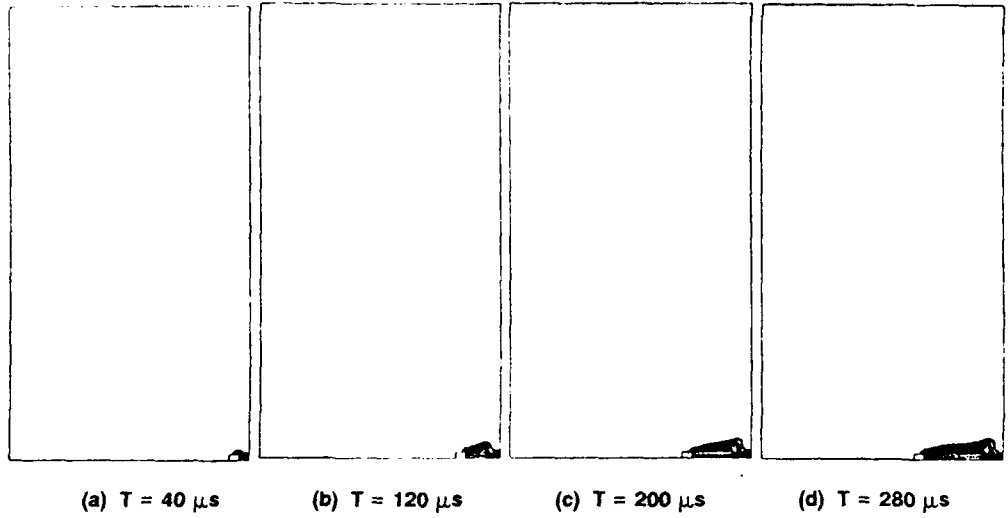
Vugraph 4



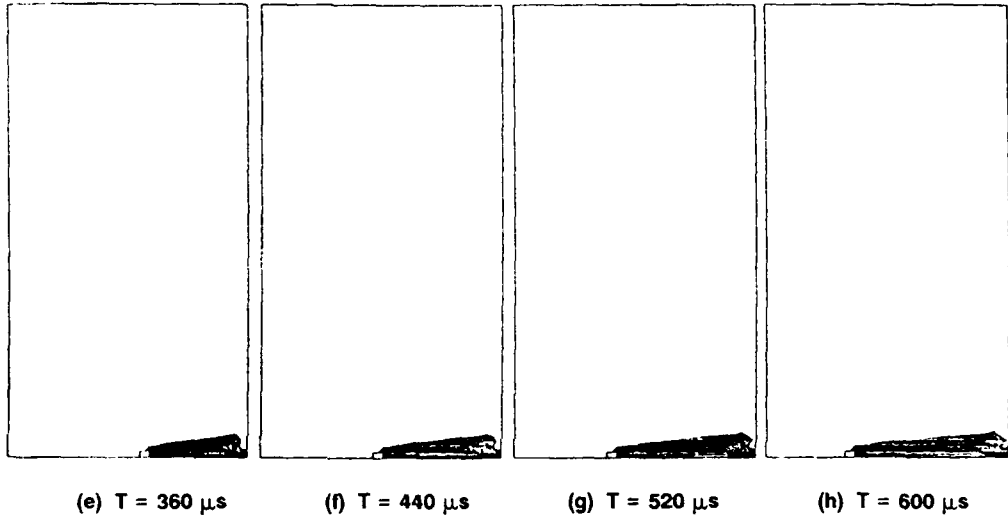
Vugraph 5



Vugraph 6

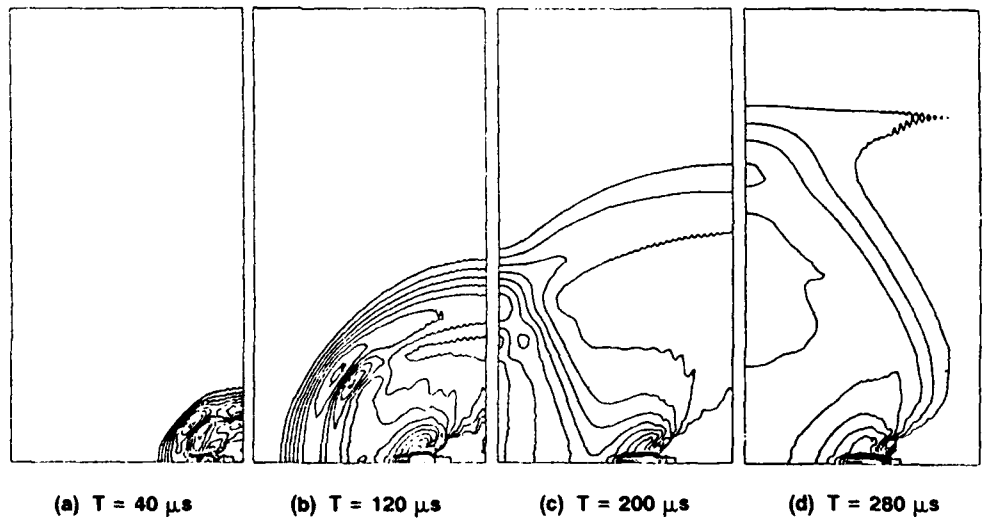


Vugraph 7



**Density Contour Plots - Projectile motion inside a water-filled tank $u = 328$ m/s.
Finite Difference Results**

Vugraph 8



**Pressure Contour Plots - Projectile motion inside a water-filled tank $u = 328$ m/s.
Finite Difference Results**

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Notes on "RAE Composite Certification" by Dr. A.W. Cardrick

Vugraph 1 summarizes the position for UK military aeroplanes and helicopters. Vugraph 2 draws attention to the principal features of the policy.

Tolerance to Impact

Vugraph 3 illustrates the tentative standard blunt and sharp impactors that are intended to be used for gathering data on the tolerance of metal and composite structures to impact and for the certification testing of composites. The background is summarized in Annex A as distributed prior to the meeting.

Fatigue

It was considered that those types of detail with high-static notch sensitivity and properties governed by high-modulus fibres, rather than matrix or bond properties, would usually be insensitive to fatigue. Nevertheless, it was argued that this insensitivity must be demonstrated before fatigue could be dismissed from further consideration. In view of the multiplicity of different loading spectra that must be considered in design and the further complication of hygrothermal cycles, it was accepted that the sensitivity check could be based on test data under less than realistic conditions providing a suitable margin on fatigue strength was demonstrated to allow for this uncertainty. A suitable demonstration might take the form of tests on a few (typically 10 to 15) structural elements representing each type of detail using loading of constant amplitude. The test could be done on elements in the room temperature – as received condition or with prior moisture conditioning. The endurance obtained would be used with Miner's rule, and an adjustment for changes in mean stress under each of the spectra to be considered

in design. Only if the fatigue allowable approached the static allowable (for the corresponding material condition) would further testing under more realistic conditions be necessary. Vugraph 4 illustrates the procedure used in the new UK military requirements ("Tolerance to Fatigue Damage", Chapter 201 Def Stan 00-970, to be published shortly) for allowing for scatter in fatigue performance; the mean curve for a detail is reduced by a factor on life where it is relatively steep and by a factor on stress where it is flat and the life factor has no meaning. These reduced curves are blended together to give a "safe S-N curve" which is used in calculations of the safe life of the detail under any loading spectrum. The factors are derived from tests under spectrum loading.

