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Executive summary

Fatigue and Damage Tolerance Evaluation of Structures: The Composite Materials Response

22nd Plantema Memorial Lecture

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This report contains the second International Committee on Aeronautical Fatigue (ICAF) Plantema Memorial Lecture dedicated to composites, more than three decades after the first one given by Herbert Hardrath in 1977.

Description
Herbert Hardrath's Plantema Memorial Lecture preceded the mid-1980s introduction of composites in commercial aircraft primary structures. Since then much has been learnt about composite primary structures, their advantages and limitations. Composites have not totally superseded metallic structures, even in the latest projects, the Boeing 787 and Airbus A350 XWB. In this 22nd Plantema Memorial Lecture, presented at the 25th ICAF Symposium, emphasis is laid on the fatigue and damage tolerance aspects of composite structures in the context of the certification methods that have been specially developed for them.

This report is based on a paper that has been presented as the 22nd Plantema Memorial Lecture on the first day of the 25th symposium of the International Committee on Aeronautical Fatigue, on 27 May 2009 in Rotterdam, The Netherlands.

Fatigue and damage tolerance evaluation of structures - The composite materials response

22nd Plantema Memorial Lecture

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FATIGUE AND DAMAGE TOLERANCE EVALUATION OF STRUCTURES

THE COMPOSITE MATERIALS RESPONSE

22nd Plantema Memorial Lecture

Jean Rouchon*

INTRODUCTION

The International committee on Aeronautical Fatigue - ICAF - was founded in 1951 on the initiative of Dr. Frederick J. Plantema, in response to growing concern regarding fatigue problems in aircraft structures. At that time, a hundred percent of the commercial and military aircraft structures were made out of metal and Dr. Plantema was deeply convinced that sharing practical experience and academia research on a much broader range was essential to overcome fatigue of structures. Two decades later, while very rapid progresses were being observed in fracture mechanics and spectrum fatigue, unforeseen help came with the occurrence of a new generation of materials, reputed to be insensitive to fatigue and corrosion, the composites. In 1977, Herbert Hardrath, from NASA, gave a Plantema memorial lecture [1] entitled ‘Advanced composites-the structures of the future’ and this turned out to be eight years before the first serial application of such materials on a commercial aircraft primary part. A comprehensive review of all the exploratory works in progress in the seventies to prepare a reliable and valuable introduction of composites in aircraft structures can be found in his lecture. The present paper is the second Plantema memorial lecture dedicated to composites and it comes more than three decades after the Herbert Hardrath one. We know today to what extent, and at what rate, composites have been introduced in primary structures and where the limitations are. These materials have not totally superseded metals even in the very ambitious projects which are the Boeing 787 and the Airbus A350 XWB. The need for research and expertise in metal fatigue has not decreased as most of the parts remaining metallic are often critical ‘structural knots’. These may be sized with lower margins to benefit from a more advanced modeling and contribute as well to more and more stringent weight saving objectives. In addition to that, different certification methods tailored to composite attributes had to be developed to achieve reliable and safe structures. These methods are briefly outlined in this paper and, in the context of this 25th ICAF symposium, emphasis will be laid on fatigue and damage tolerance aspects. Challenges created by an increasing composite ratio are also pointed out and ways for new research and developments are proposed.

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A BRIEF SURVEY OF THE INTRODUCTION OF COMPOSITE MATERIALS IN AIRCRAFT STRUCTURES

Because the very first airplanes – built by the Wright brothers, in 1903 - were made out of wood, a natural composite, and fabric, it is easy to state that composite materials are as old as aviation. It is correct, but does not bring any value to the purpose of this paper. In the context of the today aviation construction, composite materials are those heterogeneous materials made from low density, high stiffness and high strength fibers embedded in a matrix system, which is most often a thermoset resin. Sticking to this definition, the very first serial application of composite materials on an aircraft is the horizontal tail-plane of the F14 Tomcat (first flight in 1970) just followed by the F15 Eagle (first flight in 1972), with the same part made out of composite plus the fin box. In both cases, boron fibers reinforced plastics were used. However, the very high cost of this material together with potential health and safety problems with the handling of such very stiff fibers (due to their diameter ranging from 0.1 to 0.2 millimeters) were very compromising for their future. Then, as soon as carbon fibers could be produced with a proven and constant quality, boron fibers disappeared in new projects. They were replaced by these new fibers with much better cost reduction perspectives and no health and safety related concern. From a very low percentage of the airframe empty weight in these early 70's projects, the composite ratio climbed to over 5% with the F16 fighter and 10% with the F18 (first flight in 1978). Today, in the most advanced fighter project, the Joint Strike Fighter F35 Lightning II, the composite ratio ranges between 35 and 40%. A comparable figure can be found with the A400M airlifter. A composite ratio between 25 and 30% of the airframe can be found in less recent projects (years 90's) such as the Rafale, the Eurofighter and the F22 Raptor. Around 20% of the JAS39 Gripen structure is composite.

