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PREFACE

Certification is the procedure which provides the possibility of making certain that any aircraft, whether civil or military, has an acceptable safety level for a given future utilization. For metallic structures, it is expressed as a set of rules which, from experience, it has been possible to transform into numerical specifications.

Composite materials, the advent of which is too recent for calculated data yet to be generalised, do however offer a number of very specific characteristics which are quite often rather imperfectly understood.

Consequently, the four papers which are included in this Report do not claim to provide certification principles in the form of regulations. For the time being, they describe various conservative approaches which, through experimental programmes which are often most impressive, have retained two simultaneous major objectives:

1 – firstly, to fly a certain number of high performance aircraft structures, without major risk.

2 – to accumulate experimental and technical data which will help to define general safety factors, defect tolerances, propagation criteria and critical thresholds so as to have available a set of rules limiting and describing accurately the necessary physical checks and testing.

It is, indeed, important to draw attention, as the authors do, to the fact that transposition of experience gained on metallic aircraft certification is defective on several points, notably:

The fatigue concept, considered as the accumulation of mechanical stresses, is complicated in the present case by the effect of humidity and, even for subsonic flight, by the effect of temperature. Consequently, it is not possible to accelerate structural fatigue testing. These tests can be conducted only in real time.

Analysis of the failure process of composites is so little advanced that certification can be achieved only when the success of a complete structural assembly test is recorded. A reinforcement deemed necessary cannot be justified by a mere design calculation, as it is for metallic materials. However imperfect are the rules (Miner, for example) by which life duration can be assessed, they have, at present, no equivalents applicable to composite materials.

The efficiency and behaviour of repairs can be treated at present only by specific tests.

The development of non-destructive testing and, notably, of simple procedures applicable to in-service structures seems to be absolutely necessary in order to avoid the cost of periodic inspections of composite structures becoming a deterrent factor.

Although they may be conservative, the approaches described herein evidence an optimism already supported by significant demonstrations. It may be anticipated that, once the advantages provided by structural weight savings, together with performance increases, have been confirmed, composite materials will also claim their place, above all from safety aspects.

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CONTENTS

U. S. NAVY CERTIFICATION OF COMPOSITE WINGS FOR THE F-18 AND ADVANCED HARRIER AIRCRAFT

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ABSTRACT

The Naval Air Systems Command (NAVAIR) anticipates that graphite/epoxy composites will be used in the wings of production aircraft of the Model F-18 and AV-8B Advanced Harrier aircraft. The F-18 wing will utilize composite, full span, upper and lower skins. The AV-8B wing will incorporate composite sine wave substructure as well as full span upper and lower skins. This paper describes the criteria which presently determine the permissible use of composites in production aircraft and the adaptation of an existing framework of aircraft certification practices to the special characteristics of composites. Factors involving the scheduling of full-scale static and fatigue tests and their effect on the use of composites are discussed. Static and fatigue design and test loads are described. Small and large scale element testing is covered. Some of the structural design problems and load/strain criteria are described. Typical procedures governing the development and interpretation of design data, laboratory and flight tests are explained. The authors wish to acknowledge the important contribution of L. F. Impellizzeri from McDonnell Douglas Corporation to the F-18 information presented in this paper.

INTRODUCTION

NAVAIR is in the process of procuring two new fighter aircraft. The F-18 is a derivative of the YF-17A, designed by Northrop Corporation, as the prototype of a light weight fighter. The F-18 aircraft, for which McDonnell Douglas Corporation is the prime contractor with Northrop as principal subcontractor, is a further development for use by the Navy and the Marine Corps and features the capability for operation aboard aircraft carriers. The F-18 includes many technological advances, among which is the use of graphite/epoxy for wing skins, empennage, speed brake, leading edge extension, and miscellaneous doors as shown in Figure 1. About 10 percent by weight of the structure is graphite/epoxy, providing a substantial weight savings.

The AV-8B Harrier is an advanced development, by McDonnell Douglas and Hawker Siddeley, of the basic AV-8A including a new supercritical airfoil wing and other features but retaining, insofar as possible, the existing fuselage. Graphite/epoxy is again used for the wing torque box skins and also for most of the wing substructure as shown in Figure 2. The flap, slot door, aileron, outrigger fairing, and overwing fairing also utilize graphite/epoxy. About 70 percent of the wing weight (15% of the aircraft structural weight) is graphite/epoxy resulting in a projected wing weight saving of 20 percent. At the time of this writing, consideration was also being given to the use of graphite/epoxy for the forward fuselage skins in certain aircraft configurations.

The F-18 and AV-8B airplanes are production aircraft being developed for service operations and not as experimental vehicles for development of composites. The airplane programs are pragmatic with the usual concerns for timeliness, cost, performance, and quality. The solution to problems is approached from an open-minded engineering viewpoint. The structural design requirements are specified in terms of the MIL-A-8860 series of specifications and apply equally to composites and metal. In compliance with these requirements, the evolving development program for composites is planned as described in the paper and is only partially accomplished as of this date.

This paper discusses the basic criteria which determine the locations and conditions in which the use of composites is considered permissible and the design and qualification procedures used by NAVAIR to verify the structural adequacy of the F-18 and AV-8B composite structures.

POLICY ON THE USE OF COMPOSITES

From the start, there has been a recognition of the risk involved in the use of composite materials in primary structure of a production airplane. However, the large number of development components manufactured and tested by the U. S. military, NASA, and industry, as well as the successful production empennage applications on the F-14, F-15, and now on the F-16, have provided the confidence that a successful large production run of major composite components could be undertaken if certain prudent precautionary measures were taken. Because of the concern at many levels within the Department of Defense about using composite materials and the impact of this concern on authorizing procurement of airplanes which are largely dependent on composites, it appeared that assuring the adequacy of composite structure and thus the airplane program, required a fallback plan to salvage the project if composites were to cause a major development problem.

NAVAIR proposed and the procurement specifications required that all applications of composite structures be such that the structure can be inspected from both sides, be repairable, and be replaceable. The principal result of this policy was that wing skins would have to be installed with mechanical fasteners rather than bonded to internal structure as has sometimes been proposed. Secondly, the policy precludes the general applications at present to structures such as fuselage bulkheads which are buried or hidden. As the design of the two airplanes progressed, this policy has been interpreted liberally where there have been difficult design decisions to be made, but the intent of the basic policy has been adhered to.

One of the considerations in the use of composites was the assurance of a fallback plan in the event of a catastrophic failure situation during development. Such a situation might prohibit flight or might even require the development of a substitute component, in either case, causing delays which would surely jeopardize the continuance of the basic airplane development program, not to mention the huge dollar expenditure which would ensue. In the case of the F-14A airplane, which features a boron/epoxy aluminum-honeycomb horizontal stabilizer, the specifications required early development and laboratory test of a composite stabilizer to allow time for development of a conventional metal component, if necessary, without impacting the airplane schedule. In other words, a fallback aluminum or titanium tail was in the basic requirement. The boron stabilizer was successful with no apparent difficulties to date, which precluded the need for a metal fallback and added to the confidence for going ahead with composite wings. The requirement for a stand-by or fallback metal structure has changed into the current philosophy of allowing time for recovery from a major, premature failure in tests, which permits a re-test as required so as not to impact the total airplane schedule, particularly the date of first flight which is always a significant milestone.

In the case of the Harrier wing, the composite program started as part of a general technology development program of many composite components for many military aircraft, unassociated with any production airplane program. However, the advanced Harrier airplane program was being initiated and was contemplated to be constructed with a composite wing. Therefore NAVAIR decided that so far as practical, the technology development wing be built in a production configuration including provisions for all of the clips, brackets, and holes which would appear in production. The early schedule of this development program precedes by several years the schedule of tests which would be required for a conventional wing in a normal production program. This development program includes all of the element, subcomponent and full-scale wing torque box testing and will be commented on later in this paper. Similarly, the F-18 airplane schedule, also allows the necessary time for a recovery from a major failure during development tests.

In summary then, the use of composites is limited in application and in scheduling such that aircraft programs cannot be trapped into a situation from which there is no recovery during development or service use. There have not been and, with the schedule allowances, there should not be holdups to the development schedule because of major errors in the design of composite structure. For fleet applications in production aircraft, the components are all inspectable, repairable, and replaceable should there occur some unforeseen phenomenon which would gravely affect the structural integrity of the composite structure.

RISK ASSESSMENT

At all levels within the administrative chain of approval for these two airplane projects. including the top levels of the Department of Defense, there have been questions concerning the loss of strength from moisture absorption, high temperature effects, galvanic corrosion of aluminum, lightning protection, fatigue life, and so on, all of which bear on the risks associated with composites. The development program has procedures to cope with these questions and are discussed in greater detail later in the paper.

For both of these wings, the maximum design strain level has been kept by McDonnell to about 4000 to 5000 μ inches/inch. Comparison of laminate failure strains with wing cover design strain levels shows that there is an adequate margin of safety for the deleterious effects of elevated temperature and moisture absorption for the F-18 and Harrier service environments. For the F-18, the maximum temperature in the wing box does not exceed 220°F and for the AV-8B, the speed-altitude envelope does not pose an elevated temperature problem. Thus, the loss of strength from moisture absorption and temperature can be accommodated because of the working strain level in the composites. The design strain levels that have been developed to accommodate fastener holes also serve to provide an inherent damage tolerant structure.

DEVELOPMENT TESTS

Both programs include the following procedures typical of the development of any production airplane.

(1) Tests of small elements to determine allowable loads. These tests are performed with lay-ups typical of the structures, including holes. These tests include effects of moisture and temperature and other considerations.

(2) Subcomponent tests of detail design features, some of which are performed in deleterious environment.

(3) Large elements, such as box beams.

(4) Pre-production components.

(5) Full scale, total airplane, static and fatigue tests to failure.

Regarding fatigue, in conformance with NAVAIR practice, both airplanes are designed for a minimum of two lifetimes of a severe spectrum of severe loads. The design and test spectra include positive and negative loads applied on a flight by flight basis. The requirement is 6000 hours times the scatter factor of 2.0. The specifications however require testing to a scatter factor of 4.0, if attainable, to assure determination of the mode of failure.

INTERPRETATION AND EVALUATION OF TEST RESULTS

The full scale certification tests of composite structures are meaningful in confirming the adequacy of a structural design in so far as the full scale tests can be related to design allowables and development test results in a way that will allow the effects of material property scatter and severe environmental conditions to be included in the assessment of structural adequacy. In the certification of a structure, each test failure is analyzed to determine the cause, implications and the necessity for corrective actions, if any. In the certification practices evolving for composite construction, evaluation of full scale test results will rest not only upon failure analyses but also more generally will depend upon comparisons of experimental strain measurements on the full scale structure, predicted strains, design allowables and design development test results. The purpose of these evaluations is, of course, to demonstrate with high probability that strength, rigidity, damage tolerance and durability design requirements are met in metal construction as well as composites.

Among the factors which demand special consideration in the certification of composite structures are the variability of material properties and degradation due to temperature and moisture. Generally it is planned that element and component tests will be relied upon to account for these factors. Thus in the composite certification process, design allowables and development test results will serve not only their traditional role in design development but will also provide for interpretation of full scale test results. Design allowables and design development test data will be evaluated to predict the failure mode and quantitative property changes of environmental degradation in various laminates and geometric details.

Design allowables for laminate strength properties are being derived as shown schematically in Figure 3. Sufficient replicate tests are conducted to establish the distribution $f(\varepsilon)$ of strength properties for a baseline condition. From these data, a mean value $\bar{\varepsilon}_b$ and design allowable ε^*_b for the baseline condition are statistically determined.

A reduced number of specimens are tested for combined temperature-moisture environmental conditions so that a mean value $\bar{\varepsilon}_e$ can be established for each of these conditions. The design allowable ε^*_e for a given environmental condition is established by reducing the baseline allowable ε^*_b by the difference in the mean strength values of the baseline and the given environmental condition; that is, $\varepsilon^*_e = \varepsilon^*_b - (\bar{\varepsilon}_b - \bar{\varepsilon}_e)$. An assumption implicit in this procedure, of course, is the similarity of population distributions of the baseline and environmental conditions. Additional tests will be conducted to establish, for selected conditions, the utility of this assumption.

As presently planned, full scale test articles may not be environmentally preconditioned to a specified state although the possibility of doing so with an F-18 wing is under study. However, selection of critical full scale test conditions and evaluation of results will be based upon consideration of potential failure modes and quantitative "compensation" or "Knockdown" factors derived from the environmentally compensated design allowables and development test results.

The actual failure load for a full scale structure will be required to exceed the 150% design limit load value by a compensation factor dependent on failure location, failure mode, metal or composite, environmental test condition and material variability. Additionally, at 150% design limit load all strains measured (and so far as practicable analytically extrapolated to other locations on the structure) must not exceed allowable strain levels, ε^*_e developed for the worst expected load-environmental degradation comninations. Figure 4 schematically depicts strain versus applied load at one of many locations on the structure, not necessarily the failure location. In accordance with above requirements, a strain-load response along the 0-I path would be acceptable whereas, a response along the path 0-II would not be accpetable even though the actual failure load at II is equal to that at I. The response along the path 0-II would not be acceptable because this response indicates that expected weaker members of the population would not have sustained 150% of design limit load for the most severe environmental condition.

INTRODUCTION TO F-18 AND AV-8B DEVELOPMENT TEST PROGRAMS

The remainder of this paper will deal with the specific aspects of the composite wing development programs for both the F-18 and the AV-8B. Because of the similarities in the development processes some duplication of material will be presented. Hopefully, this will help to convey the systematic manner in which such programs are conducted, the adherence to the aforementioned policy in the use of composites, the orderly development of design allowables using only representative laminate structure as used in the aircraft, the manner in which the influence of environment is accounted for, and the use of subcomponent test specimens for design development leading up to the full scale airframe ground and flight test programs.

This paper covers the scope of the many parameters investigated in depth to assure the satisfactory use of composite materials. Discussion in detail is beyond the intent of this publication. Therefore, only the most questionable and significant problem, - that of the effects of moisture and temperature -, is discussed with a little detail.

DESCRIPTION OF F-18 COMPOSITES

The F-18 uses graphite/epoxy in applications for the wing skins, leading edge extension, trailing edge flap, rudder, horizontal stabilizer, and vertical fin. The stabilator, rudder, and speed brake are full depth honeycomb. The wing torque box substructure is metal.

The wing is thin and must carry high load intensities. Its planform is shown in Figure 5. Note that the wing is not continuous through the fuselage but is instead attached by titanium fittings to the fuselage carry-through structure. This requires that the loads be distributed from the skins of a thin wing into 3 lugs, top and bottom, creating very high load intensities indeed. A cross section of the wing through the center lugs is shown in Figure 6. To save weight and also to preserve fuel volume stepped titanium fittings, bonded to the skin at the wing root, are used. The principle of inspectability is retained since the entire skin is removable for inspection on both sides of the fitting. Similar splice fittings at the stabilizer root ends of the F-14 and F-15 aircraft have not shown any deterioration after five or six years of service. The F-18 wing root step-lap bonded joint configuration development has been tested with approximately 150 specimens to date to evaluate different adhesives, to determine the impact of variations in cure cycle time and temperature, and the effects of manufacturing anomalies, to demonstrate the static and fatigue strength of the wing root connection, and finally to optimize its geometry.