Vugraph 6 illustrates the mean S-N curve for a bolted joint with low shear transfer through the fasteners in a carbon fiber composite (CFC). The tentative safe S-N was drawn using a life factor of 3 and stress factor of 1.5. Using Miner's rule and an adjustment for mean stress (Gerber parabola reflected about the axis of zero mean stress) the allowable fatigue stress (at limit load) was calculated for Falstaff (European standard combat aircraft spectrum). This was significantly above the corresponding static allowable stress. The allowable fatigue stress for a CFC-bolted joint with high shear transfer through the fasteners, Vugraph 5, and for a bearing-transfer specimen, Vugraph 7, also showed insensitivity to fatigue and therefore would not need to be the subject of testing under more realistic conditions. It is noteworthy that for each of these elements the required safe life of 6000 hours under Falstaff can be obtained with the peak load corresponding to the safe value of static strength (as it happens the same for each element as defined in Vugraph 4). Only if greater lives were required would the fatigue allowables for each element begin to diverge.

Similar exploratory fatigue data are being obtained using elements representing co-bonded CFC structure and for these the fatigue allowable may be overriding.

STRUCTURAL CERTIFICATION OF UK MILITARY AIRCRAFT - DERIVATION OF MATERIALS ALLOWABLES, PROOF OF VALIDITY OF FAILURE CRITERIA AND STRUCTURAL ANALYSIS, MAJOR STRUCTURAL TESTS

Static

Fatigue

1 Allowables, failure criteria and structural analysis

Obtain:

- Mean and scatter data for each failure mode (coupons + elements)
- Failure criteria (elements)
- Validation of allowables + criteria + structural analysis (elements + boxes)

Obtain:

- Constant amplitude S-N data at several values of mean stress (coupons + elements)
- Spectrum loading data on scatter (to give safe S-N factors on life and stress) (coupons and elements)
- Use constant amplitude data to "calibrate" Miner's rule (i.e. value of $\sum \frac{D}{N}$) in estimating life under spectrum loading.
- Use safe S-N and calibrated Miner's rule with design spectrum and structural analysis to calculate fatigue allowable stress for details. If fatigue allowables are near to static value refine data.

Room Temperature as received

plus

Hot and Wet if appropriate

plus

Cold and Dry if appropriate

2 Major Structural tests

Vugraph 1

Aim is to show that degraded structure remains air-worthy (stiffness, control deflections, absence of distress) to highest possible proportion of Design Ultimate load (DUL) and that DUL can be sustained.

Options:

- a. Undegraded test to DUL x factor to allow for absence of hygrothermal degradation (unlikely to be attractive for degradation > 5% unless reserves available due to fatigue design, for example).
- b. Undegraded test to DUL with extensive strain measurement for correlation with structural analysis confirmed by degraded and undegraded tests on instrumented, representative large boxes (likely to be used for degradation up to 15%).
- c. Fully degraded test - greatest effect is Hot and Wet. Accelerated conditioning presents problems of moisture gradients in different thicknesses and can create special corrosion problems (seldom a realistic option).

Aim is to show that weakest structure (scatter) in degraded condition can sustain at least 80% DUL at the end of its life.

Options:

- a. Undegraded test with loads amplified to allow for absence of hygrothermal degradation (unlikely to be attractive unless loads are small in relation to static strength).
- b. Undegraded test without amplification to allow for absence of degradation. Must have extensive strain measurement for correlation with structural analysis. Validity of structural analysis confirmed by degraded (hygrothermal-mechanical cycling) and undegraded tests on instrumented, representative large boxes (likely to be used for degradation up to 15% on strength).
- c. Fully degraded test - a formidable task involving accelerated hygrothermal - mechanical cycles of varying intensity on different parts of the structure (seldom a realistic option).

STATIC

FATIGUE

1. ALLOWABLES, FAILURE CRITERIA 7 STRUCCTURAL ANALYSIS

- MEAN & SCATTER EACH FAILURE MODE (COUPONS)
- VALIDATION OF STRUCTURAL ANALYSIS AND FAILURE CRITERIA (ELEMENTS AND BOXES)
- CONSTANT AMPLITUDE S-N
- DATA ON SCATTER AND ADJUSTMENTS TO MINER'S RULE
- CALCULATION OF FATIGUE ALLOWABLES FOR COMPARISON WITH STATIC VALUES
- IF FATIGUE ALLOWABLE POTENTIALLY OVERRIDING REFINE DATA

Vugraph 2

ALL: ROOM TEMPERATURE/AS RECEIVED
HOT & WET IF APPROPRIATE
COLD & DRY, IF APPROPRIATE

2. MAJOR STRUCTURAL TESTS

OPTIONS:

- UNDEGRADED TEST TO DUL X FACTOR FOR NO DEGRADATION
- UNDEGRADED TEST TO DUL (NO FACTOR) WITH EXTENSIVE STRAIN GAUGING & STRUCTURAL ANALYSIS SUPPORTED BY BOX TESTS DEGRADED & UNDEGRADED
- FULLY DEGRADED TEST
- UNDEGRADED TEST WITH LOADS AMPLIFIED FOR NO DEGRADATION
- UNDEGRADED TEST (NO DEGRADATION FACTOR) WITH EXTENSIVE STRAIN GAUGING, STRUCTURAL ANALYSIS, BOX TESTS DEGRADED AND UNDEGRADED
- FULLY DEGRADED TEST