Serial composite applications in commercial aircraft came around 10 years later. Focusing on primary structures of a significant size, the Vertical tail-plane of the A310/300, an aircraft certified in 1985, can be considered as being the first one ever. A description of this composite part with a brief survey of the certification work carried out at that time can be found in reference [2]. With all the control surfaces and the whole tail-plane made out of composite, the A320, certified in 1988, turned out to be a significant breakthrough in the introduction of such materials in aircraft structures. With this programme, a 10% ratio of the structure empty weight was exceeded for the first time. With an application now to the wings and the fuselage, the Boeing 787, followed by the Airbus A350 XWB, represents the second major step of this evolution, and likely the last one, in term of overall ratio. In both programmes a proportion of composite approaching 50% is now achieved. Figure 1 illustrates how this composite ratio evolved over the time for commercial aircraft.

All first serial applications have been preceded by exploratory programmes with, as far as possible, composite parts really installed on regular service aircraft over a limited period of time for evaluation purposes. In its ICAF'77 Plantema memorial lecture [1], Herbert Hardrath gave a comprehensive review of all the NASA supported programmes which had taken place in the seventies. The most prominent of them was launched in the frame of the so-called ACEE (Aircraft Energy Efficiency) initiative following the first oil shock in 1971, with the aim to substantially reduce fuel consumption in air transportation. The ACEE programme targeted three domains: improvement of the aerodynamics, engine efficiency and structure weight saving. For the latter, composite materials were regarded as capable to potentially achieve a weight reduction of around 20% because of their very high specific strength and modulus. However, industrial and certification problems were still to be explored. For this purpose, decision was made to design, certify and deploy into service three significant composite applications, one for each commercial aircraft manufacturer existing in those days in the United States: A 737 horizontal tail-plane at Boeing, a DC-10 Fin at Mc Donnell Douglas and a L-1011 fin at Lookheed. Starting in the late 70's, this program went up to its complete achievement and all the parts put into service have now been removed after a sufficient period of time for a technical evaluation of service effects. Among the lessons learned, it is noteworthy that no fatigue or corrosion could be found, but just accidental impacts and some lightning strike damages.