DESCRIPTION OF AV-8B COMPOSITES

The AV-8B Harrier uses graphite/epoxy in applications for the wing skins, the wing torque box substructure, flap, slot door, aileron, outrigger fairing, and overwing fairing. The advanced Harrier wing pictured in Figure 7 is continuous from tip to tip and attaches to the fuselage at six points as in the original AV-8A Harrier. There are three pylon stations on each wing. The two inboard stations carry external fuel tanks. The composite torque box is a multi-spar, monolithic cover design. There are eight sine wave composite spars in the main torque box. Figure 8 shows the sine wave front spar design which is fabricated with woven graphite/epoxy cloth. The upper and lower wing covers are simple one piece monolithic laminates extending from wing tip to wing tip. The entire torque box structure is assembled primarily with titanium fasteners. Approximately 80% of the torque box span serves as an integral fuel tank. Sealing is accomplished around the tank periphery by injecting a compound into a channel groove incorporated into the composite spar caps. Figure 9 shows a cut-away view of the assembled torque box. The high load intensities which require stepped titantium fittings on the F-18 at the wing root do not exist on the Harrier except at the wing root re-entrant corner. A proposed redesign is in process to reduce load intensities and to eliminate the stepped fitting.

At the present time, there are no other applications of high strength composites on the Harrier. Basically, this is to preserve as much as possible of the AV-8A fuselage and tail structure. However, there is under consideration a major change to the forward fuselage geometry. For this change, the use of composite external skins is under study.

F-18 STRUCTURAL TEST PROGRAMS

Numerous element and small component tests are being performed during the F-18 design phase. Figure 10 shows that a small part of these will be continuing through drawing release. The major considerations evaluated by testing small element specimens and the status of that test program are outlined in Figure 11.

The basic approach being used to size the F-18 wing skins is to establish design allowables in tension and compression at room temperature from small test specimens with typical fastener holes. All of these tests are conducted on a number of different layups covering the range of laminates that will be used. The test results are then reduced when necessary based on additional tests evaluating environmental and temperature effects, fastener bearing stresses, off-axis and biaxial loading, etc.

One of the more important questions in composite structure design is the effect of moisture and temperature on laminate strength. Figure 12 shows test results for a 48 ply laminate of AS/3501-6 in a 42/50/8 layup - 42 percent zero degree plies, 50 percent plus or minus forty-five degree plies, and 8 percent ninety degree plies. The data indicate that the critical design condition is wet and hot for compression and dry and cold for tension. The airplane is being designed to withstand 150 percent of design limit load for these extreme conditions. The highest temperature for the F-18 wing skins is about 220°F and the lowest temperature is about -65°F. The moisture absorbed by the specimens illustrated in Figure 12 was 0.6 percent by weight. This corresponds to the maximum amount of moisture that is expected to be absorbed in the thinnest section of the F-18 wing skins during deployment in highly humid zones including the effect of thermal spikes to 220°F.

Additional significant test results that have been generated to date are listed in Figure 13. In particular, a total of 135 specimens were tested to evaluate the effects of compression dominated fatigue on AS/3501-6. The variables tested included moisture content, different layups, loaded and unloaded holes, and cyclic loading rate. The fatigue tests of these specimens were conducted to 24,000 hours of the F-18 flight-by-flight fatigue spectrum simulating upper wing skin stress excursions. None of the specimens failed in fatigue. The residual strengths of the specimens were determined after 24,000 spectrum hours. Maximum strength reduction was 15%. Specimens with moisture contents of from 0.6% to 1.5% exhibited no static strength reduction due to fatigue cycling. Residual strengths were also unaffected by variations in cyclic loading rate from 2 per minute to 9.4 cps.

Element tests have also been conducted to evaluate the effects of ply stacking sequence, interference fit fasteners, fastener hole location geometry, and the propping benefit of installed fasteners in a graphite/epoxy laminate under compression loading. The conclusions from these tests are listed in Figure 13.

As a result of all the element specimens tested to date, design stresses at ultimate load are selected at about 50,000 psi in both tension and compression for a 46/50/4 layup which is the baseline selected for the F-18 wing skins for spanwise loading. With a modulus of about 10 million, this converts to 5000 μ in/in. Areas near the leading and trailing edge where large chordwise loads are introduced have a higher percentage of ninety degree plies. The stress quoted above is restricted to zones of very low fastener bearing stresses. Design allowables are reduced by the ratio of increasing stress concentration factor in zones of high fastener load transfer.

The detail design development test articles shown in figure 14 represent the next step up from element tests to evaluate critical locations on the wing. The upper and lower cover panel test specimens will be used to evaluate load distributions around relatively large cut-outs. The results should indicate how well the analytical techniques can predict load gradients and failure stresses. Specific splice designs will be verified by test articles simulating the root, leading and trailing edges, and the pylon fitting.

The three box beams representing the wing root, wing fold, and pylon fitting regions of the torque box are each two-cell torque boxes about seven feet long. The box beams will duplicate the wing critical areas that are to be evaluated. It is planned to fatigue test all of the specimens shown in Figure 14 for 24,000 hours and then static test to failure.

The next to last step in demonstrating F-18 structural integrity is the preproduction component verification tests shown in Figure 15. The test goal for these specimens is to reach the contractual 12,000 hours without failure, followed by additional cycling to failure or 24,000 hours. If the test articles do not fail in fatigue, they will be static tested to failure. All of the preproduction component tests will be completed early in 1978.

The laboratory program includes three full scale articles - one for static test, one for drop test, and one for fatigue. The static program includes 5 major flight load conditions. Drop tests are performed to simulate the 1000 most severe carrier landings. Most of 2 lifetimes (12000 hours) of flight-by-flight testing will be completed on the fatigue article prior to start of pilot production. The full scale development program includes eleven flight test aircraft.

AV-8B COMPOSITE WING DEVELOPMENT PROGRAM

The AV-8B Composite Wing Program began on 1 November 1975 to verify the multi-spar monolithic wing cover torque box design selected for the advanced Harrier wing. The entire test program was structured to develop and verify the design through an orderly process in which the end result would be a well proven and tested wing torque-box design. This program featured element tests to develop basic design allowables data; environmental tests to determine the effect on strength and to evaluate the corrosion protection system; fatigue/fracture tests to characterize the effect of ballistic damage and determine the effect of manufacturing imperfections on fatigue and static strength; subcomponent tests to verify the fatigue and static strength of crtical structural areas; and major box beam tests of critical design areas on the wing torque box structure. Figure 16 summarizes the tests conducted in this program.

Element tests were conducted to determine the tension failure strains versus bearing stress levels for the wing cover laminates with holes. Figure 17 presents the test results for tension which, as for the F-18, show the cold, dry specimens to exhibit the lowest failure strains while the 200°F specimens were not critical for design. The effects of moisture absorption on the tension/bearing results was investigated and found to have essentially no effect. Testing was also conducted to establish the compression failure strains versus bearing stress levels for the wing cover laminates with holes. The compressive failure strains were not lowered due to the influence of bearing stress as was the case with the tension/bearing interaction. However, the effect of hot, wet test conditions at 200°F did reduce the compressive failure strains by 15% at a 1.0% laminate moisture content as shown by Figure 18. Similar results were just shown for the F-18.

A data base for design allowables was established by generating at least thirty data points at the room temperature ambient condition and a reduced number of data points at all other test conditions. The test data were then transformed into design allowables using a statistical analysis procedure. Essentially, the statistical method accounts for the spread of the test data and the number of specimens tested. Allowables at other temperatures and conditions are then established as previously noted.

A substantial amount of additional testing was undertaken to determine the effects of environment on composite structure. Figure 19 summarizes the environmetal test program. Tests were conducted to determine the effect of moisture and temperature on laminate stiffness properties; sine wave spar elements were subjected to moisture and JP-4 fuel soak and tested in fatigue and static strength at both low and elevated temperatures; and a compression panel representing the inboard aft section of the upper cover was moisture conditioned and subjected to 12,000 hours of spectrum fatigue loading at elevated temperature. All of the above tests showed that laminate stiffnesses remained essentially unchanged and the strengths were not substantially different from unexposed specimen strengths. The corrosive effects of dissimilar material on graphite/epoxy and aluminum joints were investigated with the combined effects of temperature, load cycling, and disassembly prior to spectrum testing to verify the adequacy of the corrosion protection system. Physical-biological environmental investigations were also conducted to determine if any permeation or microbial growth problems existed.

A major section of the outer wing fuel tank was fabricated and subjected to moisture and JP-4 fuel exposure. Inspection of the tank after exposure revealed that the corrosion protection system was adequate and the tank was then successfully subjected to pressure fatigue testing for 12,000 simulated flight hours to verify the strength and tank sealing characteristics for repeated loading. The tank was then successfully tested to 30 psi, the maximum design burst pressure. A major box beam test specimen representing the inboard pylon attachment to the torque box was preconditioned with moisture exposure and is currently undergoing structural testing at elevated temperature.

The structural subcomponent test specimens that were moisture conditioned were exposed until a moisture content of 1.0 percent by weight was attained as determined by control coupons. The rationale for the 1.0 percent moisture content was determined by establishing the locations for service operation in the world and the resulting times, temperatures, and humidity levels at these various locations for the fifteen-year aircraft design life. Analyses were conducted to determine the critical combinations resulting in the highest accumulation of moisture in the thinner laminates. The AV-8B wing skins are thinner than those of the F-18, thus a higher moisture level is required for the AV-8B laminates.

The structural subcomponents test program illustrated in Figures 20 and 21 consisted of basic tension/compression strength tests on the wing cover, shear strength tests on the sine wave spars, and fatigue/static strength tests on the wing centerline joint. Tests were conducted to determine the basic compression stability and strength of wing cover panels with and without an access door. Both uniaxial and biaxial tension tests were conducted to verify the design for cutouts and concentrated loading in the lower cover. Both the front and rear sine wave spars were tested in combined shear, tension, and bending loads, which simulated critical design conditions. Two competing designs for the interior sine wave spars were comparison tested in combined shear and tension. A section of the wing representing the centerline joint was tested in both fatigue and static strength to verify the design.

Five major areas of the wing torque box structure were selected for design verification tests. As shown in Figure 22, structural tests of these major box beam specimens will verify the basic design approach for the Harrier torque box structure and provide essential information on fabrication and assembly procedures.

The AV-8B ground test program and a schedule of major milestones are outlined in Figure 23. The structural test program for the preproduction program consists of a full scale, full strength torque box and two flight demonstration aircraft incorporating the new wing but with AV-8A fuselages. The full scale production program with the AV-8B fuselage consists of two full scale ground test articles for static and fatigue test, and four flight test aircraft. The pilot production program consists of twelve aircraft.

The ballistic damage tolerance program for the wing consists of establishing the relationship between damage size and failure strength for both tension and compression and performing wing finite element model analysis with simulated damage. The results of the analysis together with the expected failure levels for various sizes of damage and locations in the wing structure as determined by ballistic tests on box beams is then used to predict the extent of damage that could be tolerated for the full flight envelope. To verify the battle damage tolerance in the wing structure, ballistic damage tolerance tests are tentatively planned to be performed on the prototype wing torque box after completion of all other programmed tests.

QUALITY ASSURANCE

Comprehensive quality assurance procedures are used to provide the necessary checks required to produce consistent high quality composite structure for use on high performance aircraft. It is siginificant to note that the extensive structural development test programs previously covered certify the structural design for the airframe, but the quality assurance procedures establish the criteria that each composite assembly must meet to certify the airframe as it comes off the production line. Stringent controls are exercised defining the quality assurance requirements by material specifications, controls on reproducibility of processing and consisting of properties, establishment of pre-preg allowable working life, process control specifications, engineering drawings, and extensive detailed nondestructive testing procedures.

SUMMARY

For the development of the composite wings of the F-18 and AV-8B aircraft, normal NAVAIR procedures and criteria are used as would apply to any airplane for large production runs. A policy was developed concerning the use of composites which minimizes over-all program risk in the event of major problems with composites. Qualification of the structure is substantiated by an extensive test program of small and large elements concerning both strength and environmental characteristics. An approach is evolving for a means to evaluate the structural adequacy of fleet aircraft as determined from full scale failing load tests, which is based upon relating the strains from small element tests in ambient and deleterious environments with strain levels from the full scale tests.