Vugraph 3

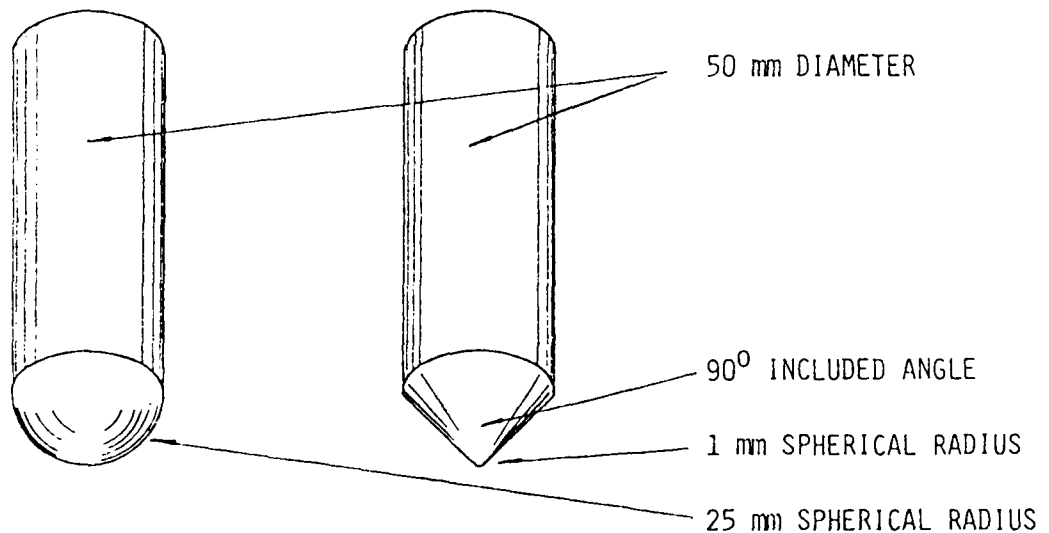
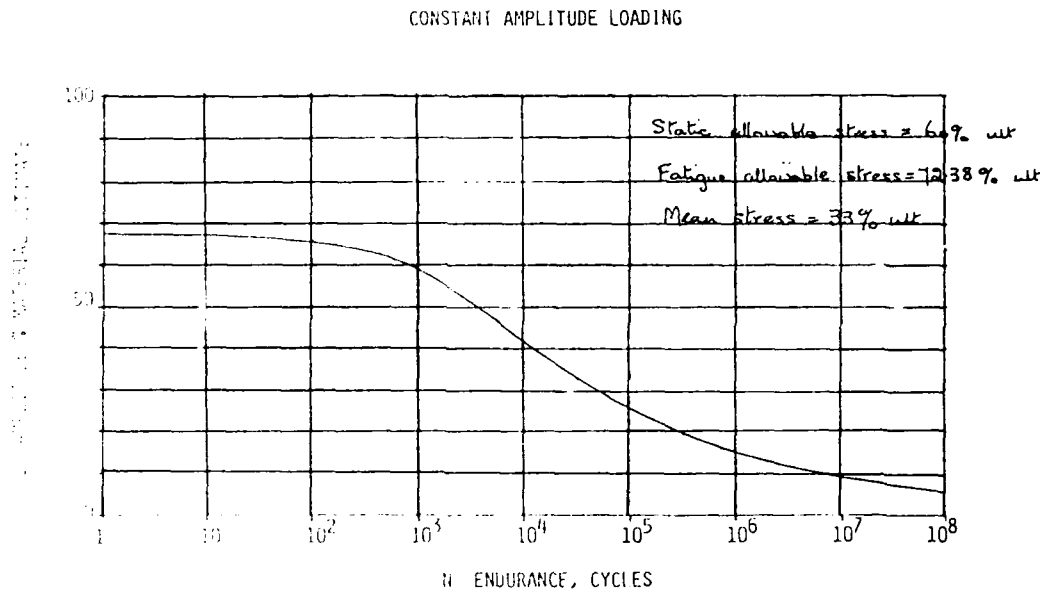
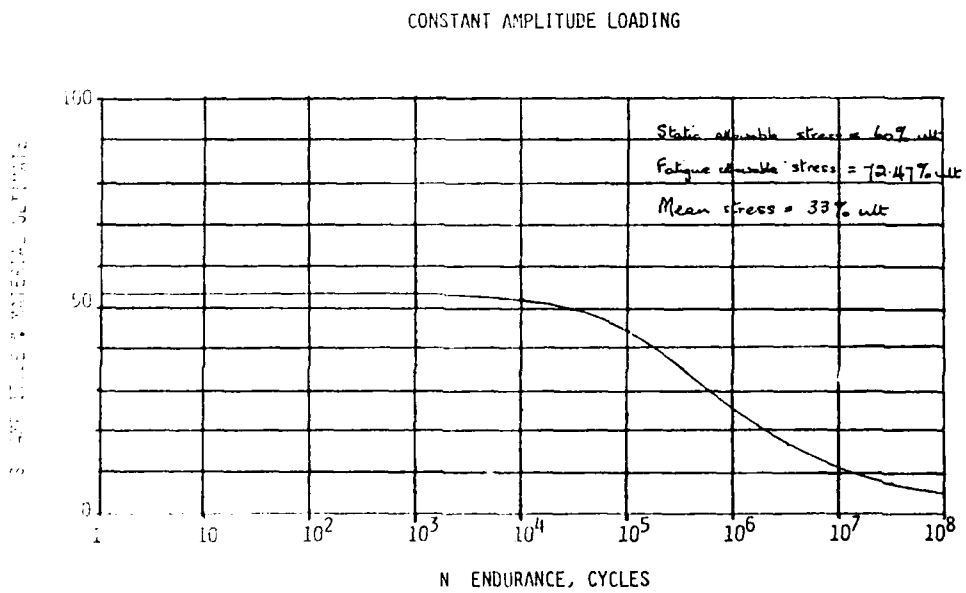


FIG 2 TENTATIVE STANDARD IMPACTORS

Vugraph 6



Vugraph 7



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The effect of observed climatic conditions on the moisture equilibrium level of fibre-reinforced plastics

T.A. COLLINGS

(Royal Aircraft Establishment, UK)

Calculations have been made to determine the moisture absorption behaviour of fibre-reinforced epoxy-matrix composites after exposure to prescribed outdoor climatic environments. A review of world-wide meteorological conditions has been made and six environments were chosen to represent a variety of regions. The results suggest that the level of moisture currently assumed to be absorbed by a composite during the service life of an aircraft is too low and that a much higher value should be used. Since the absorption kinetics of resin matrices differ widely and also change with physical ageing, the validity of specifying a moisture level to define the degree of environmental degradation in structural assessment is questioned. An alternative criterion, a constant relative humidity environment that will produce a representative moisture level in all parts of the structure and for all matrices currently in use, is proposed. Using this philosophy it is suggested that the world-wide worst environment might best be simulated by a constant humidity of 84%.

Key words: *composite materials; water-absorption tests; environmental testing; climatic conditions; relative humidity; carbon fibres; epoxy resins*

The use of fibre-reinforced plastics (FRP) in aircraft structures is increasing as new generations of aircraft take advantage of their attractive structural properties. However, a disadvantage is their readiness to absorb moisture, with consequent degradation¹ of those strength properties of the composite which are matrix dependent, particularly at high temperatures. Allowance has to be made for this degradation effect during the airworthiness substantiation, either by the use of extra strength factors in design or by testing to establish the true strength of components in a degraded state. The degradation that these structures will see in service is linked with the level of moisture absorbed during the service life of the component, usually about 25 years. Thus there is a need for a realistic assessment to be made of the amount of moisture likely to be absorbed during the service life of an aircraft (currently assumed to be 1.0%), and the necessary artificial ageing to reproduce this.

The need to secure agreement on an international collaborative project has led to a re-assessment of the validity of this value. A survey has therefore been made of world-wide climatic records to select the temperature and humidity data necessary for the calculation of moisture equilibrium levels for FRP. Six locations have been chosen to represent temperate regions, such as UK and mainland Europe, hot humid coastal equatorial areas and Middle East regions with variable seasonal climates.