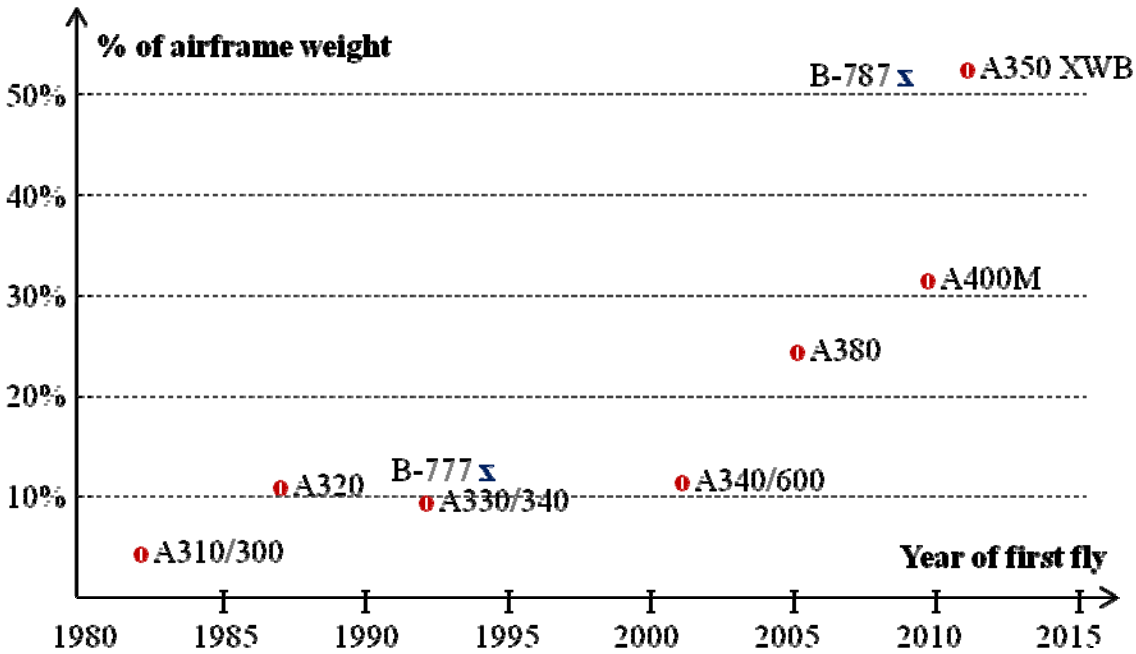

Figure 1: Evolution of the composite ratio on commercial aircraft

The reason why this ACEE program is put forward in this paper is that it served as the main support or argument to the development of the first certification document prepared by the airworthiness authorities to address composites. This is the well-known Advisory Circular AC 20-107A, whose first version was published in April 1978.

HOW COMPOSITE MATERIALS ARE ADDRESSED IN CIVIL AIRCRAFT CERTIFICATION

People involved in civil certification are familiar with the section 16 entitled 'Special Conditions' of Part 21 of the Codes of Regulations. A Special Condition (SC) is defined as a special detailed technical specification prescribed for a product if the related airworthiness code does not contain adequate or appropriate safety standards for the product because:

- *There are novel or unusual design features, relative to the design practices on which the applicable standard is based,*
- *The intended use of the product is unconventional,*

- *Experience from other similar products in service has shown that unsafe conditions may develop.*

Practically, a special condition deletes or modifies an existing regulatory paragraph, or adds a new specification. When it is shown that a special condition does not apply to one product, only, but appears to be systematically extended to a broad range of products, it is transformed into a new rule in the Codes of Regulations.

Because the applicable standards have been originally built up from the experience gained with metal structures, the introduction of composites should fall in the first category covering novel or unusual design features. However, in the past, very few special conditions have been raised for composite structures on the basis that certification could be adequately addressed through tailored means of compliance as we can find in the Advisory Circulars. The situation is now moving since special conditions covering fuselage in-flight fire flammability resistance, crashworthiness, and tire debris penetration of fuel tank structures have been publically released in 2007 for the Boeing 787. Nevertheless, most of these special conditions are outside the purpose this paper

In Part 25, Damage Tolerance and Fatigue Evaluation of Structures is covered by only one paragraph, § 25.571, whatever the material. There are numerous references, many of them having been presented in ICAF symposia, explaining how § 25.571 has been evolving over the last 40 years (Car 4b 270 became § 25.571, actually in 1966) on the basis of the lessons learned and progress made in academic knowledge of metallic structure behaviour, mainly in fracture mechanics. It is important to notice that all updates have taken place without any input coming from the composite community and without any need for a composite dedicated special condition that could have become a specific regulatory paragraph later on. This situation has already been addressed by the author in the paper [3] he gave at the 21st ICAF Symposium in Naples in 2007.

Unlike transport category aircraft (Part 25) the need for such a dedicated Fatigue and Damage Tolerance rule was identified for small aircraft category (Part 23) and a new paragraph 23.573 incorporated in September 1993, by amendment 23-45. As far as rotorcraft are concerned, new composite-dedicated regulatory paragraphs, i.e. § 27.573 and § 29.573, are expected soon following an ARAC (Aviation Rulemaking Advisory Committee) preparation work which took place in the years 2000-2002.

When § 25.571 states the following:

a) *General. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, manufacturing defects, or accidental damage, will be avoided throughout the operational life of the airplane,*

It is clearly a performance based requirement even if the rest of the regulatory paragraph is more prescriptive. Neither more nor less is expected from a damage tolerant composite structure, except that, no degradation of the existing level of safety already achieved with metals can be admitted. This is what we call the 'benchmark' principle. Special Conditions can require a comparative assessment between the composite and the appropriate proven original metal solution, as it is mainly the intent of the SC above mentioned for the Boeing 787. If not, this must be the intent of the Advisory Circulars to propose acceptable means of compliance.

It was in July 1978 that the first Advisory Circular covering composite structures was published by FAA, under the reference AC 20-107 and, as said before, the main reason was to have an Authoritative document to support the certification of the applications launched in the ACEE programme framework. A general presentation of the first version of that Advisory Circular, placed the context of that period, can be found in reference [4]. However, it is in the early 80's that an intense activity took place which ended up with a new version in April 1984, the AC 20-107A (then referenced as ACJ 25-603 in Europe). The official reason to update this Advisory Circular was essentially to match the amendment 45, December 1978, introducing Damage Tolerance. Moreover, for the first time an Advisory Circular was prepared by a joint team both from Europe (Joint Aviation Authorities) and the United States. References [5], [6] and [7] give details about the purpose of this revision and its outcome. Nevertheless, if we look into the details of the first version of the Advisory Circular, it appears that Damage Tolerance evaluation was already mentioned in the text. It was ahead of the forthcoming regulatory paragraph, with demonstrations nearly 'copied and pasted' from the methods under development with metals, i.e. the 'slow growth' concept, as can be read below in this first composite AC:

a. Damage Tolerance (Fail-Safe) Evaluation.