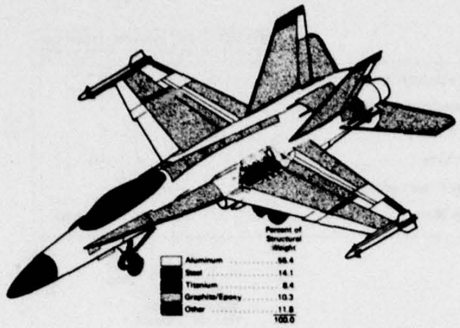


FIGURE 1 F-18 MATERIALS DISTRIBUTION

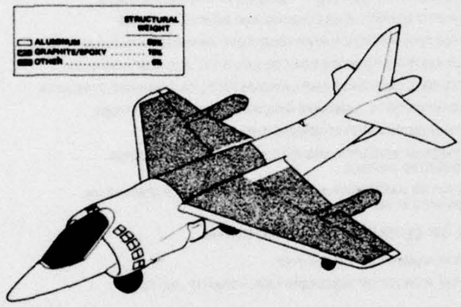


FIGURE 2 YAV-8B MATERIAL DISTRIBUTION

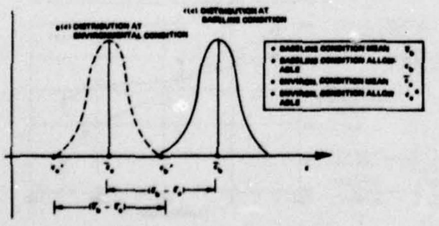


FIGURE 3
DESIGN ALLOWABLE DEVELOPMENT AND COMPENSATION FOR SEVERE ENVIRONMENTS

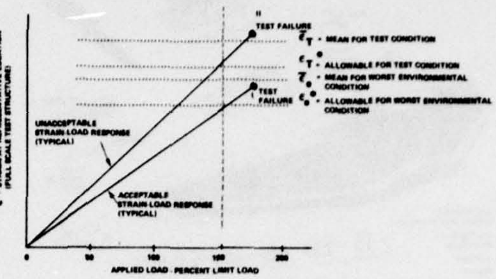


FIGURE 4
STRAIN VERSUS APPLIED LOAD ASSESSMENT CONCEPT FOR FULL SCALE STRUCTURAL TEST

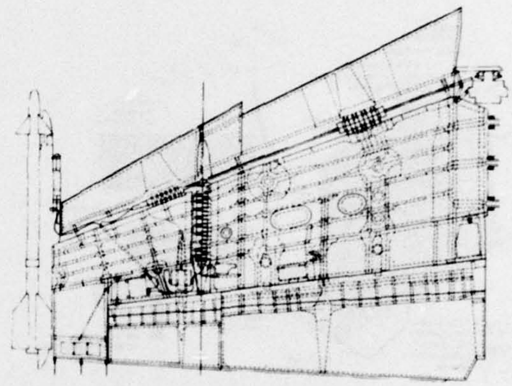


FIGURE 5 F-18 WING STRUCTURE

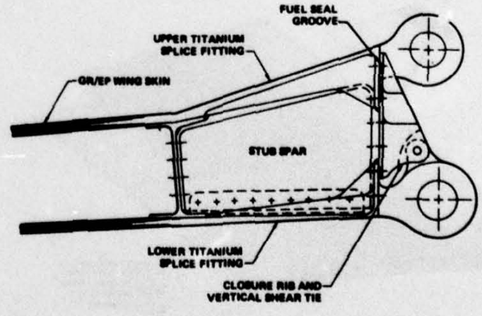


FIGURE 6
WING ROOT ATTACHMENT CONFIGURATION

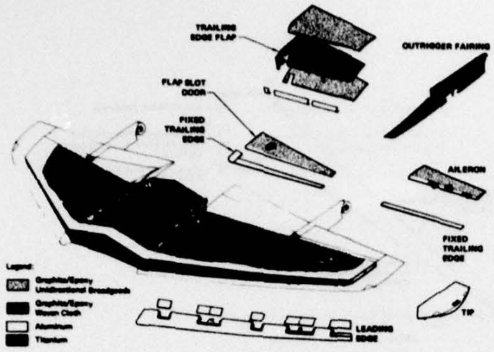


FIGURE 7 AV-8B COMPOSITE WING

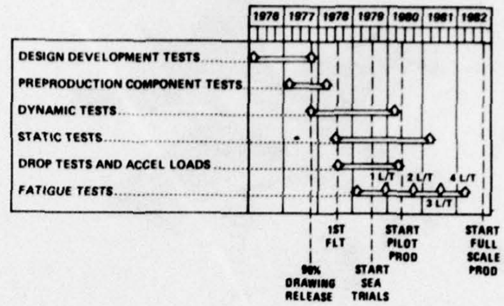


FIGURE 10 F-18 LABORATORY TEST PROGRAM

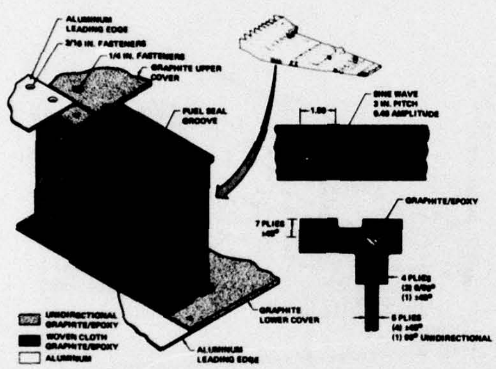


FIGURE 8 AV-8B FRONT SPAR-FUEL TANK AREA

CONSIDERATIONS EVALUATED IN ELEMENT TEST PROGRAM

- MOISTURE/TEMPERATURE EFFECTS ON LAMINATE STRENGTH
- OFF ORTHOTROPIC AXIS LOADING AND BIAXIAL LOADING
- EFFECTS OF FASTENER BEARING STRESS COMBINED WITH AXIAL STRESS
- EFFECTS OF PLY ADD ON RATE ON LAMINATE BUILD-UP STRENGTH
- HOLE SIZE, THICKNESS, AND LAYUP EFFECTS ON LAMINATE STRENGTH
- CORRELATION OF LAMINATE STRENGTH TO NDT INDICATIONS
- HIGH MODULUS/HIGH STRENGTH Gr/Ep EVALUATION
- EFFECTS OF MOISTURE AND CYCLIC RATE ON COMPRESSION DOMINATED FATIGUE
- IMPACT OF FASTENERS INSTALLED AND FASTENER SPACING ON LAMINATE STRENGTH

STATUS OF ELEMENT TEST PROGRAM

- TOTAL NUMBER OF TESTS = 2700
- TOTAL NUMBER OF TESTS COMPLETE = 2000 (17 DEC 1976)

FIGURE 11 SUMMARY OF ELEMENT TESTS

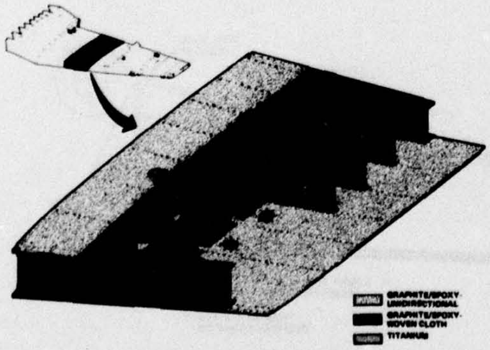


FIGURE 9
AV-8B TORQUE BOX
STRUCTURE-INBOARD PYLON AREA

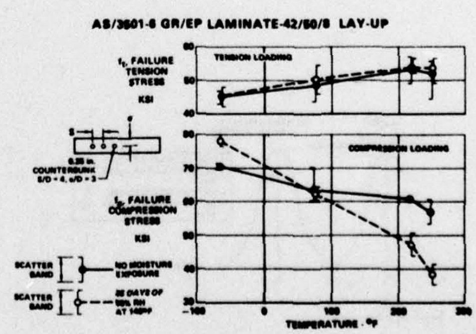


FIGURE 12
MOISTURE AND TEMPERATURE EFFECTS
ON LAMINATE STRENGTH

- COMPRESSION DOMINATED FATIGUE EFFECT ON AS/3501-6
  - LAMINATE STRENGTH REDUCTION AFTER 24,000 HOURS LESS THAN 15%
  - MOISTURE DOES NOT DEGRADE FATIGUE STRENGTH
  - CYCLIC RATE IS NOT SIGNIFICANT
- STACKING SEQUENCE
  - GREATEST LAMINATE STRENGTH WITH ±45° PLIES STACKED TOGETHER
- INTERFERENCE FIT FASTENERS (TAPERED AND STRAIGHT SHANK)
  - REDUCE STRENGTH BY 20%
- HOLES IN A ROW (4D SPACING)
  - STATIC STRENGTH IS 17% HIGHER THAN FOR END HOLE
- EFFECTS OF FASTENERS INSTALLED
  - COMPARED TO OPEN HOLE, COMPRESSION STRENGTH INCREASES 13%

FIGURE 13 MAJOR ELEMENT TEST RESULTS

TEST	TEST CONDITIONS	NO. SPECIMEN
ELEMENTS		
TENSION/BEARING LAMINATE ALLOWABLES	STATIC [illegible]°F, RT, -[illegible]°F	42
COMPRESSION/BEARING LAMINATE ALLOWABLES	STATIC [illegible]°F, RT, WET	30
TITANIUM STEP LAP SPLICE ALLOWABLES	FATIGUE [illegible]°F, RT, -[illegible]°F WET	21
ENVIRONMENTAL EFFECTS		
LAMINATE STRENGTH VS PERCENT MOISTURE	STATIC COMP [illegible]°F, [illegible]°F WET	30
SINE WAVE SPAR TENSION/COMP WITH MOISTURE	STATIC/FATIGUE [illegible]°F WET	6
SINE WAVE SPAR TENSION JP-4 ABSORPTION	STATIC/FATIGUE [illegible]°F	7
LAMINATE SHEAR PROPERTIES VS PERCENT MOISTURE	STATIC COMP [illegible]°F, [illegible]°F, WET	60
ALUMINUM/GRAPHITE JOINT WITH CORROSION ENVIRONMENT	FATIGUE/LOAD, TEMP, ENVIRONMENT	3
COMPRESSION PANEL WITH MOISTURE	FATIGUE/RESIDUAL STRENGTH [illegible]°F WET	1
FUEL TANK WITH ENVIRONMENTAL EXPOSURE	PRESSURE FATIGUE/WET	1
FATIGUE/FRACTURE		
TITANIUM STEP LAP SPLICE JOINTS WITH DEFECTS	FATIGUE [illegible]°F, RT, -[illegible]°F, WET	15
LAMINATE TENSION STRENGTH VS DAMAGE SIZE	STATIC RT	16
COMPRESSION PANEL STRENGTH VS DAMAGE SIZE	STATIC RT	3
STRUCTURAL SUBCOMPONENTS		
COMPRESSION PANEL	STATIC RT	1
ACCESS DOOR SILL STRENGTH TESTS	STATIC RT	3
COMPRESSION PANEL WITH ACCESS DOOR	STATIC RT	2
TENSION PANEL WITH ACCESS HOLE	STATIC RT	1
BIAXIAL TENSION PANEL WITH ACCESS HOLE	STATIC RT	1
BIAXIAL TENSION BOX WITH AILERON ACTUATOR	STATIC RT	1
SINE WAVE SPAR RUN-OUT AT FUSELAGE [illegible]	FATIGUE/RESIDUAL STRENGTH RT	[illegible]
SINE WAVE SPAR TENSION SPECIMENS	STATIC/FATIGUE [illegible]°F, RT, -[illegible]°F	3
SINE WAVE SPAR SHEAR TESTS	STATIC COMBINED SHEAR/TENSION RT	4
WING CENTERLINE JOINT	FATIGUE RT	1
BOX BEAMS		
NO. 1 INBOARD AFT UPPER/LOWER WING COVERS	STATIC PRESSURE [illegible]°F	1
NO. 2 CENTERLINE JOINT WING SECTION	STATIC PRESSURE [illegible]°F	1
NO. 3 INBOARD PYLON ATTACHMENT	STATIC WET ENVIRONMENT [illegible]°F	1
NO. 4 WING ROOT FUSELAGE ATTACHMENT	FATIGUE/RESIDUAL STRENGTH RT	1
NO. 5 OUTRIGGER GEAR ATTACHMENT	STATIC RT	1

FIGURE 16
AV-8B COMPOSITE WING DEVELOPMENT TEST PROGRAM

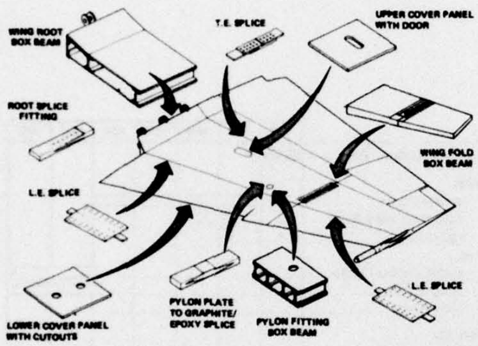


FIGURE 14
F-18 WING STRUCTURAL DESIGN DEVELOPMENT TESTS

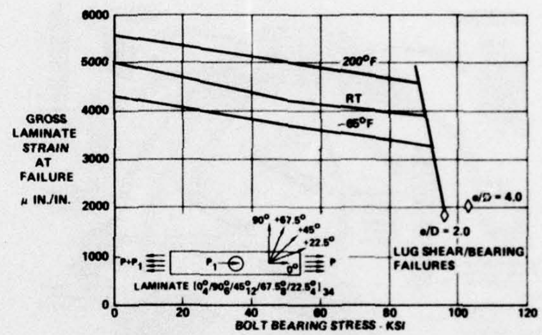


FIGURE 17
TENSION/BEARING LOADED HOLE TEST RESULTS

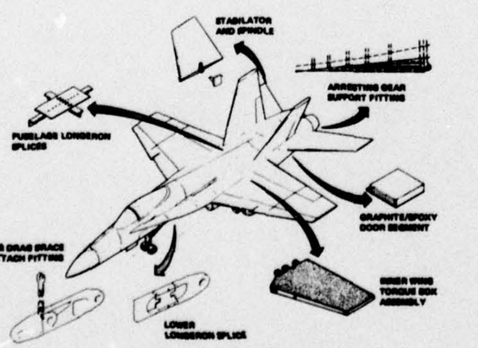


FIGURE 15
F-18 PREPRODUCTION COMPONENT VERIFICATION TESTS

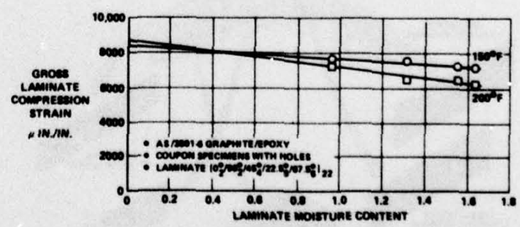


FIGURE 18
LAMINATE COMPRESSION FAILURE STRAIN
VS
PERCENT MOISTURE CONTENT

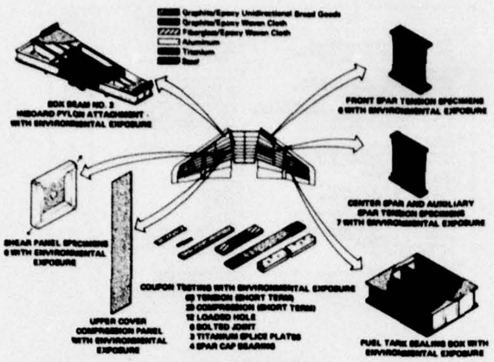


FIGURE 19 AV-8B ENVIRONMENTAL TESTING

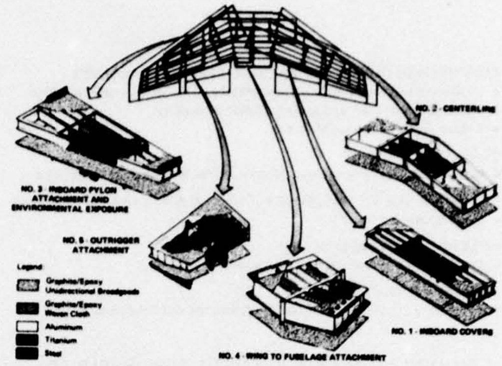


FIGURE 22
AV-8B MAJOR ELEMENT TESTS (BOX BEAMS)

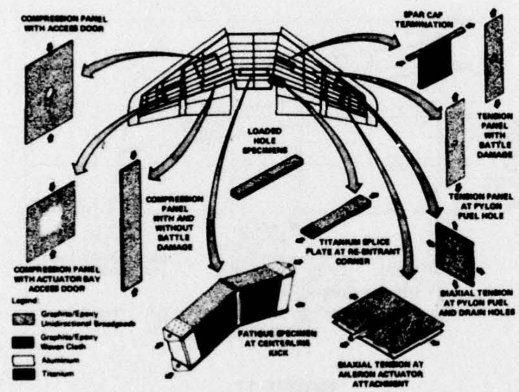


FIGURE 20 COVER TEST SPECIMENS

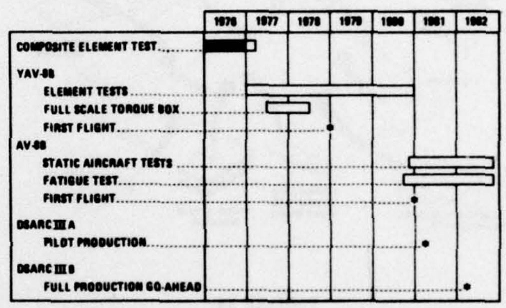


FIGURE 23 YAV-8B/AV-8B STRUCTURAL DEVELOPMENT PROGRAM

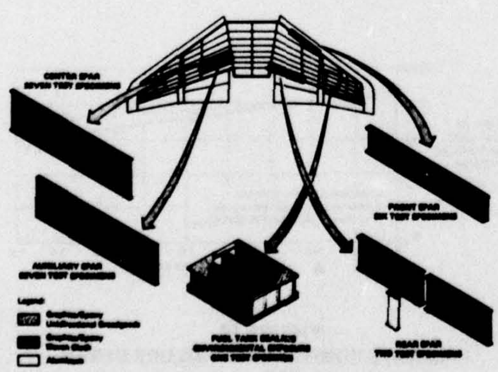


FIGURE 21 AV-8B SUBSTRUCTURE TESTING

APPENDIX A

DETAIL ANALYSIS OF STEP LAP BONDED JOINTS

Because of the applications of adhesive bonding on the F-18, a comprehensive analysis and test program was initiated early in the airplane design phase. The first step was to select a good adhesive for the job. FM 400 has been used widely in aircraft applications and has performed satisfactorily after years in service. It was selected as the baseline to compare other candidate adhesives. The reason for not selecting FM 400 was that the adhesive had higher temperature capabilities than required for F-18 applications. Adhesives with lower temperature capabilities are more ductile and have lower shear moduli. This lower stiffness is desirable because it reduces the peaking in the shear stress along the bondline. In other words, the lower the shear stiffness, the lower the peak shear stress. The higher ductility adhesives also can withstand a substantially higher shear strain.