The effects of these climates on total moisture level and distribution are reported here for various thicknesses of laminate freshly made from carbon fibre-reinforced plastic (CFRP) (XAS/913 carbon fibre/epoxy resin system). These moisture levels are then used in the reverse calculation to give the steady-state artificial ageing conditions which will simulate the moisture levels typical of natural climates.

Table 1. Weather data for Port Harcourt, Nigeria

Day time period (hours)	Jan		Feb		Mar		Apr		May		June		July		Aug		Sep		Oct		Nov		Dec	
	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH
0-2	22	90					23	92					23	93					23	94				
2-4	21	92					22	93					22	94					22	95				
4-6	21	94	95		95		22	95	95		95		22	95	95		95		22	96	95		95	
6-8	21	94					23	94					22	94					22	96				
8-10	24	85	81		83		26	85	82		85		26	89	86		87		27	90	85		82	
10-12	29	75	60		67		31	74	73		76		28	80	75		75		29	79	71		69	
12-14	31	63					32	70					29	78					29	76				
14-16	31	60	58		68		31	70	75		80		28	81	80		84		29	78	74		61	
16-18	30	63					30	74					26	84					27	82				
18-20	28	75					28	79					24	88					25	88				
20-22	24	84					25	85					23	90					24	91				
22-24	22	89					24	89					23	91					23	93				

$D_{KAV} \times 10^7$ ($\text{mm}^2 \text{ s}^{-1}$)	0.159			0.166			0.153			0.157
M_{KAV} (%)	1.45			1.55			1.73			1.73
Number of days	31	28	31	30	31	30	31	31	30	31

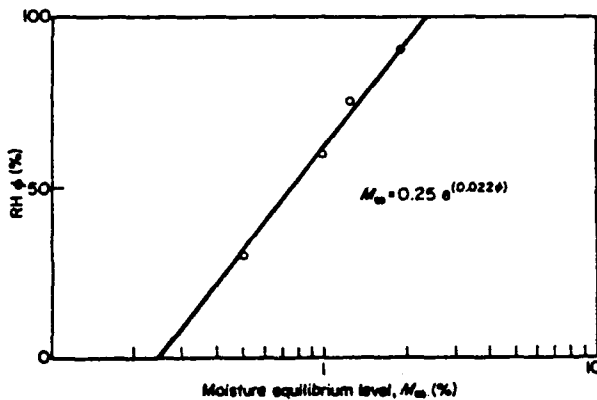


Fig. 1 Variation of moisture equilibrium level with relative humidity for a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$)

DETERMINATION OF MOISTURE CONSTANTS

In order to calculate the level and distribution of moisture in a laminate, two constants must be known: the equilibrium level, which is the maximum amount of moisture that can be absorbed at a given humidity, and the diffusion coefficient, which determines the rate of uptake of water by a laminate.

Equilibrium level

The equilibrium level M_∞ depends only on the concentration of moisture or relative humidity (RH) of the environment and is generally assumed to be independent of temperature. Experimental work done by Collings and Copley² has established values of M_∞ for different RH levels on freshly made XAS/913, carbon fibre/Ciba Geigy epoxy resin laminates, and these are presented in Fig. 1 and Table 1.

Diffusion coefficient

The diffusion coefficient, D , is normally assumed to depend only on absolute temperature, T , as:

$$\log_e D \propto \frac{1}{T} \tag{1}$$

although the effect of applied stress has yet to be established.

Under conditions of steady-state temperature and humidity, the moisture uptake or loss in a composite can be expressed as a percentage of the original dry weight using:

$$M = \left(\frac{W_i - W_d}{W_d} \right) 100 \tag{2}$$

where M is the percentage moisture uptake or loss, W_d is the original dry weight of the specimen and W_i is the weight of the specimen after a time t_i .

The diffusion coefficient is defined by the equation:³

$$D = \pi \left(\frac{h}{4M_\infty} \right)^2 \left(\frac{M_1 - M_2}{\sqrt{t_1} - \sqrt{t_2}} \right)^2 \tag{3}$$

where M_1 and M_2 are the percentages of water uptake at times t_1 and t_2 respectively, h is the laminate thickness, and M_∞ is the moisture equilibrium level for a given relative humidity. The term:

$$\left(\frac{M_1 - M_2}{\sqrt{t_1} - \sqrt{t_2}} \right) \tag{4}$$

is the slope of the linear portion of the plot of M against \sqrt{t} .

The value of D in Equation (3) is obtained by

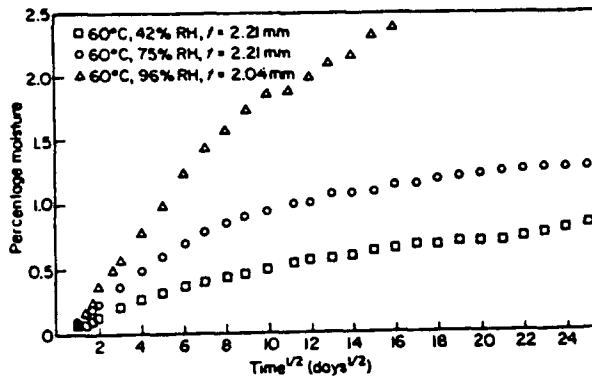


Fig. 2 Moisture uptake in a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$, $b = 50$ mm, $l = 50$ mm) exposed on both surfaces at a constant temperature of 60°C

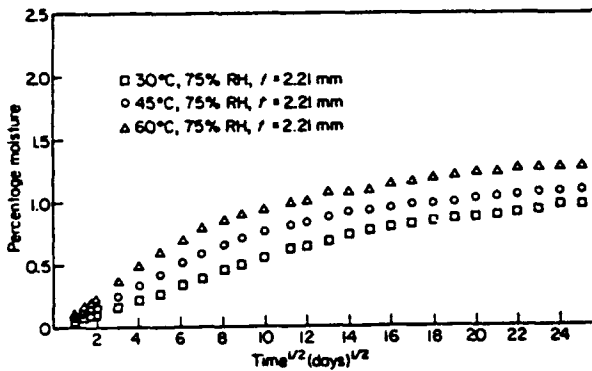


Fig. 3 Moisture uptake in a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$, $b = 50$ mm, $l = 50$ mm) exposed on both surfaces at a constant relative humidity of 75%

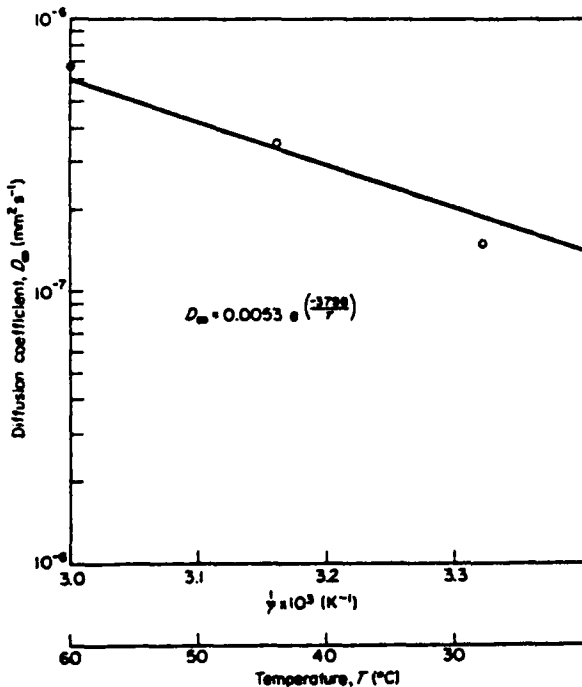


Fig. 4 Arrhenius plot of moisture for a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$)

measuring the moisture absorption of a specimen of finite size and will therefore include moisture diffusion from all six surfaces. To obtain the true one-dimensional diffusion coefficient D_m , needed later for the calculation of the through-thickness moisture distribution, a correction factor given by Shen and Springer² can be used, namely:

$$D_m = D \left(1 + \frac{h}{b} + \frac{h}{l} \right)^{-2} \quad (5)$$

where b and l are the laminate breadth and length respectively.