....

> *(1) ....The tests should demonstrate that the residual strength of the structure can withstand the limit loads (considered as ultimate) with a damage extent consistent with initial detectability and subsequent growth under repeated loads including the effects of temperature and humidity. Growth rate data should be used in establishing a recommended inspection program.*

In fact, the revision recognised that the slow growth principle would not be practicable with composites and that a <u>no-growth concept</u> would be therefore the foundation of the Damage Tolerance evaluation. In the revision A in 1984, the text then became:

a. Damage Tolerance (Fail-Safe) Evaluation.

> *(1) Structural details, elements, and subcomponents of critical structural areas should be tested under repeated loads to define the sensitivity of the structure to damage growth. This testing can form the basis for validating a no-growth approach to the damage tolerance requirements.*

Of equal importance, the revision introduced the specific concern of low velocity accidental impact damage with composites; a point completely overlooked in the first version. More details about composite attributes with respect to certification and the first implementation of this AC in Europe can be found in reference [8] of the author.

Being 25 years of age, this Advisory Circular AC 20-107A really needed to be updated to be more reflective of lessons learned and today's practices. A proposal for a revised version has already been prepared by Airworthiness Authorities specialists and should be submitted for public comments this year 2009. This project is outlined in reference [9].

WHAT FATIGUE FAILURE MEANS WITH COMPOSITES

Fatigue failures in metal structures are a well-understood phenomenon documented by a vast amount of publications. No less than 554 references are listed in the Walter Schütz paper [10] entitled 'A History of Fatigue', and there are likely many more papers in the libraries. The remarkable book 'Fatigue of Structures and Materials' [11] by Jaap Schijve is now a reference that nobody should ignore in this domain. A comprehensive and step by step presentation of the physical phenomena involved in a fatigue failure and to what extent each of them is modeled can be found in this book. How fatigue analysis has progressed over the past 50 years, along with in-service facts (and ICAF history), has been purpose of the Plantema memorial lecture presented by Anders Blom at the ICAF 2001 symposium in Toulouse [12].

One must recognize that fewer references addressing fatigue of composite materials and structures are available now and a vast majority of them are more providing material data than a rational explanation of the physical phenomena which are involved.

Whatever the material, either metal or composite, there are three phases in the fatigue life of a structure: crack initiation, crack growth and final failure. The very early stage of crack initiation – cyclic plastic deformation at the micro scale, moving dislocations and then crack nucleation – is very well understood with the crystalline micro structure of metals. To what does it correspond at the same micro scale of either the matrix itself or the fiber-matrix interface? It is still a mystery. Much more described is the second phase, fatigue crack growth, and figure 2 shows how fatigue damages in a composite laminate may develop from a stress concentration area (open hole).