A comparison of stress/strain curves for FM 400 and FM 300, a lower temperature and more ductile adhesive, is presented in Figure A1. Note that the effective stiffness of the FM 300 is one half that of FM 400. In addition, the shear strain at failure is about six times as great for FM 300 as compared to FM 400.

A total of eight adhesives including FM 400 and FM 300 were evaluated in the initial screening test program which consisted of static shear and bell peel tests at cold, hot, and room temperatures. Four adhesives were selected from this group to perform additional static, fatigue, and environmental tests. The static and fatigue test results are summarized in Figure A2 which show FM 300K and 8GG480 to be the two best candidates. FM 300K was the final selection because 8GG480 exhibited low peel strength at cold temperatures. The environmental test data also showed FM 300K to be the best choice. It is of interest to note that the only difference between FM 300 and FM 300K is that the latter has a more open weave scrim cloth (adhesive carrier).

The next step in the design was to determine the optimum geometries of the various step lap bonded joints on the airplane. The analytical tool that was used to assist in this process was strain compatibility including the bondline shear stiffness and the titanium and graphite/epoxy axial stiffness. The adhesive is assumed to be elastic/perfectly plastic as indicated by the dashed lines in Figure A1.

The number of steps in each joint and the number of plies dropped on each step were determined by considering both strength and manufacturing. The higher strength design would have the greater number of steps, because the greater the number of steps the lower the peak shear strain. However, manufacturing considerations dictate that the minimum practical step height is about 0.02 inches. The F-18 wing skin is fabricated from 0.01 inch thick plies. Thus, no less than two plies can be dropped per step.

The length of steps and the total length of each joint were based on an average shear stress which was determined using the strain compatibility analysis.

The adhesive shear stress distribution along the bondline for the F-18 wing root step lap bonded joint is shown in Figure A3. The average shear stress on each step was computed and was limited to about one-third of the adhesive ultimate strength for the maximum load in the F-18 fatigue spectrum. The adhesive ultimate strength was determined from simple double lap specimens duplicating the length and height of the critical end steps.

The final step lap bonded joint configuration was selected based on tests of actual stepped joints such as that shown in Figure A4. A total of about 150 step lap joint specimens were tested to finalize the wing root attachment configuration. These were tested cold and hot as well as at room temperature. Additional specimens are being moisture conditioned and will be tested at elevated temperature.

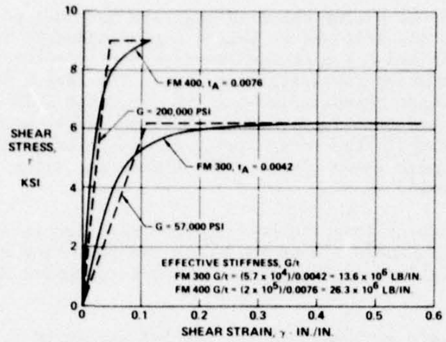


FIGURE 1A COMPARISON OF STRESS/STRAIN DATA FOR FM300 TO FM400

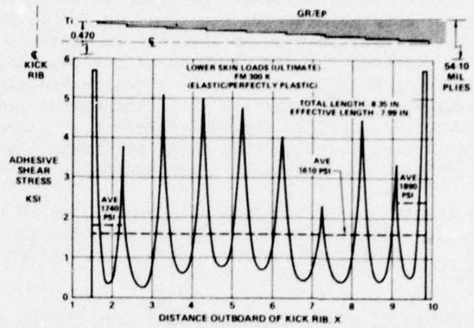
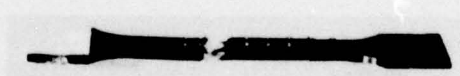


FIGURE A3 ADHESIVE SHEAR STRESS DISTRIBUTION WING ROOT BONDED JOINT

ADHESIVE	STATIC STRENGTH (PSI)	(1)RESIDUAL STRENGTH (PSI)	(2)RESIDUAL STRENGTH (PSI)
FM400	3,770	13,700 HR	140 HR
	3,740	2,270	130 HR
	4,270	3,020	1,040 HR
FM300	5,790	5,350	12,350 HR
	5,620	5,280	12,320 HR
	5,690	5,280	11,600 HR
FM300K	6,170	5,760	4,800
	6,030	5,680	4,050
	6,110	5,440	5,480
8GG480	5,950	5,770	5,250
	5,870	5,870	5,290
	5,930	5,780	5,540

Residual Strength Specimens Fatigue Tested for 24,000 hr
Highest Average Shear Stress in Fatigue Spectrum:
(1) 2,000 psi (approximate design value)
(2) 3,000 psi

FIGURE A2 ADHESIVE COMPARISONS DOUBLE LAP SPECIMENS



STEP LAP BONDED JOINT CONFIGURATION:

- 54 10 MIL PLIES
- 8.35 INCHES TOTAL LENGTH
- 27,000 LB/IN. LOAD INTENSITY INBOARD END OF JOINT
- 1880 PSI AVERAGE SHEAR STRESS ON CRITICAL END STEP

FIGURE A4 SPECIMEN SIMULATING WING ROOT BONDED JOINT

THE UK APPROACH TO THE CERTIFICATION OF COMPOSITE COMPONENTS FOR MILITARY AEROPLANES AND HELICOPTERS

by

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SUMMARY

The paper assesses the problems of the certification of composite components for military aeroplanes and helicopters as seen by the authors at the present time. An Advisory Group including members from Industry and the Official side has been set up and will guide the establishment of airworthiness requirements and clearance procedures.

Currently the UK has little information on how structures incorporating fibre composites will behave under service conditions. What is known suggests that the present airworthiness requirements could fairly readily be interpreted for fibre composites and the clearance procedures suitably adapted. One of the most important areas may well be that of structural variability and the incorporation of terms involving variability could be valuable in the clearance procedures, not least if it encouraged rigorous control of material specifications and manufacturing processes in order to achieve the lowest level of variability practicable.

It is important to find out how fibre composites fail under service conditions and what are the critical parameters. It may well be possible to include effects of environment and the like in the loading actions but some modification may be needed to the requirements.

1. INTRODUCTION

There is very little experience in the UK at the present time of how structures incorporating fibre composites behave under service conditions and particularly how they fail and what variability is associated with the failure. The time is approaching, however, when components of primary structure will require clearance for flight and acceptable procedures must be defined for this purpose. Test programmes are in hand to gather information.

Airworthiness is not a precise science. So much depends upon experience and progress is usually the result of taking a calculated step forward as an extension of the present position.

There is a variability in the strength of nominally identical structures and a variability too in the loadings to which they are subjected. The safety of airworthiness clearance lies in avoiding the overlap of the population of the loading distribution at its higher end with the lower end of the population of the strength distribution. Thus the weakest example must be able to withstand the highest load. Various attempts have been made to express this in statistical terms but the problem of the intersection of two tails of distributions is notoriously difficult and fraught with assumptions which can never be verified because the samples are always too small.

It seems likely that present structures incorporating fibre composites have a greater variability in strength than present aluminium alloy structures, although it is always possible and certainly desirable that more experience in producing the basic materials and in fabricating them into a structure could bring down the level of variability.

In times past super factors have been applied to cover structures of higher than normal variability, but recently, in the case of castings in new materials, the clearance procedures have been written to enable acceptable test factors to be calculated from a knowledge of the variability of the casting. Such a procedure has a number of attractions for structures incorporating fibre composites, not least being the incentive it gives to reduce and control the variability.

In this paper some aspects of this procedure are examined. It is already being used as the basis for a demonstration programme associated with a helicopter rotor. An alternative procedure of testing all components to an acceptable level of loading before entering service is also considered.

The views expressed in the paper are primarily those of the authors. An Advisory Group including members from Industry and the Official side has been set up to consider airworthiness requirements and clearance procedures. The authors are, in fact, Chairman and Secretary of this Group.

2. EXISTING DESIGN REQUIREMENTS AND CERTIFICATION PROCEDURES

Design Requirements for military aeroplanes and helicopters are laid down formally in AvP970 and modified or supplemented as necessary in the Specification for particular military aeroplanes or helicopters. These Requirements state the design cases and the factors of safety required to be met. Compliance with the Requirements is in the first place achieved by calculation but it is usual to require demonstration tests of static and fatigue strength culminating in separate full-scale static and fatigue tests on airframes representative of production standard. It is required that the test programme shall be discussed with Structures Department, RAE Farnborough and the responsibility falls on the Airworthiness Division.

2.1 Static Strength

There are two main static strength requirements:

(a) The structure shall not collapse before the specified ultimate load is reached.

(b) At all loads up to the specified proof load, no part of the structure shall sustain deformation detrimental to safety and moving parts essential to safety shall function satisfactorily. After removal of the proof load no effects of that loading shall remain which might reasonably cause the structure to be deemed unserviceable.

For most design conditions:

(a) the ultimate load is defined as 1.5 x Limit load and

(b) the proof load is defined as 75% of the ultimate, ie 1.125 x Limit load.

Limit load is generally that load which is expected to occur once in the life time of the aeroplane or helicopter.

It is well known that, although nominally identical and made to the same drawings, completed structures exhibit a degree of variability in strength and thus for practical reasons it becomes necessary to accept that some examples will fall below the required strength condition, although this must clearly be kept to an acceptable minimum. In 1948 Atkinson[1] proposed than an acceptable standard might be achieved if the following conditions were met:

(a) not more than 1 in 10 of a given population should have a strength below the required value and

(b) not more than 1 in 1000 should have a strength below 90% of the required value.

These conditions form a working basis for design along statistical lines and define the mean of the population according to the *coefficient of variation* (standard deviation divided by the mean).

In acceptance testing, allowance must be made for the unknown place in the population of the specimen, or specimens, tested. Atkinson[1,2] proposed a statistical interpretation based on confidence limits and Bullen[3] examined the problem by considering progressive samples. A comparison of these methods has been made by Stagg[4]. There are advantages in each method. By and large the Bullen method gives slightly lower test factors when only a few specimens are tested.

Figure 1 illustrates the test factor required when testing 1, 2 or 5 specimens from populations having a variability up to a coefficient of variation of 0.20, based on Bullen's approach. Below a coefficient of variation of about 0.05 the 1 in 10 condition is operative and at coefficients of variation above this the 1 in 1000 condition takes over.

2.2 Fatigue Strength

In acceptance testing it is normal practice to test at least one full-scale specimen to demonstrate fatigue life, damage tolerance and durability under a spectrum of loads which are expected to occur under the specified operating conditions. In order to take into account the variability of the population and the unknown position of the test specimen within that population it is usual for aeroplanes to require a factor on life. This procedure would involve an impossibly long testing time for helicopters and thus a factor on load is used instead, associated with a shorter testing time. The factors used for aluminium alloy structures are broadly consistent with the conditions proposed by Atkinson as described above.

3 AIRWORTHINESS CLEARANCE OF STRUCTURES INCORPORATING FIBRE COMPOSITES

At the present time the UK has very little experience on how structures incorporating fibre composites behave under service conditions. Most examples so far have been designed conservatively because advantages in such aspects as weight saving and ease of fabrication could be obtained without strength becoming critical. The pattern may change when attempts are made to realise the full strength and stiffness potentials.

In these circumstances, therefore, it is highly desirable to base airworthiness clearance on the results of structural tests, performed under conditions as representative as possible. In particular it seems likely that environmental effects will have to be included in the loading actions. The inclusion of the thermal cycle in the Concorde tests has shown that major engineering tasks of this nature can be tackled successfully.

For the same reasons, confidence in the results of calculations may not be very high unless an effective read-across can be obtained from similar structures, in similar environments, already cleared by test. Where clearance by calculation is desired it may be necessary to apply a super factor to parts where the confidence in the calculation is low.

Broadly it seems likely that the existing Airworthiness Requirements will function satisfactorily for the new structures. At present fibre composite structures appear to have a higher variability in strength than conventional aluminium alloy structures and this could be accommodated by building into the clearance procedures a term dependent upon the variability. Such a procedure would be an incentive to produce structures of lower variability, since the associated lower acceptance factors would enable the production of more efficient structures. Coupled with this would be the need for stringent control of the fibre and matrix materials and probably of the manufacturing processes, to ensure that the variability assumed for the clearance was not exceeded in production.

3.1 Extension of Static Strength Requirements to Structures of Higher Variability

In extending any requirement it is important to consider how it may be interpreted and how the overall safety will be affected. Usually it is desired to retain the same level of safety as has been enjoyed in the past. One of the difficulties is to define precisely what that level of safety has been.

The UK Military Airworthiness Requirements define 'standard' and 'typical' components and structures. A standard component is the weakest one which can be made, complying with the relevant drawings and material specifications, all limits and tolerances being taken in the most adverse direction. An exception is made in the case of parts fabricated from rolled sheet and strip where the nominal thickness dimension is used instead of the most adverse one. A typical component is one made in accordance with usual workshop procedure.

The Requirements call for strength calculations to be based on the assumption that standard components and structures are used. However it is recognised that strength tests will have to be made on typical and not standard structures and the Requirements call for appropriate allowance to be made for this by correcting for dimensional and material tolerances.

Until 1965, the Requirements went on to state that if such correction was impracticable, it was necessary in the test to realise factors 20 per cent greater than those specified. Where there was doubt as to the practicability of making the appropriate corrections, the case had to be referred to Structures Department RAE. Some thirty years experience of applying this requirement was reviewed in 1964 and it was felt that as in practice the 1.2 test factor had seldom been enforced it should be removed from the requirement, while making the discussion of the results of each test mandatory with Structures Department RAE to decide whether an acceptable level of strength had been achieved. This requirement was adopted in 1965.

One of the problems with this procedure is that, whereas the correction to 'standard' conditions is a reasonably effective guide at the point at which failure actually occurred, it leaves doubts about other typical parts which may have been very near failure and whose appropriate correction to 'standard' conditions might have been of significantly different magnitude.

Figure 1 shows that a test factor of 1.2 on a typical structure of unknown place in the population would be appropriate to a population having a coefficient of variation of about 0.065. Present experience shows that in most cases it is practicable to correct the test result for material and dimensional tolerances and hence the requirement tends to establish the position that the weakest sample which could be made to drawing would just satisfy the strength requirement. In practice there might still be a small tail of the population below this due to imperfections in inspection and material selection.

Comprehensive information on the variability of aluminium alloy structures is not readily available. In 1944 Marmion and Starkey[5] derived a coefficient of variation of about 0.03 from tests of nominally identical tailplanes. A recent and more extensive analysis by Freudenthal and Wang[6] has suggested a coefficient of variation of about 0.06 although this may have been influenced by the results of initial tests. It seems to the authors that modern fully developed structures in aluminium alloy are likely to have a coefficient of variation near to 0.03. Incidentally Stagg[7] in 1969 looked at the scatter in fatigue of elements and sections taken from aircraft structures.