Experimental results of moisture uptake against the square root of time for the XAS/913 fibre/resin system² are given in Figs 2 and 3. The Arrhenius plot of moisture diffusion for the same fibre/resin system is given in Fig. 4.

CLIMATIC DATA

A survey was carried out of world-wide climatic⁴⁻⁷ records to establish what are the most severe environmental conditions likely to be met by composite materials in service. Six geographical locations were chosen and these are:

● Port Harcourt, Nigeria ⁶	0451 N	0701 E:
● Schleswig, Germany ⁴	5431 N	0931 E:
● Woodbridge, UK ²	5205 N	124 E:
● Singapore ⁷	0016 N	10043 E:
● Guam ⁵	1334 N	14455 E: and
● Bahrain ⁴	2616 N	5037 E:

Tables of typical daily cycles of temperature and humidity, for particular months of the year, are given in Tables 1 to 6.

DETERMINATION OF KINETIC AVERAGES

In an outdoor environment aircraft are subject to continuously varying temperature and relative humidity and as a result there will be periods of exposure that will produce moisture absorption and moisture desorption in an FRP laminate. The long-term net effect of these changes is one of moisture absorption until a quasi-equilibrium moisture condition is reached. This equilibrium level can be related to an average steady-state value of RH.

One approach to establishing this steady-state value has been to calculate arithmetic averages of RH, but this value has been shown to be far from realistic for some environments.⁸ Due to the variation of temperature the diffusion coefficient will not remain constant, therefore the diurnal and seasonal correlation between RH and temperature will have to be considered when choosing an averaging procedure.

Work by Augl and Berger⁴ has shown that weighted temperature and relative humidity averages (T_{KAV} and RH_{KAV}) can be calculated to give the same result as that of a variable environment. However in the calculation of moisture diffusion it is necessary to work in terms of the moisture concentration level M instead of RH. For this reason it is necessary to know the relationship $M = M(RH)$ and also the relationship $D = D(T)$. These relationships are given in Figs 1 and 4, respectively, for the XAS/913 composite material considered here.

Table 2. Weather data for Schleswig, Germany

Day time period (hours)	Jan		Feb		Mar		Apr		May		June		July		Aug		Sept		Oct		Nov		Dec	
	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH
03	0	91	-1	91	1	89	4	91	8	90	11	92	13	92	13	93	11	92	8	93	4	92	2	93
06	0	91	-2	91	0	90	4	90	8	87	12	88	14	90	13	93	10	93	7	94	4	92	2	94
09	0	90	-1	90	2	85	7	75	12	71	16	71	17	75	17	80	14	82	9	89	4	92	2	93
12	1	87	0	85	4	75	9	64	14	62	17	64	19	67	18	71	16	71	11	78	6	87	3	90
15	1	87	1	82	5	72	10	62	14	61	17	63	19	66	19	69	16	70	11	78	6	86	3	90
18	1	89	0	88	3	80	8	70	13	66	16	68	18	70	17	76	14	80	10	88	5	90	3	92
21	0	90	-1	90	2	86	6	84	10	82	13	83	15	85	15	89	12	88	9	91	5	91	3	93
24	0	90	-1	90	1	88	4	89	8	88	11	90	14	90	14	92	11	91	8	92	4	92	2	93
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$D_{KAV} \times 10^7$ (mm ² s ⁻¹)	0.049	0.047	0.054	0.067	0.082	0.096	0.105	0.103	0.091	0.076	0.061	0.057												
M_{KAV} (%)	1.78	1.75	1.55	1.40	1.33	1.38	1.44	1.55	1.56	1.73	1.82	1.90												
Number of days	31	28	31	30	31	30	31	31	30	31	30	31												

Table 3. Weather data for Woodbridge, UK

Day time period (hours)	Jan		Feb		Mar		Apr		May		June		July		Aug		Sep		Oct		Nov		Dec	
	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH
0-2	5	83	4	82	4	81	6	81	9	82	12	82	14	83	15	82	13	81	11	81	7	81	5	81
2-4	5	83	4	82	4	81	6	82	8	83	11	83	14	84	14	82	12	82	10	82	6	81	5	81
4-6	5	83	3	82	4	81	6	81	9	82	12	81	14	82	15	81	12	82	10	82	6	82	5	81
6-8	5	83	3	82	4	81	7	79	11	77	14	75	16	77	16	79	13	80	11	81	7	82	5	82
8-10	5	83	4	80	6	76	8	71	12	70	16	67	18	69	18	70	15	73	12	77	7	79	5	81
10-12	6	81	6	78	7	71	10	65	14	64	17	62	19	63	19	64	16	69	13	72	8	76	5	80
12-14	6	79	7	74	8	66	11	61	15	61	18	59	20	60	20	60	18	62	15	68	9	75	6	78
14-16	6	79	6	75	8	67	10	62	14	62	18	59	20	61	20	61	17	64	14	70	8	76	6	79
16-18	6	81	6	77	7	69	9	65	13	64	17	62	19	63	19	64	16	67	13	73	8	79	5	80
18-20	6	82	4	80	6	74	8	70	12	69	16	67	18	68	18	70	15	74	12	78	7	80	5	81
20-22	5	83	4	80	6	77	7	75	11	75	14	74	16	75	16	76	14	77	11	79	7	81	5	81
22-24	5	83	4	81	5	79	6	78	10	80	12	79	15	80	15	79	13	79	11	80	7	81	5	81
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$D_{KAV} \times 10^7$ (mm ² s ⁻¹)	0.063	0.061	0.064	0.071	0.085	0.099	0.109	0.11	0.097	0.086	0.071	0.062												
M_{KAV} (%)	1.515	1.432	1.307	1.232	1.228	1.184	1.219	1.227	1.276	1.355	1.435	1.469												
Number of days	31	28	31	30	31	30	31	31	30	31	30	31												

Weighted averaging is achieved^a by breaking down the 24-hour daily weather data into j time steps ($i = 1$ to j) with an arbitrarily chosen value of absolute temperature \bar{T} and surface moisture concentration M_i , during which it is assumed that \bar{T} and M_i remain constant. For each time period (24/ j) an equivalent time τ_i can be calculated that will give the same moisture absorption at temperature \bar{T} and surface moisture concentration M_i , as that of the natural climate at temperature T_i . The equivalent time is given by:

$$\tau_i = \frac{24}{j} \frac{D(T_i)}{\bar{D}} \quad (6)$$

where $\bar{D} = D(\bar{T})$.