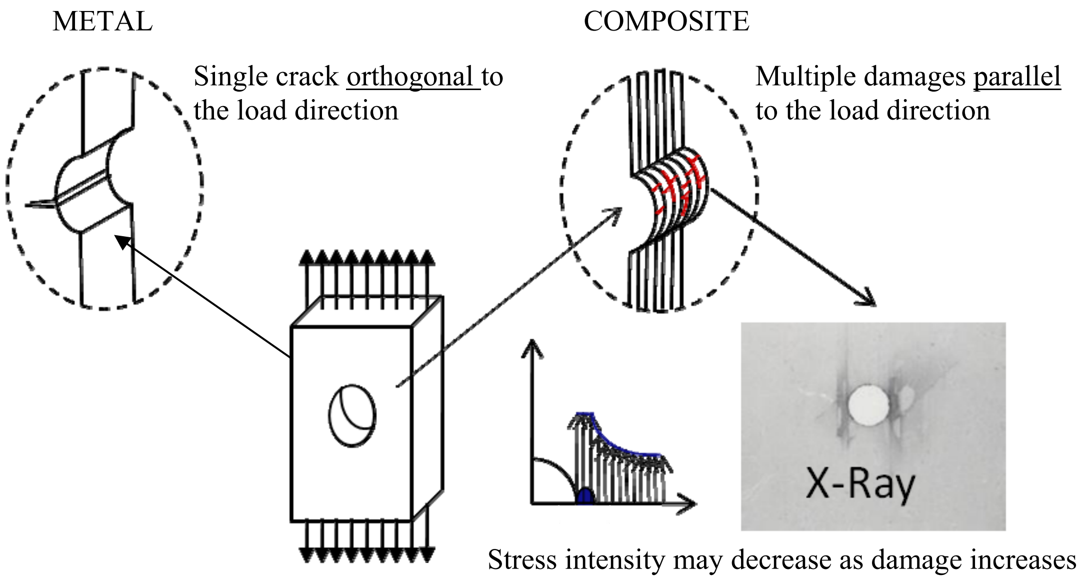

Figure 2: Fatigue initiation and damage growth around a hole

In contrast with metals where single cracking transverse to the maximum tensile force would have developed, here, in the composite, there is a very complex phenomenon involving numerous translaminar and interlaminar cracks plus likely interface cracks and fiber failures at a lower scale level. At the end, fatigue damages with composites are mainly delaminations which are, by nature, oriented parallel to the axial forces.

After crack initiation and growth the last phase of a fatigue failure will be totally different between a metal and a composite structure. For the metal one, it will be a static failure, in a brittle or tearing mode, under a load of the magnitude induced by the load spectrum. For a composite it will be an unacceptable loss of stiffness. For an element loaded in compression this may lead to a static failure by buckling. In other cases (rotorcraft rotating elements for instance), it may correspond to a loss of functionality.

This physical observation of fatigue damages in a composite shows how much difficult analytical modeling would be, if deemed necessary. Fortunately, from an engineering point of view, such design features (open or filled holes) are not fatigue critical and do not need to be specifically addressed for this purpose. The reason is that the very high sensitivity of composites to stress raisers in static put the operational load sufficiently low to avoid fatigue issues. This has long been illustrated by the graph figure 3 presented by Whitehead [13], and it is still valid with current composite materials and construction.

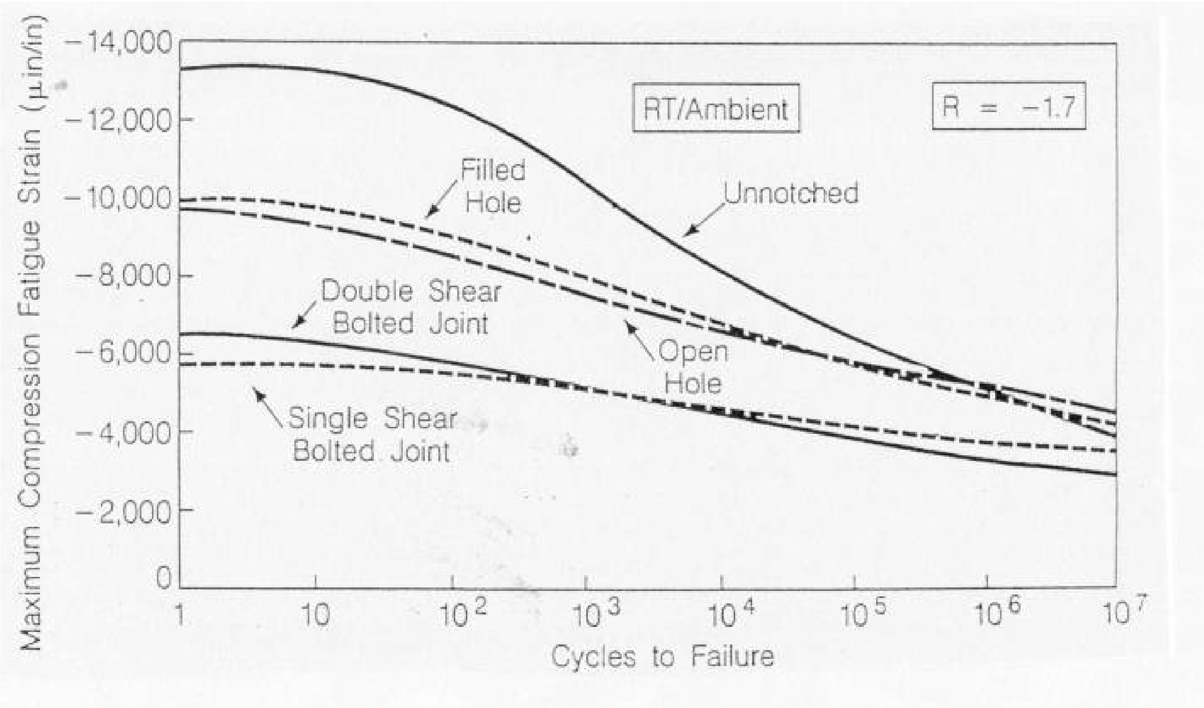

Figure 3: General fatigue behaviour of composites (Whitehead, ICAF 1987)