This then is the airworthiness standard which has been set for the static strength of aluminium alloy structures and it needs to be considered how this standard can be maintained for structures incorporating fibre composites. There is some indication that these may exhibit a coefficient of variation as high as 0.15 although carefully manufactured samples from carefully selected materials have produced results much nearer those from aluminium alloy.

If it is assumed that the specimen or specimens tested are typical then test factors such as those in Figure 1 could be used to demonstrate compliance. Lower factors would be possible if the test result could be reduced with confidence to 'standard' conditions by correcting for material and dimensional tolerances. It is not yet known how practical this would be for fibre composites. For aluminium alloy structure pro rata reductions are used to allow for dimensional oversizes and for material strengths above the minimum specified value. Such simple calculations may not be realistic for fibre composites and also at the present time there are no minimum specified values for the fibre and matrix materials individually or in practical combinations. It might well be sensible for the present to base compliance on the assumption that typical samples are being tested and to use test factors such as given by Figure 1.

While such a procedure could conveniently be applied to monolithic fibre composite structures of reasonably consistent variability throughout, there would be serious difficulties in structures where parts of significantly different variability occurred as would most likely be the case for mixed structures of aluminium alloy and fibre composites. An overall test factor appropriate to the higher variability would obviously impose severe penalties on the parts of the structure of lower variability. A solution might be to seek to test the structure in smaller components in such a way as to separate the parts with major differences in variability.

An attractive alternative route to airworthiness clearance for such structures is to test each component part or structural assembly - possibly, for example, to 100% Design Ultimate Load - before release to service.

3.2 Extension of Fatigue Strength Requirements to Structures of Higher Variability

Similar considerations can be applied to fatigue strength as have been discussed in the previous section for static strength, but the procedures are complicated by the different approaches to clearance on the one hand for aeroplanes and for helicopters and on the other hand for safe-life structures and for fail-safe or damage tolerant structures. Associated with the fatigue clearance must be a knowledge of the static strength required at all times during the life of the structure and means must be established to demonstrate this along the lines of the previous section.

For aeroplanes with a safe-life structure, the fatigue life is defined as that life during which the general level of safety is not appreciably lowered by fatigue and a reasonable standard of serviceability is maintained. Usually the acceptance test is taken to failure under the loads chosen for the test spectrum. Experience has shown that it is desirable to include loads in the spectrum up to the level of those which occur at about 10 times during the life of the aircraft.

Guidance is given in the Requirements on the factors to be used to derive the safe-life from the life under test of one or more specimens. These factors are based on the assumption of a normal distribution of log endurance and are derived using the method proposed by Bullen[3]. It follows, therefore, that similar factors can be deduced from a knowledge of the variability of fibre composite structures in fatigue.

For aeroplanes with a fail-safe or damage tolerant structure a fatigue test is necessary to give information on the location of cracks and on their rate of propagation. A fail-safe structure is defined as one which retains, after initiation of any fracture or crack, sufficient strength and stiffness for the service operation with an acceptable standard of safety until such fracture or crack is detected by inspection. The inspection period must thus be chosen such that a detectable size of crack will not propagate to critical proportions during the inspection period. In order to allow for variability, it is recommended practice to base the inspection period for aluminium alloy structures on one third of the rate of crack propagation achieved in the test. Clearly this is amenable for adaptation to fibre composite structures when the appropriate variability is known.

One of the items governing the critical proportions of the crack is the residual static strength of the cracked structure. For aluminium alloys it is recommended that the static strength of the damaged structure should never be allowed to fall below 80% of its design ultimate strength. The length of the inspection period should be fixed in relation to the rate of the spread of damage, as described above, so that the strength of the structure will not fall below this value before the damage is found. This value of 80% is regarded as the minimum strength that will allow the pilot, who may be unaware of any structrual weakness, to fly the aeroplane to its full operational standard with reasonable safety during the limited time while the weakness is undetected.

When more is known on how fibre composites structure fail it will be possible to assess whether 80% would also be a reasonable value for them. Once an appropriate value is established the arguments relating to the demonstration of static strength described in the previous sections will apply.

It is worth noting in passing that it is always desirable to run the fatigue test on a fail-safe structure to the same test life as a safe-life structure would have required, in order to ensure that there are no unexpected failures with safe-life characteristics hidden in the nominally fail-safe structure.

The problem of fatigue test acceptance procedures for helicopters is more complex. A factor on life is not feasible since the number of cycles involved is so high and the S-N curve is consequently very flat. Thus a factor on load has been adopted instead to cover both the variability aspect and to reduce the total testing time. The Requirements give recommendations for dealing with this for steel and aluminium alloy parts and for testing different numbers of specimens. These are based on statistical arguments and again are amenable for adaptation to fibre composite parts when the appropriate data are known.

Clearance of helicopters by fail-safe procedures involving rates of crack propagation and residual static strength are being dealt with as for aeroplane structures. Again these procedures would be amenable for adaptation for fibre composite structures.

4 CONCLUSIONS

This paper assesses the problems of the certification of composite components for military aeroplanes and helicopters as seen by the authors at the present time. An Advisory Group including members from Industry and the Official side has been set up and will guide the establishment of airworthiness requirements and clearance procedures.

Currently the UK has little information on how structures incorporating fibre composites will behave under service conditions. What is known suggests that the present airworthiness requirements could fairly readily be interpreted for fibre composites and the clearance procedures suitably adapted. One of the most important areas may well be that of structural variability and the incorporation of terms involving variability could be valuable in the clearance procedures, not least if it encouraged rigorous control of material specifications and manufacturing processes in order to achieve the lowest level of variability practicable.

It is important to find out how fibre composites fail under service conditions and what are the critical parameters. It may well be possible to include effects of environment and the like in the loading actions but some modification may be needed to the requirements.

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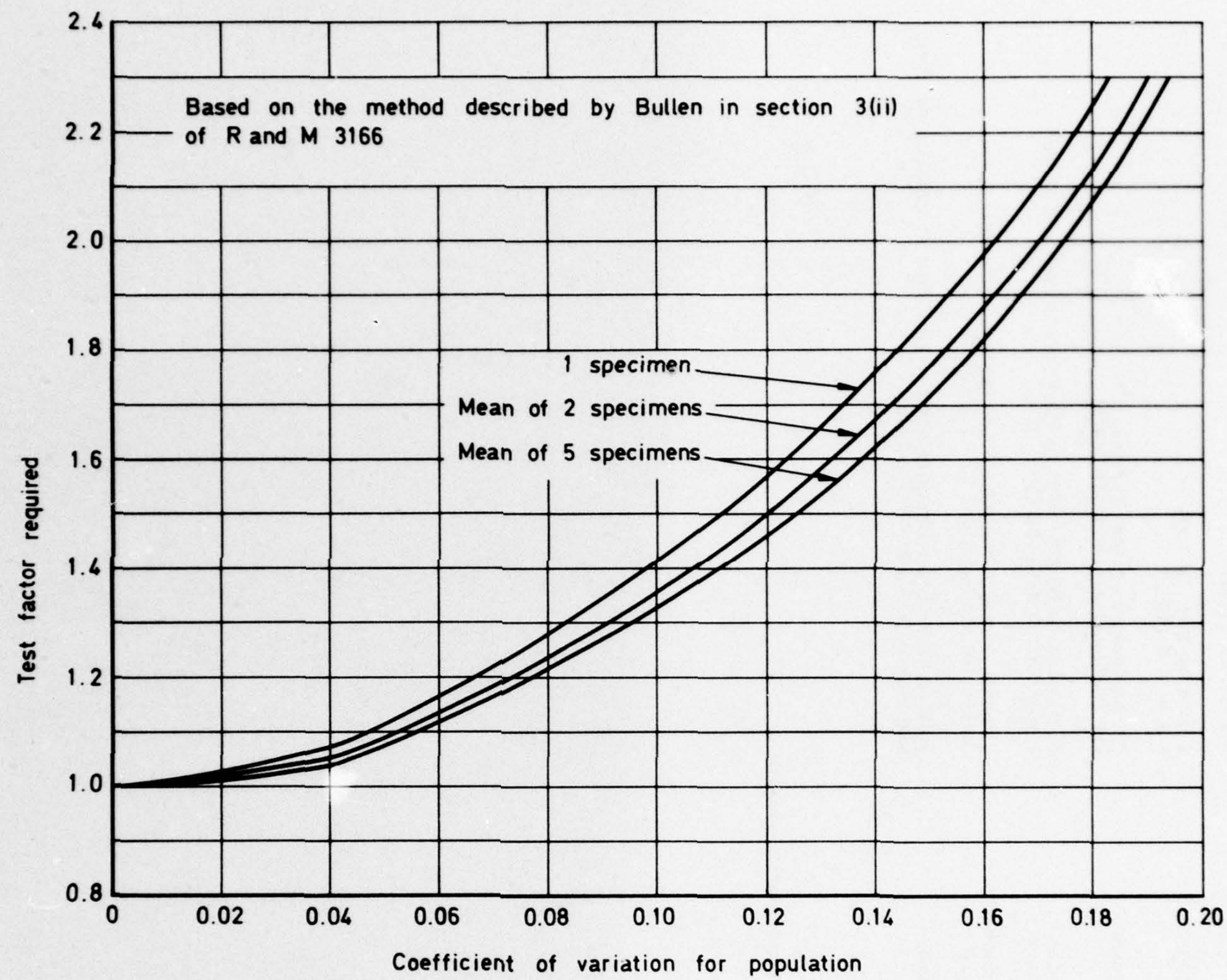

Fig 1 Test factors required to satisfy the conditions that

(i) Not more than 1 in 10 of population below required strength
(ii) not more than 1 in 1000 of population below 90% required strength

LA CERTIFICATION DES STRUCTURES D'AVIONS EN COMPOSITES

par

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1 - INTRODUCTION

Les structures composites à base de fibres de carbone ou de bore ont atteint le stade des applications en série dans l'Aéronautique. Il est donc nécessaire de mettre au point des procédures de certification permettant d'assurer la sécurité des vols.

C'est évidemment un très vaste problème et il ne saurait être question d'en aborder tous les aspects dans le cadre de cet exposé.

Les auteurs se sont donc volontairement limités au cas des structures d'avions en distinguant les programmes expérimentaux et les pièces construites en série.

Toutefois, afin de placer le problème dans son contexte général, il est utile de rappeler quel est l'objectif d'une certification et quelles sont les méthodes appliquées en FRANCE sur les structures métalliques.

2 - OBJECTIF DE LA CERTIFICATION

La certification , appliquée à un avion militaire ou civil, consiste à s'assurer que cet avion dispose d'un niveau de sécurité acceptable pendant sa durée d'utilisation. La notion de niveau de sécurité est liée à celle de probabilité d'accident, donc de rupture catastrophique pour une structure. Le niveau de sécurité exigé par l'Autorité de Certification est défini par un règlement. Ce règlement peut également préciser des méthodes de démonstration acceptables pour la certification.

Compte-tenu de cette définition, la certification sur un plan général consiste à effectuer un certain nombre de travaux avant la mise en service de l'avion et pendant son utilisation. Dans le cas particulier de la résistance de la structure, ces travaux ont pour objectif de prévoir cette résistance ainsi que son évolution dans le temps sous l'effet des conditions d'utilisation. Ils doivent être complétés par des contrôles de fabrication et des inspections en service permettant de s'assurer que l'état réel de la structure est conforme aux prévisions.

3 - CERTIFICATION DES STRUCTURES METALLIQUES EN FRANCE

Depuis plusieurs années, la FRANCE a utilisé une procédure de certification des structures métalliques qui n'a rien d'original mais qui s'est avérée satisfaisante jusqu'à présent. Le principe de cette procédure est très voisin de l'Aircraft Structural Integrity Program de l'US AIR FORCE (ref. (1)).

Cette procédure comporte les phases suivantes :

Phase I	: Définition	(Conditions de calcul (Sélection des matériaux (Choix des procédés de fabrication
Phase II	: Développement	(Calcul des efforts (Essais de développement (Contraintes admissibles (Dimensionnement
Phase III	: Justification	(Résistance statique (Résistance en fatigue (Contrôle de fabrication
Phase IV	: Comportement en service	(Programme de maintenance (Utilisation réelle (Rapports d'incidents

L'intervention des Services Officiels se fait à chaque phase dès le début du programme, ce qui permet d'effectuer en temps utile les vérifications nécessaires.

En particulier, les essais statiques et les essais de fatigue d'ensemble sont systématiquement effectués dans un Centre d'Essai Officiel.

En principe, les essais statiques sont effectués assez tôt pour que les modifications éventuelles soient appliquées sur le 1er avion de série. Les essais de fatigue, au contraire, sont effectués sur une cellule conforme à la série. Dans certains cas ils sont précédés d'essais partiels d'éléments de grande dimension si la technologie utilisée présente des risques importants.

L'interprétation des essais statiques et de fatigue lorsque les conditions d'utilisation évoluent se fait par calcul à partir des données d'utilisation réelle.

La vérification de la qualité de la fabrication se fait par les méthodes de contrôle appropriées et approuvées par le Service Technique Aéronautique.

4 - CERTIFICATION DES STRUCTURES COMPOSITES CONSTRUITES EN SERIE

La certification des structures composites construites en série est prévue suivant un processus analogue à celui des structures métalliques. Toutefois au cours de chaque phase il sera tenu compte du caractère particulier de ces matériaux.

4.1. - Contraintes admissibles

La détermination des contraintes admissibles dans les matériaux composites comporte deux étapes :

- détermination des caractéristiques unidirectionnelles
- détermination des caractéristiques d'un élément multicouche compte tenu de l'orientation des différents plis.

La première étape est analogue à ce qui est fait pour les matériaux métalliques : il faut définir des essais de réception et retenir une valeur minimale de réception. A partir de cette valeur minimale, on pratique généralement une réduction forfaitaire pour tenir compte de divers facteurs (dispersion, vieillissement, etc...).

La deuxième étape est particulière aux composites et permet au bureau d'études un degré de liberté supplémentaire pour optimiser la structure (ref. (2)). Les résultats peuvent être obtenus par calculs à partir des données précédentes.

De plus, il est nécessaire de déterminer expérimentalement les contraintes admissibles dans les assemblages.

4.2. - Résistance statique

Les méthodes de calcul de résistance statique des structures composites sont en cours de développement et ne devraient pas présenter de difficultés importantes par rapport aux structures métalliques en dehors de leur plus grande complexité.

Toutefois il n'est pas envisagé d'aller jusqu'à la suppression des essais statiques, en particulier pour les raisons suivantes :

- Nécessité de connaître les marges réelles de la structure
- Possibilité de mettre en évidence des défauts de fabrication
- Nécessité d'avoir une référence pour les essais sur structures vieillies (voir § 4.3).

En conséquence, les essais statiques doivent être faits jusqu'à rupture et avec contrôle des caractéristiques des matériaux de la pièce essayée.

4.3. - Résistance en fatigue

Le terme résistance en fatigue apparaît tout à fait impropre lorsqu'on l'applique aux matériaux composites. En fait il ne s'agit plus seulement des effets des charges mécaniques répétées, mais encore des effets de la température, de l'humidité, du temps, etc ... qui sont généralement négligés lors de la certification des structures métalliques.