Thus, the effect of the daily variations of weather can

be achieved approximately by exposing the composite material for a time period of:

$$\tau = \sum_{i=1}^j \tau_i \text{ hours} \quad (7)$$

at a temperature of \bar{T} and with a surface moisture concentration of:

$$M_{KAV} = \frac{\sum_{i=1}^j \tau_i M_i}{\tau} \quad (8)$$

Thus, as shown above, the exposure of τ hours at a

Table 4. Weather data for Singapore

Day time period (hours)	Jan		Feb		Mar		Apr		May		June		July		Aug		Sep		Oct		Nov		Dec	
	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH
0-2	24	91					26	92					26	88					25	92				
2-4	24	91					26	93					26	88					25	92				
4-6	24	92					26	95					26	90					25	93				
6-8	25	87					28	85					27	85					27	85				
8-10	28	76					30	74					29	76					29	73				
10-12	29	69					30	68					30	71					30	68				
12-14	29	70					30	70					30	70					30	69				
14-16	28	72					29	73					29	72					29	71				
16-18	26	80					28	80					28	78					27	80				
18-20	25	85					27	85					27	83					26	85				
20-22	25	88					27	88					26	85					26	88				
22-24	24	90					26	92					26	87					26	90				
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$D_{KAV} \times 10^7$ (mm ² s ⁻¹)	0.156						0.174						0.172						0.169					
M_{KAV} (%)	1.53						1.56						1.49						1.53					
Number of days	31		28		31		30		31		30		31		31		30		31		30		31	

Table 5. Weather data for Guam

Day time period (hours)	Jan		Feb		Mar		Apr		May		June		July		Aug		Sep		Oct		Nov		Dec			
	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH		
0-2	25	84	24	83	25	84	25	85	26	85	26	86	26	88	26	88	26	89	25	88	26	87	26	85		
2-4	24	84	24	83	24	84	25	85	25	85	25	86	25	88	25	89	25	89	25	89	25	89	26	87	25	85
4-6	24	84	24	83	24	84	25	85	26	85	26	86	26	88	26	89	25	89	25	88	26	87	25	85		
6-8	25	83	24	83	25	84	25	83	26	83	26	85	26	87	26	88	26	88	26	88	26	86	26	84		
8-10	26	80	26	79	26	79	26	79	27	79	27	80	27	82	27	83	27	84	27	83	27	83	27	81		
10-12	26	78	26	77	27	76	27	76	27	76	28	78	27	79	28	80	28	81	28	81	28	80	27	79		
12-14	27	76	27	75	27	75	28	75	28	75	29	76	28	78	28	78	28	80	28	80	28	79	28	78		
14-16	27	77	27	76	27	76	27	76	28	82	28	77	28	79	28	79	28	81	28	81	28	80	27	79		
16-18	26	79	26	78	26	78	27	78	27	78	28	79	27	81	27	81	27	83	27	83	27	82	27	81		
18-20	25	82	25	81	25	81	26	82	26	81	27	82	27	84	26	84	26	86	26	86	26	85	26	83		
20-22	25	83	25	82	25	82	26	83	26	83	26	84	26	86	26	86	26	87	26	87	26	86	26	84		
22-24	25	83	25	82	25	82	25	84	26	84	26	85	26	87	26	87	26	88	26	88	26	86	26	84		
<hr/>																										
$D_{KAV} \times 10^7$ (mm ² s ⁻¹)	0.156	0.156	0.136	0.161	0.165	0.168	0.165	0.143	0.165	0.165	0.165	0.167	0.163	0.162	1.58	1.53										
M_{KAV} (%)	1.48	1.45	1.44	1.48	1.48	1.52	1.58	1.57	1.63	1.62	1.58	1.53														
Number of days	31	28	31	30	31	30	31	31	30	31	30	31														

temperature of \bar{T} with a surface moisture concentration M_{KAV} is equivalent to exposure for 24 hours with a diffusion coefficient:

$$D_{KAV} = \frac{\bar{D}t}{24} \tag{9}$$

and with a surface moisture concentration of M_{KAV} . The temperature of exposure is T_{KAV} which is the temperature that corresponds to D_{KAV} so that $D_{KAV} = D(T_{KAV})$.

MOISTURE PREDICTION

Moisture prediction can be made based on the classical theory of diffusion described by Fick's second law, ie at time t .

$$\frac{\partial c}{\partial t} = \frac{\partial(D\partial c)}{\partial x(\partial x)} \tag{10}$$

where c is the moisture concentration, and x is the through-thickness location.

For composites using epoxy matrices, the diffusion

Table 6. Weather data for Bahrain (Muharrag)

Day time period (hours)	Jan		Feb		Mar		Apr		May		June		July		Aug		Sep		Oct		Nov		Dec	
	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH	°C	RH
0-2																								
2-4																								
4-6	14	78	15	76	18	69	21	62	26	59	28	56	30	57	30	61	28	60	25	68	21	71	16	76
6-8																								
8-10																								
10-12	20	66	21	63	24	57	29	52	33	51	36	49	37	50	38	53	36	54	32	56	28	61	22	66
12-14																								
14-16																								
16-18	19	76	20	74	23	71	27	66	32	64	34	63	35	67	36	70	34	70	31	70	26	71	21	77
18-20																								
20-22																								
22-24	16	81	17	80	20	76	23	74	28	72	30	69	32	72	32	77	30	77	27	79	23	77	18	82
$D_{KAV} \times 10^7$ ($mm^2 s^{-1}$)	0.111	0.112	0.132	0.136	0.190	0.183	0.221	0.226	0.210	0.182	0.152	0.121												
M_{KAV} (%)	1.306	1.251	1.125	1.018	0.973	0.934	0.977	1.06	1.06	1.127	1.16	1.31												
Number of days	31	28	31	30	31	30	31	31	30	31	31	30	31	30	31	30	31	30	31	30	31	30	31	31

coefficient is taken to be independent of concentration and hence of through-thickness location, so Equation (10) can be written in the form:

$$\frac{\partial c}{\partial t} = D \frac{\partial^2 c}{\partial x^2} \quad (11)$$

It is more practical to work in terms of moisture content M , which is defined as the moisture per unit volume of a composite expressed as a percentage of the dry weight (see Equation (2)). Thus a more useful form of Equation (10) can be written as:

$$\frac{\partial M}{\partial t} = D \frac{\partial^2 M}{\partial x^2} \quad (12)$$

This equation can be solved using a finite difference method such as that described by Copley.⁹ The method used in Reference 9 for calculating the through-the-thickness moisture distribution has been shown⁹ to agree very well with the distribution obtained experimentally using the method described in Reference 10.