While fatigue damages should not be expected in service from composite laminates loaded in membrane, which means in the absence of out-of-plane stresses, the situation is not the same wherever the local design may develop 3D stresses along the weak axis of the laminate. Today, very few fatigue findings have been recorded with Part 25 composite structures either in test or in service. Figure 4 illustrates one of them which occurred in the full-scale fatigue test of the A320 fin in 1987. The damage consisted in a delamination (one could say a disbanding, too) between the stringer array and the main skin initiated by peeling stresses while the load introduction system of the test rig might have contributed to the development of this damage, too. Details about this fatigue finding with the corrective actions implemented both in the structure test and serial production can be found in [14]. Though quite isolated, this observation of fatigue damage with a composite structure has contributed to maintain the pressure for a long time on the need for fatigue demonstration in the forthcoming certification exercises.

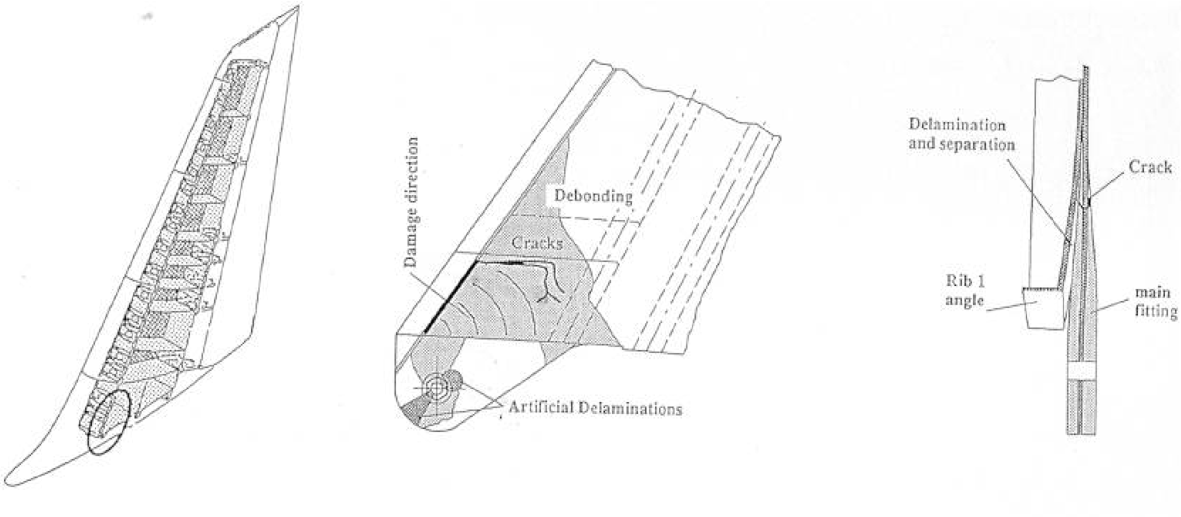

Figure 4: Fatigue failure by disbonding in the test of a vertical fin

Delamination, as being the fatigue failure mode for composites, is well recognized today and when some research programmes are carried out in this domain, they are oriented accordingly. For delamination studies, two kinds of specimens are frequently used, they are the DCB (Double Cantilever Beam) and ENF (Edge Notch Flexure) tests, as illustrated by figure 5.