Une première difficulté réside dans l'absence d'une corrélation bien établie entre les essais accélérés en laboratoire et le comportement en service réel. Les essais accélérés sont utiles pour déterminer quels sont les paramètres importants susceptibles d'agir sur la résistance de la structure. Ils devraient être complétés par des essais de longue durée dans des conditions réalistes pour obtenir des résultats utilisables pour une certification.

Devant cette difficulté, la position du responsable de la vérification est inconfortable : doit-il faire confiance à des résultats d'essais "accélérés" douteux ou doit-il demander des essais "réalistes" qui n'aboutiront que plusieurs années plus tard ? La position que nous avons adoptée en FRANCE est qu'il faut lancer le plus tôt possible des essais dans des conditions réalistes afin d'avoir toujours de l'avance sur l'utilisation sur avion *.

Une autre difficulté consiste à définir et à détecter l'endommagement d'une structure composite. Alors que les structures métalliques avaient la propriété de s'endommager sous forme de criques plus ou moins faciles à détecter, les structures composites peuvent s'endommager d'une façon beaucoup plus variée et parfois impossible à détecter par des moyens non destructifs (détérioration de la résine, délaminage, etc ...).

* Une difficulté analogue a été rencontrée lors de la certification de CONCORDE, voir ref. (3).

Actuellement la seule façon de détecter à coup sûr un endommagement est d'effectuer un essai statique à rupture et de mesurer la perte de résistance par rapport à une structure non endommagée. Il faut donc pouvoir effectuer plusieurs essais destructifs pour connaître l'évolution de la résistance en fonction du temps : ceci condamne en pratique l'essai d'ensemble au profit des essais sur éprouvettes de dimensions réduites.

On peut donc espérer, en utilisant un assez grand nombre d'éprouvettes représentatives de la structure à certifier, obtenir l'évaluation de la résistancestatique en fonction de la durée d'utilisation. Malheureusement cela représente un volume d'essais très important pour des résultats qui ne seront disponibles qu'après plusieurs années alors que les avions seront déjà en service. A l'évidence cette solution n'est pas satisfaisante et il est nécessaire de poursuivre les études théoriques et expérimentales pour améliorer les connaissances sur le vieillissement des composites.

Une autre méthode possible consisterait à utiliser l'avion lui-même comme laboratoire et à effectuer des essais d'endurance sur les premiers avions série. C'est une méthode couramment utilisée par les Soviétiques, mais il est douteux qu'elle puisse être acceptée facilement par les utilisateurs occidentaux militaires et encore moins civils ! Tout au plus peut-on utiliser dans ce sens des programmes expérimentaux et des avions prototypes pendant un temps limité.

Par contre, il ne faut pas se priver des renseignements qui peuvent être tirés de l'expérience en service sur un grand nombre d'avions. En effet il est possible de mettre au point un programme de contrôle continu des structures composites en utilisation, comportant par exemple :

- des inspections périodiques par méthodes non destructives
- des essais destructifs sur "éprouvettes suiveuses"
- des essais destructifs sur éléments de structure, lorsque ceux-ci sont facilement démontables (gouvernes, aérofrein, etc ...).

Un tel programme est nécessaire au moins pour les premières structures construites en série, afin d'obtenir des données de base sur le vieillissement dans des conditions réelles, et tant que les méthodes de prévision n'auront pas été perfectionnées.

4.4. - Problèmes particuliers aux composites

Outre les problèmes de vieillissement, un certain nombre de phénomènes physiques produisent des effets plus sensibles sur les composites que sur les structures métalliques et doivent être étudiés dans le cadre de la certification, par exemple :

- Tolérance aux dommages (propagation des défauts)
- Résistance aux chocs (outils, éclats, oiseaux, etc ...)
- Résistance à la"corrosion" (pétrole et produits divers)
- Résistance à la foudre
- Métallisation.

Ces phénomènes prennent plus ou moins d'importance suivant la localisation des composites dans la structure. Ils doivent être étudiés pour chaque cas particulier, et peuvent conduire à des modifications technologiques (renforcements, peintures, etc...) ou à des restrictions d'utilisation.

Le problème de la tolérance aux dommages mérite une attention particulière en raison de la grande variété des défauts envisageables (fissures, décollements, délaminage, impacts, etc...).

4.5. - Contrôle de fabrication

Le contrôle de fabrication est un élément essentiel dans le processus de certification, car il doit permettre d'assurer que les structures réelles sont conformes à la définition prévue par le bureau d'études. Il doit donc s'exercer au niveau des matériaux de base, du cycle de fabrication, des éléments terminés.

Sans entrer dans le détail des méthodes de contrôle qui sortirait du cadre de cet exposé, il faut cependant rappeler qu'il est nécessaire de mettre au point des méthodes de contrôle adaptées aux structures composites, tant sur le plan du contrôle de fabrication que sur le plan du contrôle en utilisation.

5 - TENTATIVES SUR PROGRAMME EXPERIMENTAL

A ce jour un seul programme est suffisament avancé dans son développement pour pouvoir être pris comme exemple. Il s'agit de la réalisation des empennages horizontaux de l'avion MIRAGE F1. Cette étude a été demandée par le Service Technique Aéronautique à la Société AMD/BA afin d'essayer en vol une pièce de structure en composite et d'évaluer les problèmes concernant la réalisation, les essais et la certification. Il n'était pas dans l'idée du STAé de préparer un développement série et seuls quatre demi empennages ont été commandés. Ils sont réalisés en structure sandwich avec un Nida en alliage aluminium et des peaux, rapportées par collage, en matériaux composites fibre de bore-résine époxy. Les introductions d'efforts au niveau de l'attache principale se font par un insert titane qui transmet les efforts par l'intermédiaire d'escaliers collés aux revêtements carbone.

Les différentes étapes de la certification se sont déroulées au fur et à mesure du déroulement des travaux.

5.1 - Choix du matériau de base et détermination des contraintes admissibles

Après un programme d'essais pour le choix, il y a eu une caractérisation complète du matériau choisi comprenant des essais avec et sans vieillissement. Ce programme avait pour but :

- de vérifier que le matériau ne possèdait aucune "tare"rédhibitoire à son utilisation
- de fixer des valeurs pour le contrôle réception
- de déterminer les contraintes admissibles dans ce matériau.

5.2 - Calculs et essais pour le dimensionnement

Des essais de principe ont été réalisés pour déterminer les règles de dimensionnement, comprenant plus particulièrement des essais de flexion cisaillement, des essais de liaison, des essais d'introduction d'effort. Puis par application des résultats obtenus il y a eu :

- les calculs statiques dans les cas majorants de chargement réel
- les calculs de rigidité et des modes vibratoires
- l'établissement des liasses de fabrication.

5.3 - Vérification par un essai technologique.

La vérification de la validité des calculs et de la technologie utilisée a été réalisée à travers un essai statique jusqu'à rupture d'un caisson technologique représentant une portion de l'empennage (comprenant la zone de l'attache principale).

L'essai s'est passé en plusieurs fois. D'abord jusqu'à la charge limite puis jusqu'à rupture. Ce caisson a tenu 2,4 CL prouvant la qualité de la fabrication. Le surdimensionnement s'explique par le fait que ce sont toujours les valeurs les plus pessimistes qui ont été utilisées. Cependant,pour plus de sécurité, aucune modification n'est intervenue pour la réalisation de l'empennage lui même.

5.4 - Fabrication d'un demi empennage:contrôle fabrication

Au cours de la fabrication du demi empennage, il a été mis au point le dossier de contrôle qui a été livré avec l'empennage. Il comprend :

- le contrôle réception matière première
- le contrôle assemblage
- le contrôle collage
- le contrôle final de la pièce.

Ce dossier a été accepté dans l'état actuel des choses mais il est à noter que c'est un contrôle particulièrement lourd et qu'il ne permet pas malgré tout de donner une garantie totale sur la qualité de la pièce. Ce demi empennage a par la suite été soumis a un essai statique qui par palier l'a amené jusqu'à la rupture. Celle-ci s'est produite à 2,55 CL. Cela confirme donc les résultats obtenus sur le caisson technologique, et il est probable que les valeurs initiales devront être réexaminées.

5.5 - Fabrication de deux demi empennages:essais en vol

Deux autres demi empennages ont été réalisés dans les mêmes conditions pour essais en vol. Afin de confirmer le dossier de contrôle de ces empennages, ces derniers ont été essayés en statique jusqu'à la charge limite. Puis des essais de vibration sur avion ont eu lieu afin d'identifier les modes qui se sont avérés comparables a ceux des structures métalliques et enfin les premiers essais en vol ont eu lieu pour ouvrir le domaine. Ces essais sont encore en cours. Par la suite,des empennages seront montés sur un avion de l'Armée de l'air,mais il reste à définir une méthode de contrôle et de maintenance.

5.6 - Fabrication d'un demi empennage pour essais de fatigue-vieillissement.

L'empennage est en cours de réalisation et cet essai reste à définir.

6 - CONCLUSION

La certification des structures composites pose des problèmes nettement plus complexes que les structures métalliques classiques.

Parmi les problèmes les plus importants, il faut insister sur le problème du vieillissement pour lequel des méthodes de prévision satisfaisantes n'existent pas encore. En conséquence, il n'est pas encore possible de garantir aux utilisateurs des durées de vie homogènes avec celles des structures métalliques. De plus, il sera demandé des contrôles supplémentaires qui se traduiront par un programme relativement contraignant d'inspections en service. Ceci risque de limiter fortement l'intérêt économique de ce type de structures, particulièrement pour l'Aviation Civile.

Pour surmonter ces difficultés, des programmes d'études théoriques et expérimentales ont été lancés dans plusieurs pays. Afin d'obtenir des résultats comparables entre eux, il serait fort utile de standardiser les méthodes d'essai de vieillissement "réalistes", dans le même esprit que le programme "FALSTAFF" pour les essais de fatigue. Ceci pourrait faire l'objet d'un programme d'activité future du Panel "Structures et Matériaux" de l'AGARD.

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STRUCTURAL ASSURANCE OF ADVANCED COMPOSITE COMPONENTS FOR USAF AIRCRAFT

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SUMMARY

Modern high performance composite materials such as epoxy resin reinforced by graphite fibers are making a transition from previous applications on secondary structure to increasing use on primary structure, that is, safety-critical components. Procedures to assure structural function, safety, and durability are undergoing a corresponding reconsideration. The U.S. Air Force has been flying advanced composite secondary structure since 1967, and some production primary structures since 1973. Structural assurance procedures have been modified to account for recently recognized sensitivity to environment. For composites in service now, confidence is derived primarily from their conservative design and high strength margins demonstrated by test. Additional confidence is being gained from in-service inspections and further structural tests. Components currently approaching service have benefited from recent data and have been qualified by programs which include, in various forms, the effects of moisture, temperature and load history on structural safety and durability. For composites in the future, research in structural integrity is planned to yield validated accelerated test procedures to reduce qualification costs and life prediction analysis for individual aircraft tracking and force management.

I. INTRODUCTION

Modern high performance composite materials such as epoxy resin reinforced by graphite fibers are making a transition from previous applications on secondary structure to increasing use on primary structure. Procedures to assure structural function, safety, and durability are undergoing a corresponding reconsideration. The current U.S. Air Force structural integrity program encompasses design, analysis, test, and program management actions developed to substantially diminish the reoccurrence of past structural problems on future aircraft programs. If problems of the same seriousness were to occur with composite materials, the effect would be to discourage selection of composites for widespread application to primary structure. Structural assurance policies for composite flight components are therefore conservative, and a comprehensive technology development program provides continuing improvements in methodology.

II. THE AIRCRAFT STRUCTURAL INTEGRITY PROGRAM

Military Standard 1530A (USAF), which was adopted in December 1975 to document new Air Force requirements for structural assurance of aircraft, applies to composites in most respects, although certain detailed provisions are intended for metallic structure. Three major elements of the Standard to be discussed in this paper are:

A. Qualification of the design configuration, including static strength, damage tolerance, and durability.

B. Individual article acceptance procedures.

C. Force management actions.

The required static strength is 1.5 times limit load for each critical design loading condition. The damage tolerance is, in simplified concept, the following: Multiple independent load path structure must safely carry flight loads, with one of the major load paths broken, until discovery of the damage is certain. For monolithic structure, flight loads must be carried for a specified length of time, typically twice the time between inspections, in the presence of a growing flaw of specified initial size. Qualification consists of both analysis and testing. For metallic structure the test phase has included an ultimate load test of a full scale assembled airframe to demonstrate static strength; an accelerated cyclic loading fatigue test of another assembled airframe to prove durability; and separate cyclic loading tests of full scale monolithic components with induced flaws to meet the damage tolerance requirements.

III. HISTORY OF USAF COMPOSITES

Qualification procedures for advanced composite components have been evolving during the past several years. In 1967 the first advanced composite components were installed on aircraft for flight demonstration. These were secondary structure, such as trailing edges, doors, rudders, and slats, and they were qualified by the same procedures as had

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been used for metallic components, including accelerated fatigue tests. In 1973 the first production composites entered service, and these also were qualified like the metallic parts.

In 1975 two things happened. First, some of the demonstration composite parts in-service were returned and examined. Some unexpected problems had occurred. The parts in question consisted of a sandwich configuration with composite face sheets and aluminum honeycomb core. Moisture had reached the core resulting in corrosion and adhesive bonded areas had delaminated. Also a number of components had experienced foreign object damage. Two components failed in flight but the aircraft were safely landed. The other news in 1975 was increasing attention to data which showed that the epoxy resin used in advanced composite structures absorbs substantial amounts (1-2 percent) of water from the environment. The effect of this absorbed moisture is a loss of modulus of the resin, especially at elevated temperatures. The structural effect on the compression and interlaminar shear strength of a composite is a severe degradation within 30 C of the cure temperature and a moderate degradation within 60 C of the cure temperature. These environmental effects, one from parts in service and the other from laboratory tests, suggested that the qualification procedures employed up to then were not fully satisfactory for composites. Beginning in 1976, the Air Force introduced new qualification procedures for the composite parts which were approaching service. The primary structural components already in service were reviewed to consider other precautionary actions. In addition, a comprehensive review was conducted of the state of the art of the technology used for qualification of composites. The result was a major reorientation of short range and long range Air Force research and development in advanced composites. A goal was established to obtain economical, validated, test procedures for qualification of future composites. A parallel goal was derivation of analytical life prediction methods, because this capability is essential for effective management of aircraft in service.

IV. PAST QUALIFICATIONS: COMPOSITES CURRENTLY IN SERVICE

Although secondary composite structures such as fairings, speed brakes, and reinforcement of metal parts are used on several USAF aircraft, the more serious concerns about structural assurance apply to primary structure, and this is addressed here. Figure 1 shows a sketch of a cross section and structural detail typical of both the horizontal and vertical stabilizers of a current fighter aircraft. The leading and trailing edges are conventional aluminum face sheet sandwich construction while the main torque box of both stabilizers has boron/epoxy covers. An interesting feature of this construction is a monolithic metallic frame, which goes all around the boron laminate. The laminate is laid on to this stepped frame and bonded in place first; then the assembly is bonded to the aluminum honeycomb core. Like most advanced composite parts in use, these laminates were designed with very low strain levels. When these components were qualified in 1973, as previously mentioned, the qualification tests consisted of a room temperature test for static strength, just as for metals. Damage tolerance was based on the slow damage growth concept with analyses and limited full scale tests. Durability was addressed using a room temperature accelerated fatigue test. Each individual article was accepted by non-destructive inspection. In 1976, after recognition of the degrading effect of the environment, additional efforts were established. First, engineers from the Air Force Materials Laboratory examined selected components on site, using portable non-destructive testing instruments. Second, plans have been made to return one horizontal stabilizer from service and perform additional structural tests. A suitable article is available from an aircraft which has experienced aerodynamic heating and long term exposure to humid conditions on the ground.