Table 7. Equilibrium levels

Geographical location	Equilibrium level (%)
Port Harcourt	1.6
Schleswig	1.5
Woodbridge	1.3
Singapore	1.53
Guam	1.55
Bahrain	1.2

MOISTURE MODELLING RESULTS

The first step in the modelling procedure was to determine M_{KAV} and D_{KAV} using Equations (6) to (9) with $\bar{D} = D(293 K)$ for the data given in Tables 1 to 6 for each of the six geographical locations. With these values of M_{KAV} and D_{KAV} and using the Copley⁹ computer program to model moisture diffusion, estimates were made for the quasi-equilibrium moisture level that would be reached in a 2 mm thick laminate for each of the six locations; these are given in Table 7. The effect of monthly or quarterly variations in the environment on the through-thickness moisture distribution were computed, in most cases, for a 1 mm, 2 mm and 3 mm thick, infinite length (l) and width (w) laminate exposed on both surfaces. The results are plotted in Figs 5 to 12. Only the extreme distributions have been labelled in each figure. Other distributions can be identified by reference to the M_{KAV} value in the appropriate weather data table.

DISCUSSION

The temperatures and humidities used here for the calculation of moisture equilibrium levels have been taken from records of climatic data¹⁻⁷ with no account being taken of the effects of direct exposure to solar radiation, surface protective coatings or flight mission type and frequency.

Exposure to solar radiation⁸ can produce quite significant changes in moisture uptake in unpainted composites. In practice most composites will be protected on the surface with paint or some other surface coating and so the effect of solar radiation will be greatly reduced. Flight mission type and frequency should not contribute greatly to moisture diffusion since the greater part of the environmental life of an aircraft is spent standing on the ground. Surface coatings such as paints, although probably reducing the rate at which moisture diffuses into a composite, will not prevent moisture ingress. Therefore the effects

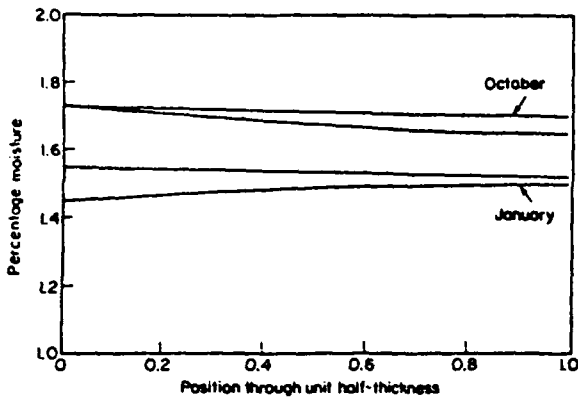


Fig. 5 Effect of monthly changes in environment at Port Harcourt on the through-thickness moisture distribution of a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$, $t = 1.0$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

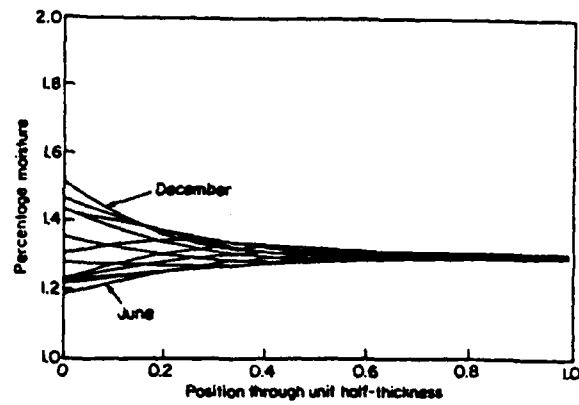


Fig. 8 Effect of monthly changes in environment at Woodbridge on the through-thickness moisture distribution of a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$, $t = 2.00$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

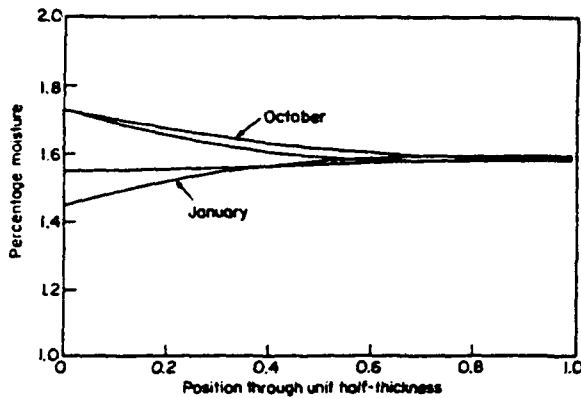


Fig. 6 Effect of monthly changes in environment at Port Harcourt on the through-thickness moisture distribution of a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$, $t = 3.00$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

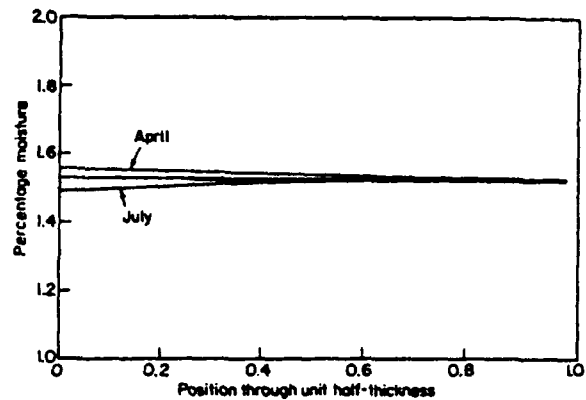


Fig. 9 Effect of changes in environment at Singapore on the through-thickness moisture distribution of a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$, $t = 2.00$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

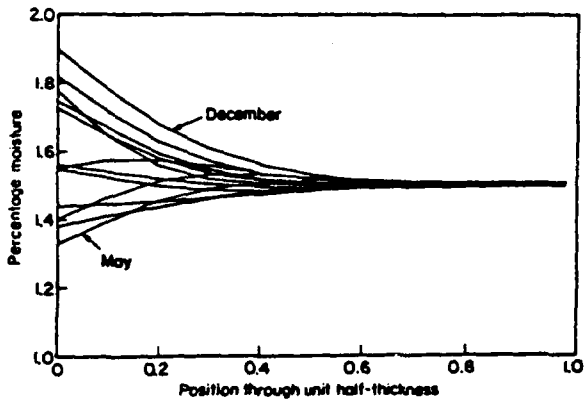


Fig. 7 Effect of monthly changes in environment at Schleswig on the through-thickness moisture distribution of a unidirectional XAS/913 CFRP laminate ($V_f = 0.6$, $t = 2.00$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

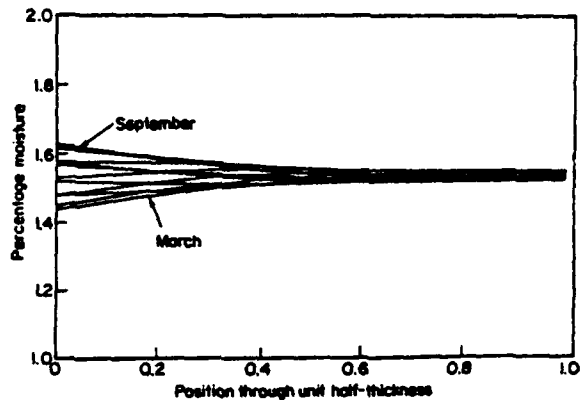


Fig. 10 Effect of monthly changes in environment at Guam on the through-thickness moisture distribution of a unidirectional CFRP laminate ($V_f = 0.6$, $t = 2.00$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

of surface protection can be ignored in respect of moisture equilibrium levels.