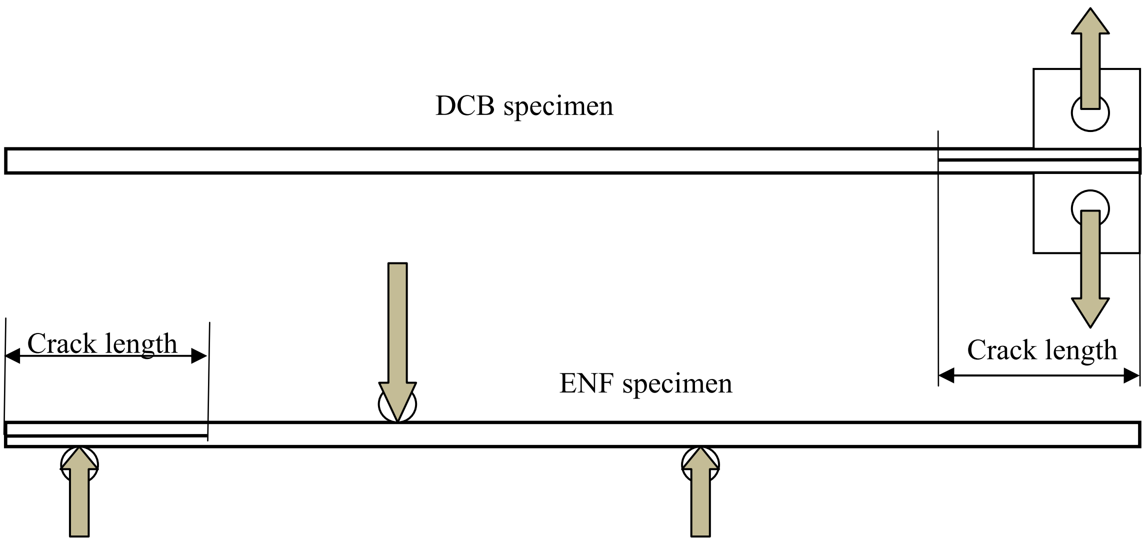

Figure 5: DCB and ENF specimens for delamination studies (static and fatigue)

Detailed modeling and experimental data from these specimens can be found in [15] and [16]. This can serve as a support for predictive analysis in composite fatigue, based on the strain release energy rates in a combination of modes I and II (G1c and G2c). However, even if dedicated analytical models for delamination growth are constantly improving, there is another issue with composites contrasting with metals and the application of fracture mechanics.

With metals, the material properties needed to calculate a fatigue crack growth rate (using for instance the Paris or a Forman law) are not expected to depart from a narrow scatter band and reliable calculations can be performed. On the other hand, when a composite structure has started delaminating, environmental factors (ageing, temperature) may have modified the intrinsic mechanical properties to such an 'unknown' extent that a reliable calculation is no longer possible. Of a much higher magnitude, this remark is relevant to structural bonding where an unexpected manufacturing deviation may have affected the bonding line quality to a non measurable value.

The situation today is that a fatigue analysis similar to what is currently carried out for metal structures – involving crack initiation, crack growth and residual strength - does not exist with composites. Fatigue of composite structures is then addressed first by design precautions to avoid the local development of out-of-plane stresses. Corners, ply drop-off, stringer runnouts are of prime importance. Then, stressing is no more than checking that the maximum strain does not exceed proven values on similar designs. At the end, these good design and stressing practices may be demonstrated by a full-scale fatigue test pursued to a sufficient number of cycles to achieve the expected level of confidence and this is there that the test factor issue arises.

SCATTER IN FATIGUE AND THE TEST FACTOR ISSUE

When composite structures are tested in fatigue for certification purpose, a factor on loads, or on both loads and life, is the common practice to cover the scatter. For Part 25 structures, the most widely used factors are 1.17 on the loads, together with one life (Nf=1), or 1.15 on the loads, together with Nf=1.5.

These figures come from a US Navy study [17] published in 1986 which is now very often referenced as the 'Whitehead' method. There are several reasons to explain the large success of this method with its more than 20 years of satisfactory usage. First, the figures looked conservative enough from an airworthiness point of view, while being easily sustainable by so much fatigue resistant composite materials. Second, the method was supported by such a comprehensive data base for composite scatter assessment that it was uneasy to challenge the outcome. Last but not least, more than a hundred million flight hours accumulated by primary composite structures on commercial airplanes, with no or very few fatigue findings, have shown that this issue has been so far properly addressed.

In fact, nobody complained before the introduction of large hybrid structures (metal-composite) when it turned out that the application of the composite methodology to cover fatigue in the full-scale test would compromise the demonstrations required for the metallic parts and reciprocally. As far as composite materials are applied on control surfaces or empennage torque boxes, it is a common practice to substantiate these parts through tests segregated from the whole airframe. Introducing now composites on large wings and fuselages with an optimized material selection - the right material at the right place - makes this solution no longer technically practicable or economically reasonable for the resulting hybrid structures. New certification methodologies and practices are therefore to be developed to cope with this new challenge. This explains the current efforts to question, or at least try to revisit the existing methodology, with the aim to come up with a unified method able to satisfactorily cover both metals and composites in the fatigue test of one single article.