V. CURRENT QUALIFICATIONS: COMPOSITES APPROACHING SERVICE

A. Outer Wing Panel

Figure 2 shows the structural arrangement for the outer wing panel on an aircraft with a folding wing. It consists of an upper and lower cover plus internal ribs and spars. Each of the covers and each of the internal members is itself a honeycomb sandwich structure. On the covers, the face sheets are a hybrid graphite/boron/epoxy laminate; on the internal members the face sheets are graphite/epoxy. The core of each sandwich is aluminum honeycomb throughout. This outer wing panel was produced in only limited quantities as part of a manufacturing development program. In fact, only eight articles have been installed on aircraft in service. However, this component is mentioned here because it received a qualification which was similar in several ways to a qualification program required for a full production run. A static strength test was performed at room temperature. Damage tolerance was based on the slow damage growth concept and durability was demonstrated by a simulated environment fatigue test described more fully below. This is largely a bonded structure, although some fasteners are employed. The acceptance of individual articles was based on a proof loading test of every article. Additional structural evaluations are planned. A unique feature is provision of coupons of the same laminate carried on the aircraft in service. These coupons will be weighed periodically to measure moisture absorption. These eight articles will be examined in the field by engineering non-destructive testing specialists. Finally, there are plans to perform structural tests of articles returned from service.

Details of the qualification testing and results are as follows. A static test article was tested at room temperature, with no effort to control humidity. The failure load was 1.88 times limit load. But when conditions for the fatigue test were being discussed in 1975, the effects of moisture had become known, so the fatigue test was modified to include moisture and simultaneous moisture/elevated temperature exposure. First, the entire article was exposed to moisture for 30 days at 66 C and 95 percent relative humidity; then it was subjected to cyclic loading at room temperature for 1470 hours. Following that it received 30 hours of cyclic loading at 56 C. Those 30 hours represent the total elevated temperature exposure time the panel would have experienced during the same 1470 hours of flight time. There was no failure of the composite parts during the fatigue test. The same article, thereupon, was subjected to a residual strength test at 56 C. That test resulted in failure at 1.97 times limit load.

B. Small Vertical Stabilizer and Small Horizontal Stabilizer.

The structural arrangement for a small vertical fin is represented in Figure 3. It consists of conventional aluminum ribs and spars as the substructure and graphite/epoxy laminates for the covers. The covers are bolted to the substructure. On the same aircraft, Figure 4 shows the different construction of the horizontal stabilizer. The covers are also graphite/epoxy laminates, but the substructure is full depth aluminum honeycomb core, plus a titanium spar/spindle. The front and rear edges are closed out with light weight spars. In the area at the root of the titanium spar, bolts are employed to reinforce the bonded joints between the composite covers and the titanium.

The major tests in this program are not yet complete, but the procedures have been selected. Full scale static and durability qualification tests will be conducted. The decision was made to conduct the full scale durability test without simulating moisture, temperature and relative loading conditions. However, a large program of subcomponent tests is being conducted including the effects of moisture and temperature. These tests will comprise a portion of the qualification. Damage tolerance for both components is based on the slow damage growth concept. Acceptance of individual production articles is based on nondestructive inspection for the bolted vertical fin. For the development aircraft currently being fabricated, acceptance of the primarily bonded horizontal stabilizer is based on a proof load test of every article. Additional structural data are anticipated from a research program discussed later which will subject one of the horizontal stabilizers to a durability test which includes moisture, temperature, and real-time loading.

C. Large Horizontal Stabilizer

Subcomponents of this unique composite component are shown in Figure 5. This is the first large component with a composite rather than metallic substructure. The internal members are graphite/epoxy laminates. The lightly loaded spars are configured as beams with curved webs resembling a sine wave in cross section; in addition to utilizing the stabilizing influence of curvature to a shell, the sine wave spar can be attached to adjacent structure by a single row of bolts through the centerline of the cap. More heavily loaded spars resemble I beams built up from two channel members. The covers themselves are primarily graphite/epoxy; boron fibers are incorporated in some areas to provide additional strength and stiffness. The whole structure is bolted together.

The program of qualification tests has been completed as follows. The static test article was loaded at both room temperature and elevated temperature but there was no attempt to condition the full scale test structure for moisture absorption. The full scale fatigue test was conducted at room temperature, like the previous example. However, these tests were supplemented by a very large group of subcomponent tests involving both areas of the face sheets and the internal structural members. These were static and fatigue tests, at room temperature and elevated temperature with previous moisture absorption by exposure to high humidity, and the test included simulated in-service environmental temperature-humidity cycles. Damage tolerance is based on the slow damage growth concept, and in this program there were additional subcomponent tests of structure with flaws and a simulated environment to prove the capability for damage tolerance. Individual article acceptance is based on non-destructive inspection.

Damage tolerance requirements were satisfied, for this component, by a group of tests of flawed subcomponents. The basic requirement is that the component perform safely for a specified length of time even if the part contains a flaw. The design concept was to limit growth of the flaw rather than the alternative of providing independent separate load paths. For the purpose of demonstrating satisfaction of the requirements, the flawed areas were considered to be non-inspectable. Damage tolerance analysis is not yet developed; the demonstration was completely by test. The test methods include static tests, cyclic loading tests, and residual strength tests. The tests included the effects of absorbed moisture and sequenced temperature and load. The flaws selected for test were at least twice as large as the smallest consistently detectable by non-destructive inspection. Types of flaws included several which had been previously seen in fabricated articles and a few other hypothetical but potentially severe conditions as follows:

Voids 1.2 x 1.2 cm
Spread Tows 0.8 x 2 cm
Surface Scratches 2 plies deep
Stringlike voids 0.1 x 12 cm in the flange radius of ribs and spars
Excessively thick bond lines
Broken fibers at the back side of a drilled hole
Repair patches

The specimens are shown in Figure 6. They included large tension components with a section of skin and substructure; composite-to-metal bolted joints; bending of the flanges of a beam, and simple beam loading of sine wave spars to test the shear capabilities of the web. Figure 7 shows the rather elaborate environmental spectrum which was employed in the fatigue test. First, all specimens were exposed to 77 C at 95 percent relative humidity until moisture content reached 1 to 1.5 percent. The cycle shown represented the time at temperature expected in 100 flights, and each cycle was completed in the laboratory in 24 hours.

The damage tolerance tests were successful. All specimens completed the environmental fatigue tests without failure and without apparent growth of flaws. A residual strength of 1.0 times limit load after fatigue testing was required. No specimen failed at less than 1.5 times limit load in the residual strength test which was conducted at elevated temperature. It was therefore concluded that the composite structure tested is tolerant of the types of fabrication defects simulated, when subjected to the usage spectra employed in the tests. The flaws employed are detectable by non-destructive evaluation techniques; acceptance of individual flight articles may therefore be based on non-destructive inspection. However, the response of the structure to other flaws or to other environmental and load spectra can be determined only by additional testing.

VI FUTURE QUALIFICATIONS: TECHNOLOGY DEVELOPMENT

The qualification of USAF composites in the future will be based on information being developed in a "Roadmap for research for advanced composites." These necessary research tasks were selected as a result of a formal assessment of currently available methodology for structural qualification and subsequent force management. Table 1 and Table 2 summarize that assessment, comparing the status of the methodology for metals and composites. Analysis for static strength is satisfactory for both metals and composites and the full scale static test is well established for both. For metals, damage tolerance analysis is addressed by fracture mechanics procedures. But there is no calculation method for composites analogous to fracture mechanics. Damage tolerance tests for metals are routinely performed by introducing cracks at critical locations, while characterization of flaw configuration and definition of critical test conditions for composites are quite limited. Except for corrosion and weakening of adhesive bonds to metals, the conservatively designed advanced composite parts in actual service experience have not exhibited significant degradation of structural function. As design procedures change with composites, degradation modes observed in the laboratory could conceivably occur in service, and this is a concern. Analysis for durability of composites is simply not available. Such analyses for metals are routine either by conventional fatigue cumulative damage theories, or by assumption of propagation of a very small, undetectable crack existing from the time of fabrication. Finally, the durability test for metallic component is the accelerated loading fatigue test. Its validity has been established. For composites, the influence of environment may be strong, and the synergistic interaction of environment and real time loading history may be equally strong. An accelerated full scale durability test with appropriately simulated environmental conditions has not been evolved nor validated.

Force Management was named earlier in this paper as a major task in the Aircraft Structural Integrity Program. The Force consists of the group of operational aircraft of a specific design. The aspects of Force Management of concern here, and listed in Table 2, are Repair Procedures, Inspection Methods, Inspection Accept/Reject Criteria, and Individual Aircraft Tracking. Repairs for metallic components are well understood, including the many plausible appearing methods which have been discovered, by hard experience, to be unacceptable. But in composites only a limited range of repair configurations have been developed, and even these have not been proved over the years as the repairs to metals have. Non-destructive field inspection capabilities are limited for both metals and composites. This is no surprise; it really is difficult to conduct a sophisticated inspection under field conditions. But in metals we are reasonably certain whether a crack of a specific size should be rejected after it is found. In composites, when a delamination or other flaw is discovered, its structural effect and hence the criterion for rejection is often not known.

The most important line in Table 2 is Individual Aircraft Tracking, that is, maintaining a continuous record of the life expended and the useful life remaining for every aircraft in the force. This function is crucial because there are wide variations in usage. For example the USAF employs a large force of one airplane designed as a trainer. Most of the aircraft are indeed used as trainers, but several have been adopted for service as fighter tactics trainers, experiencing far more severe usage. Without the predictive capabilities of fracture mechanics, the safety of these severely used aircraft would be difficult to establish. The tracking task includes selection of parameters which, measured during the service life of an airplane, characterize its structural history. For metals, the loads, and sometimes the load sequence, are sufficient. For composites, loads are also needed, but so are temperature and humidity, and possibly their sequences. However, even if these parameters were recorded, tracking of composites would not be practical because the method to calculate expected life is not yet known.

After the formal assessment identified the missing capabilities in methodology for qualification and force management of composites, research programs were conceived to develop those capabilities. The result was the Structural Integrity Roadmap for Composite Structures, which is a list of research programs with schedules. An outline is given in the Appendix. After completion of the Roadmap research, it is expected that validated test procedures and analysis will be available for economical qualification and force management of composite components. A major program in the Roadmap is the durability test of one or more full scale components employing loads, moisture, and temperature applied in real-flight-time. Such a test will take years to accomplish, but it will serve as a reference against which other, simpler tests can be measured. It is intended that articles identical to those receiving the long real-flight-time test will be subjected to accelerated tests. The goal is a validated, economical, accelerated durability test procedure. The body of research in the Structural Integrity Roadmap will provide other benefits in addition to new methods for qualification. Inherent in the process is an evaluation of past and present qualification methods. Equally important is development of the procedures for force management of the components already in service. As the research elicits more information about durability mechanisms in composites, future use of composites will be encouraged by removing apprehension about long term durability. A final bonus is a fallout of data for improved designs, component development strategies, and verification procedures.

VII. CONCLUSION

For composites in service now, confidence is derived primarily from the knowledge of their conservative design. Additional confidence is being gained from in-service inspections and further structural tests. Components currently approaching service have benefited from recent data and have been qualified by programs which include, in various forms, the effects of moisture, temperature and load history on structural safety and durability. In particular, safety is achieved by the slow damage growth concept for damage tolerance; individual article acceptance at present is based on proof test or non-destructive testing. Finally, for composites in the future, a Roadmap for research in structural integrity has been established and the initial programs are underway. The principal goals of this research are validated accelerated test procedures to reduce qualification costs and life prediction analysis for individual aircraft tracking and force management.

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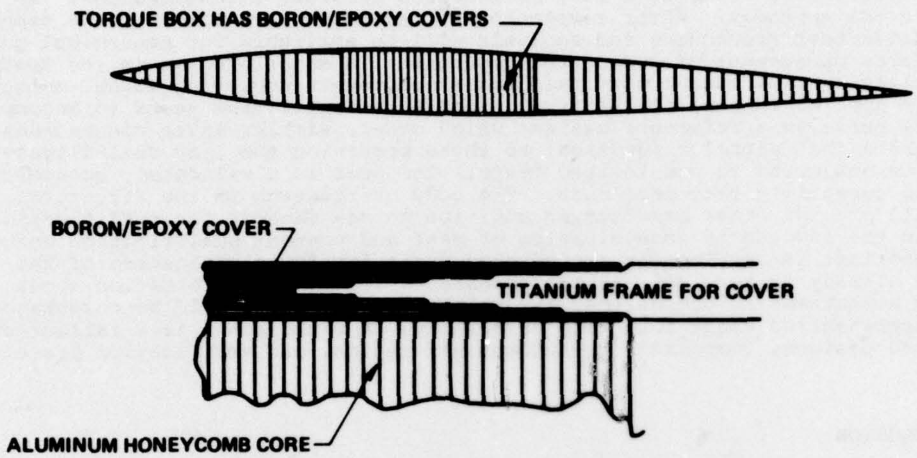