The variability of the environment *ie* the monthly temperature and humidity changes, has been shown to feature strongly in deciding both the final moisture

level and the way in which moisture is distributed in the outer surface layers of a composite. This is adequately demonstrated in Figs 5 to 12 for each of the six different climates and, in most cases, for three different laminate thicknesses.

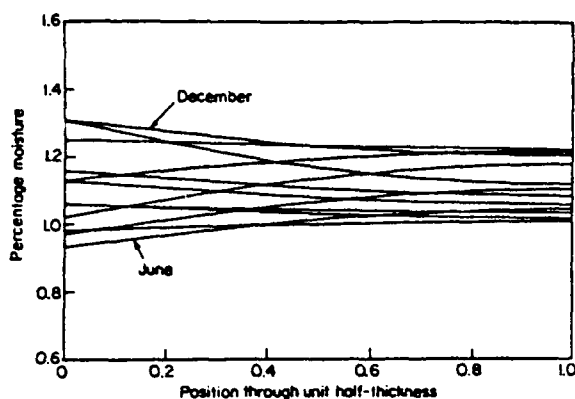


Fig. 11 Effect of monthly changes in environment at Bahrain on the through-thickness moisture distribution of a unidirectional CFRP laminate ($V_f = 0.6$, $t = 1.00$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

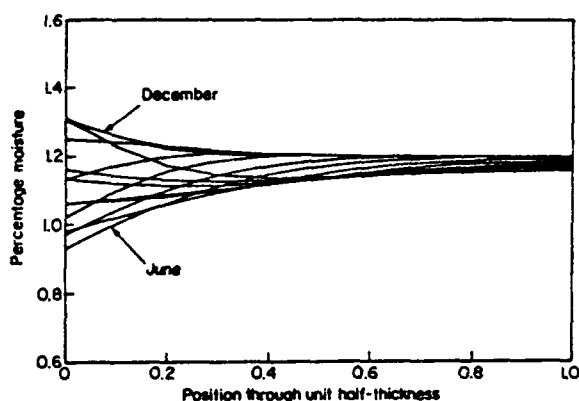


Fig. 12 Effect of monthly changes in environment at Bahrain on the through-thickness moisture distribution of a unidirectional CFRP laminate ($V_f = 0.6$, $t = 3.0$ mm, infinite b and l , exposed on both sides) after reaching an equilibrium condition

The results presented here show that, using data appropriate to freshly made XAS/913 laminates, an equilibrium moisture level of about 1.6% corresponds to the worst realistic world-wide environment. Fig. 1 shows that this level can be produced artificially by exposing the laminate to a constant relative humidity of 84%. The time taken to reach the appropriate equilibrium level or distribution will of course depend upon laminate thickness and temperature; for the 913 resin system the accompanying temperature should not exceed 45°C.¹⁰ It should be noted that, in calculating the artificial ageing environment needed to simulate the worst actual climate, the characteristics of the fibre/resin system act only as a transfer function. The 84% RH environment should therefore be suitable for the simulation of world-wide environmental degradation of any fibre/resin system.

Whether the XAS/913 system would ever reach a moisture level of 1.6% in service is thrown into some doubt by recent work at the Royal Aircraft Establishment¹¹ and elsewhere.¹² Reference 11 has shown, for unidirectional 2 mm thick specimens made from the XAS/914 system, that exposure to a steady

60°C and 75% RH environment commencing within 24 hours of manufacture produced a moisture equilibrium level of 1.3%. Identical specimens from the same laminate, but stored in a desiccator at 20°C for 6 months, only reached an equilibrium level of 1.1% under the same conditions. Other work¹² has shown that composites physically age when stored below their glass transition temperature, T_g , resulting in a reduction in absorbed moisture of about 20% compared with that of a freshly made laminate exposed to the same environment. Examination of composite components returned after service use¹³ has also failed to show moisture levels as high as 1.6%. However if allowance is made for the reduced moisture absorption properties of physically aged laminates, the moisture levels found in service would equate to between 1.25% and 1.5% in a freshly made laminate which had seen the same environment. This agrees quite well with the worst case level predicted in this paper.

In view of this physical ageing behaviour the use of moisture content to describe composite degradation is no longer adequate since the appropriate level would depend on physical age. Indeed it is suggested that an alternative criterion for describing composite degradation is a constant RH environment.

The choice of constant RH as a conditioning parameter allows a number of environmental conditioning problems to be overcome:

- 1) any differences in physical ageing between specimens can be accommodated by environmental conditioning to a steady-weight state;
- 2) an RH value can be selected to give an exact steady-state equivalent to any complex climatic environmental cycle; and
- 3) this equivalent RH value can be used for most fibre/epoxy resin systems.

In justifying the above approach, account will have to be taken of the significance of differences in physical age and of any corresponding change in moisture equilibrium level on the mechanical properties of a composite structure. It has been shown in Reference 14 that diffusion of moisture into an epoxy matrix occurs both through a polymer-water interaction (two-thirds of the absorbed moisture) and through water simply occupying the free volume (one-third of the absorbed moisture). In Reference 12 it is suggested that physical ageing reduces the free volume (free volume is defined as the unoccupied volume, or a distribution of holes between molecules or macromolecular segments) with a corresponding reduction in the moisture absorbed. It can be argued that the moisture which is hydrogen bonded to -OH groups is likely to have most influence upon mechanical properties, therefore any loss in moisture content due solely to a drop in free volume moisture could be postulated to have no significant effect. This hypothesis has yet to be proven.

On the other hand, if a drop in moisture level during physical ageing does have an effect on mechanical properties, be it for better or worse, one of the following two approaches to the total ageing of a structure is possible:

- 1) a freshly made laminate after environmental ageing can be considered to be in the most degraded state and will therefore give a conservative measure of mechanical properties, and

- 2) the physical ageing process can be accelerated by annealing at a temperature below T_g for a specified time,¹² thus establishing a standard condition before environmental ageing.

Clearly some understanding of the effects of physical ageing on British resin systems, and work to promote this understanding, is needed.

CONCLUSIONS

- 1) Using a prescribed moisture level to define degradation appropriate to given service experience is not valid.
- 2) A suggested better alternative is to prescribe a fixed RH value for long enough to ensure any given matrix system reaches an equilibrium condition in thin structures or a representative through-thickness moisture distribution for thick structures.
- 3) Work needs to be done to compare the moisture equilibrium levels and the mechanical properties of freshly made and physically aged laminates after they have been conditioned to a fixed RH environment.

ACKNOWLEDGEMENT

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