Figure 1. Composites in Service: Fighter Empennage

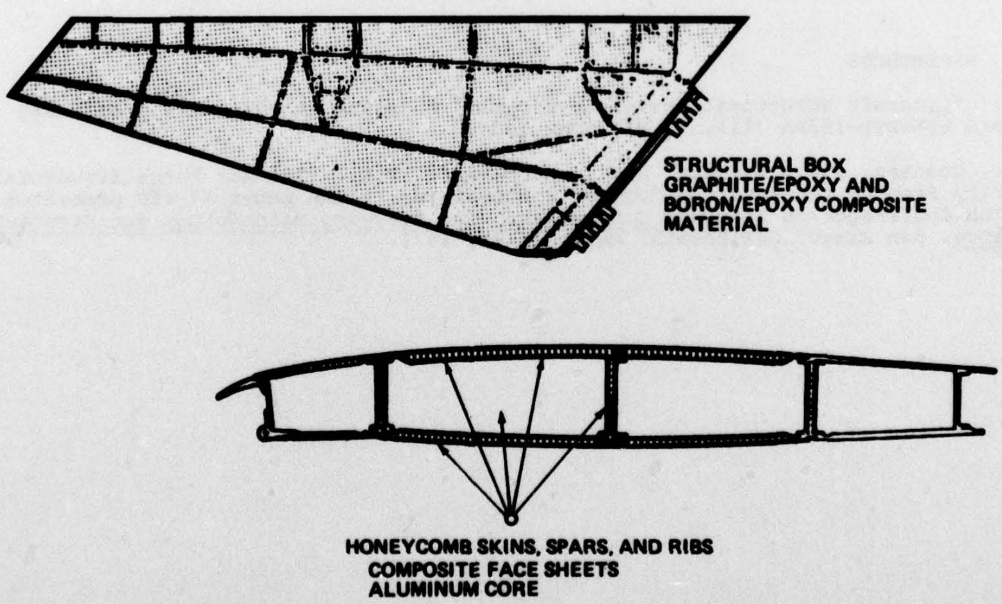

Figure 2. Composite Approaching Service: Outer Wing Panel

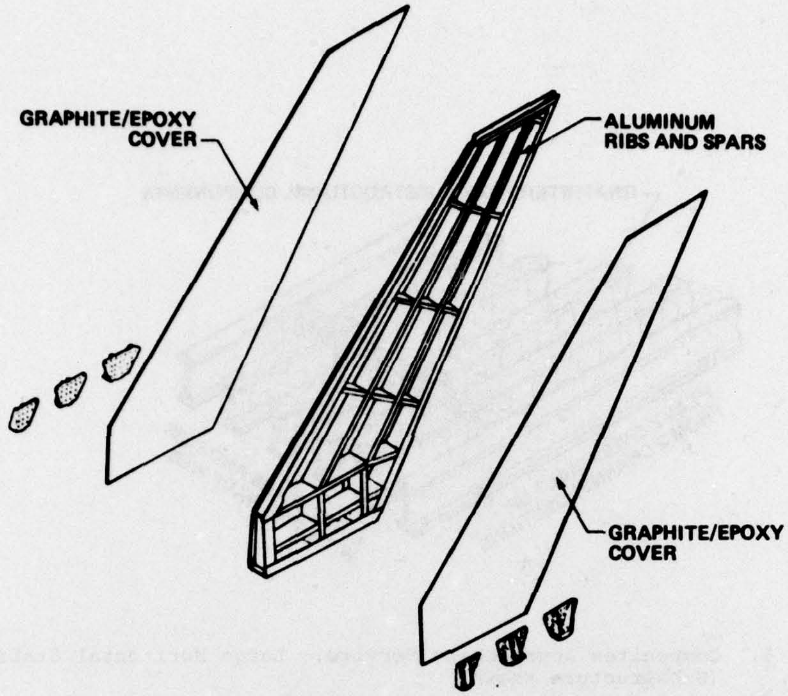

Figure 3. Composites Approaching Service: Small Vertical Stabilizer

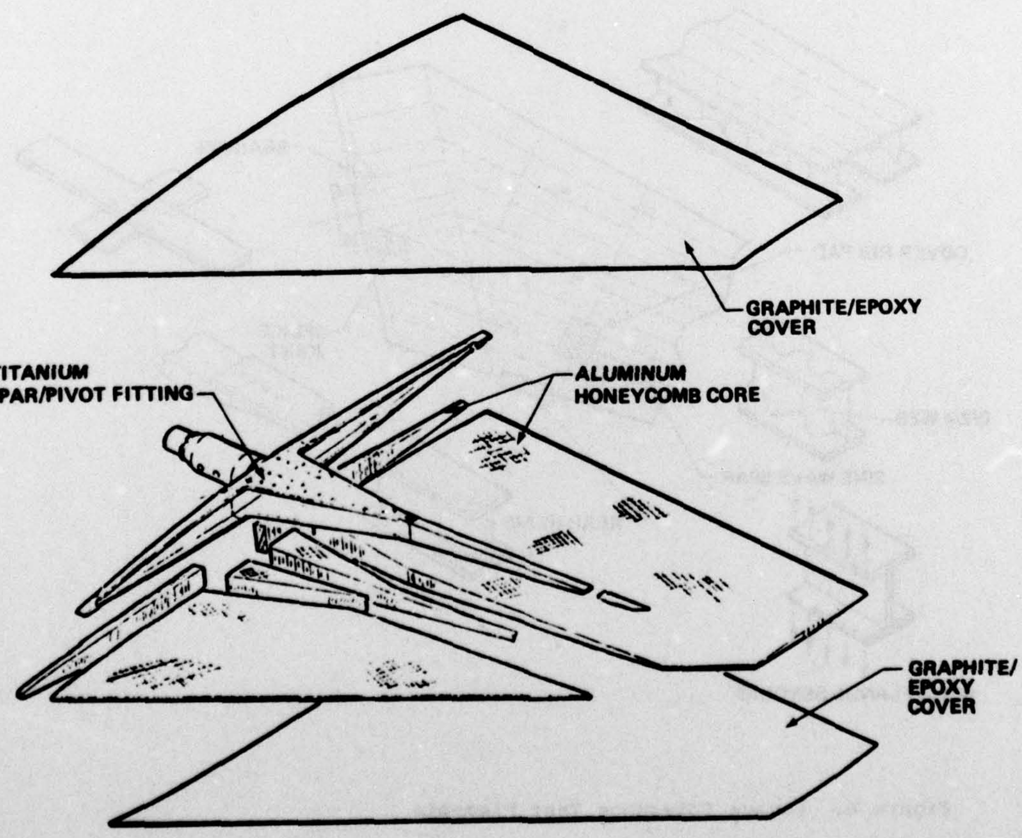

Figure 4. Composites Approaching Service: Small Horizontal Stabilizer

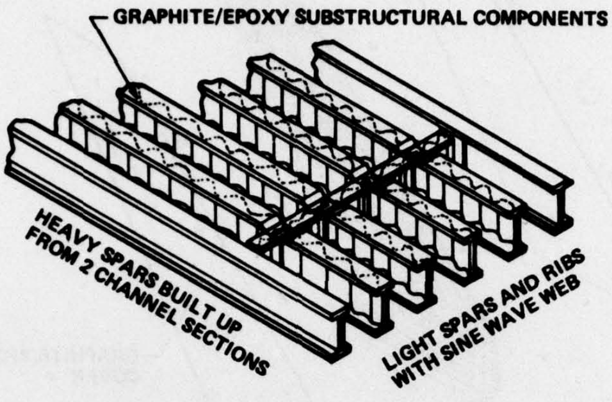

Figure 5. Composites Approaching Service: Large Horizontal Stabilizer (Substructure Shown)

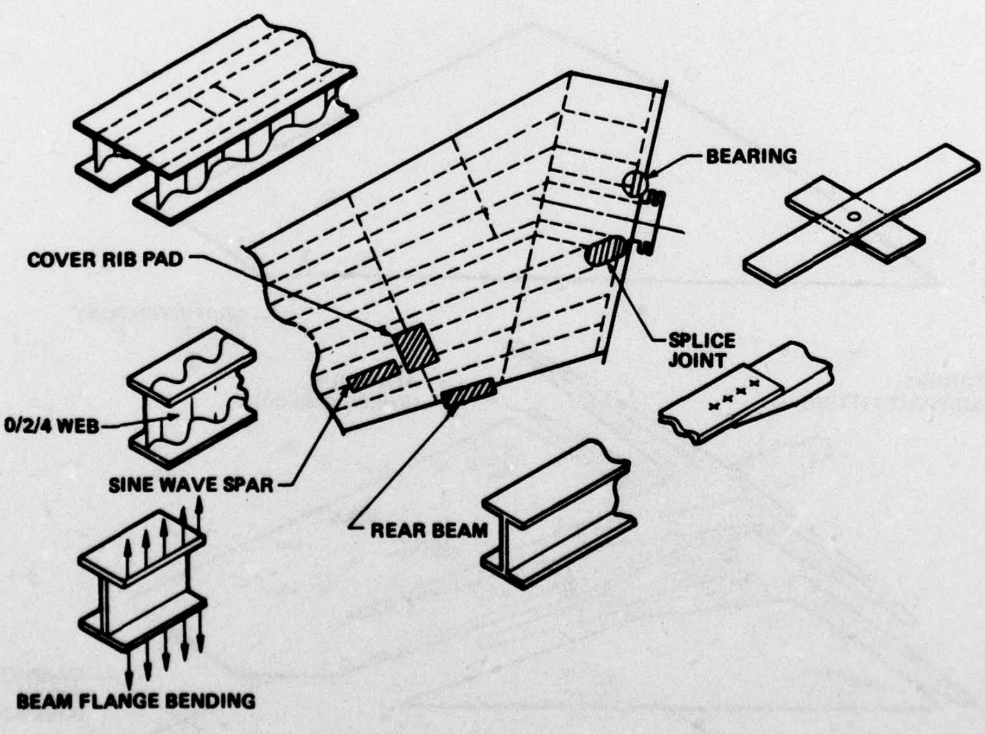

Figure 6. Damage Tolerance Test Elements

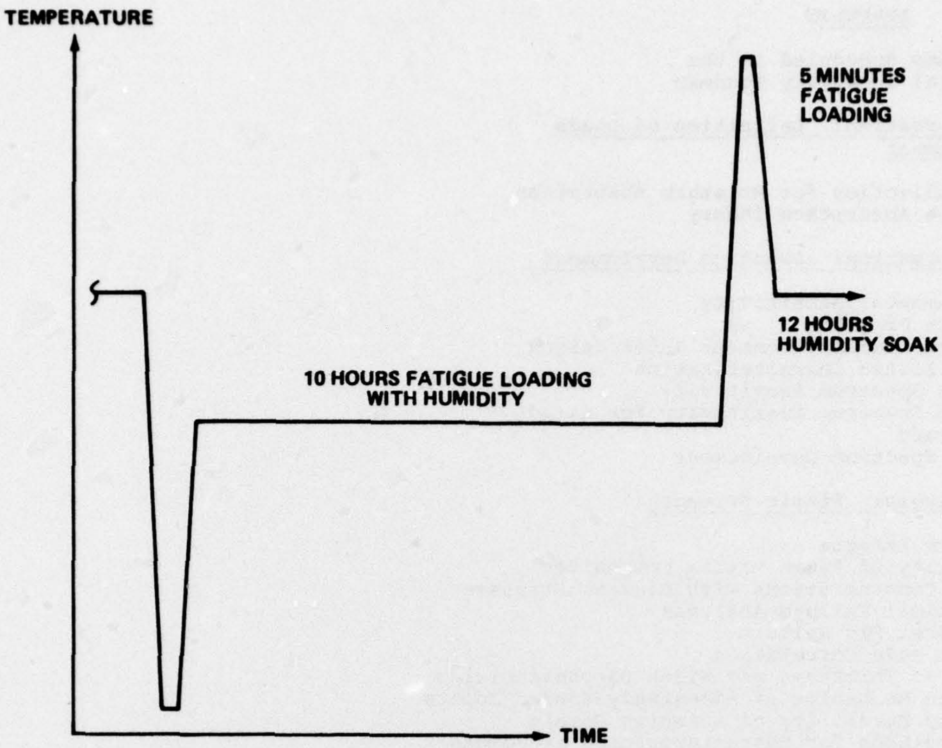

Figure 7. Environmental Test Spectrum

TABLE 1: Assessment of Capability for Qualification of Metallic and Composite Aircraft Structure

	Metals	Composites
Static Strength Analysis	YES	YES
Static Strength Test	YES	YES
Damage Tolerance Analysis	YES	NO
Damage Tolerance Test	YES	LIMITED
Durability Analysis	YES	NO
Durability Test	YES	NO

TABLE 2: Assessment of Capability for Management of a Force of Operational Aircraft Having Metallic and Composite Primary Structure

	Metals	Composites
Repair Procedures	YES	LIMITED
Nondestructive inspection:		
Equipment and methods:	LIMITED	LIMITED
Accept/Reject criteria:	YES	NO
Individual Aircraft Tracking	YES	NO*

*Because analysis method is not available

APPENDIX

Programs Scheduled in the Structural Integrity Roadmap

Design Information: Definition of Loads and Environment

Data Collection for Moisture Absorption
Moisture Absorption Theory

Design Information: Spectrum Development

Environmental Sensitivity
Hardware Programs
Spectrum Load/Environment Interactions
Probabilistic Characterization
Fatigue Spectrum Sensitivity
Fatigue Spectrum Sensitivity for Large Aircraft
Design Spectrum Development

Design Analysis: Static Strength

Moisture Effects
Durability of Resin Matrix Composites
Stress Concentrations with Biaxial Stresses
Statistical Failure Analysis
Post First Ply Failure
Failure Mode Correlation
Effect of Thickness and Width Discontinuities
Fracture Mechanics of Adhesively Bonded Joints
Improved Durability of Adhesive Joints
Tests Methods for Characterization of Joints
Improved Fatigue Behavior of Adhesive Joints
Mechanical Joint Failure Criteria
Effect of Manufacturing and Design Tolerances in Joints
Development of Special Fasteners
Structural Criteria
Effects of Surface Notches
Advanced Composites Serviceability Program
Buckling Sensitivity of Optimized Structures
Buckling Tests of Stiffened Panels
Analysis Methods for Unequal Properties
Improved Methods for Automated Sizing
Sizing for Strongly Mixed Strength and Stiffness
Heat Transfer and Thermal Stress Analysis
Aircraft Structural Optimization Program
Large Deflection Analysis
Stiffened Panel Design Optimization
Force Method Advanced Optimization

Design Analysis: Damage Tolerance: Slow Damage Growth

Time Dependent Environmental Behavior
Long Term Loading on Adhesive Joints
Sustained Load Temperature-Moisture Effects
Effects of Compressive Fatigue Loading
Biaxial Fatigue
Defect/Property Relationship
Life Assurance Testing
Spectrum Load-Environment Interaction
Effect of Design Variables on Life of Mechanical Joints
Service/Maintainability
Cumulative Damage Modeling
Advanced Methods for Prediction of Strength Degradation Rates

Design Analysis: Damage Tolerance: Failsafe

Battle Damage Tolerance
Structural Concept Evaluation
Crack Arrestment Concepts
Effect of Unrepaired Damage on Service Life

Design Analysis: Durability

Component Evaluation
Fighter Empennage
Service Evaluation: Outer Wing Panel
Field Inspection
Procedures for Field Inspection
Protection Systems
Sonic Fatigue Design Charts
Dynamic Scaling
Time Compression
Long Term Acoustic Durability Tests In Service
Advanced Composites Design Guide and Repair Handbook

Full Scale Testing

Real Time Small Horizontal
Accelerated Small Horizontal
Production Small Vertical Stabilizer
Production Small Horizontal Stabilizer
Large Vertical Dry Durability Test
Large Horizontal Dry Durability Test

Individual Article Acceptance

Nondestructive Inspection for Holes and Edges
NDI Methods Improvement
Correlation of NDI with Defect Tolerance Tests
Proof Test Concept
Large Vertical Stabilizer Proof Test
Certification Procedures by Proof Test
Evaluation of Chemical Composition of Epoxy Resins
Quality Assurance of Chemical Composition
Monitoring of Cure
Improved Cure Process and Control
Composite Material Properties
Epoxy Resin Control for Environmental Resistance

Force Management

Composite Repair
Large Area Repair
Facility Requirements for Repair
Individual Airplane Tracking

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14. Abstract

Certification is the procedure which provides the possibility of making certain that any aircraft, whether civil or military, has an acceptable safety level for a given future utilization. For metallic structures, it is expressed as a set of rules which, from experience, it has been possible to transform into numerical specifications.

Composite materials, the advent of which is too recent for calculated data yet to be generalised, do however offer a number of very specific characteristics which are quite often rather imperfectly understood.

Consequently, the four papers which are included in this Report do not claim to provide certification principles in the form of regulations. For the time being, they describe various conservative approaches which, through experimental programmes which are often most impressive, have retained two simultaneous major objectives:

1. firstly, to fly a certain number of high performance aircraft structures, without major risk.
2. to accumulate experimental and technical data which will help to define general safety factors, defect tolerances, propagation criteria and critical thresholds so as to have available a set of rules limiting and describing accurately the necessary physical checks and testing.

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