In the reference [3] presented at the 24th ICAF symposium in Naples, 2007, the author started to address this issue, pointing out the sensitivity of the test factors to the various assumptions on the composite static and fatigue property scatter. In that paper, he could show that just a slight shift of the assumptions on the scatter of fatigue properties was necessary to come up with test factors that a metallic structure could accommodate. This paper will go further in the critical analysis of the so-called 'Whitehead method' with a brief comparison with other approaches e.g. for metals and rotorcraft elements.

The general principle of the 'whitehead method' is that the test factor must cover the **typical gap** between the mean S-N curve (50% survivability) and the B-Basis curve (90% survivability), as illustrated by figure 6.

Assuming two-parameter Weibull density functions for both static and fatigue properties, the following formula allows the calculation of a factor on life N_F (which means applied on the number of cycles to be covered) as it is usual practice with fixed wing structures made out of metals:

$$N_F = \frac{\Gamma\left(\frac{\alpha_L + 1}{\alpha_L}\right)}{\left[\frac{-\ln(p)}{\chi^2_{1-\gamma}(2n)/2n}\right]^{1/\alpha_L}}$$

Where :

$$\Gamma = \text{Gamma function} = \int_0^\infty e^{-t} t^{(x-1)} dt$$

α_L is representative of the scatter of the composite fatigue properties on life,

n is the number of test articles (in general one for a full-scale),

p is the survival probability : 90% as it is for the 'B' value definition,

γ is the confidence : 0.95.

$\chi^2_{1-\gamma}(2n)$ is the value of the Chi-square function with 2 n degrees of freedom at 1-γ probability.

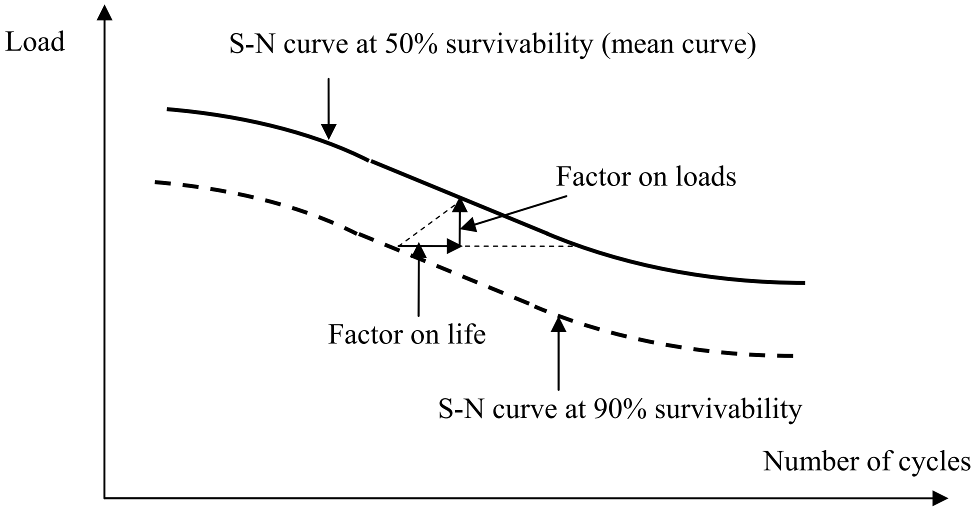

Figure 6: Load/life factor principle to cover a 90% reliability

Introducing the typical variability of composite fatigue properties in this formula, it appeared that the required NF value should be in the range of 13-14, which was quite unrealistic to achieve in a test within a reasonable period of time. For this reason, Whitehead proposed an alternative solution with a Load-Life factors principle covering the same 'B' basis reliability level. Then, the relationship here below allows the calculation of any combination of factors on load and life.

$$LEF = \frac{\Gamma\left(\frac{\alpha_L + 1}{\alpha_L}\right)^{\frac{\alpha_L}{\alpha_R}}}{\left[\frac{-\ln(p).N^{\alpha_L}}{\chi^2_{1-\gamma}(2n)\big/2n}\right]^{1/\alpha_R}}$$

The added parameters are:

α_R , representative of the typical scatter of the static strength properties,

N being the coefficient applied on the life,

Details about the development of this formula can be found in reference [18].

Typical means the most likely to occur. Thousands of test data have been processed in reference [17] and histograms representing the scatter of scatter have been drawn for both the static and fatigue strength properties. The authors suggested to use the modal values of the two populations of scatter (α_R =20 and α_L=1.25) and the result is the well-known factors (1.177 on the loads, together with one life, Nf=1, or 1.15 on the loads together with Nf=1.5). Because of the asymmetry of the population representing the scatter of scatter, as shown figure 7, there was some deliberate conservatism in selecting the modal value instead of the mean one.

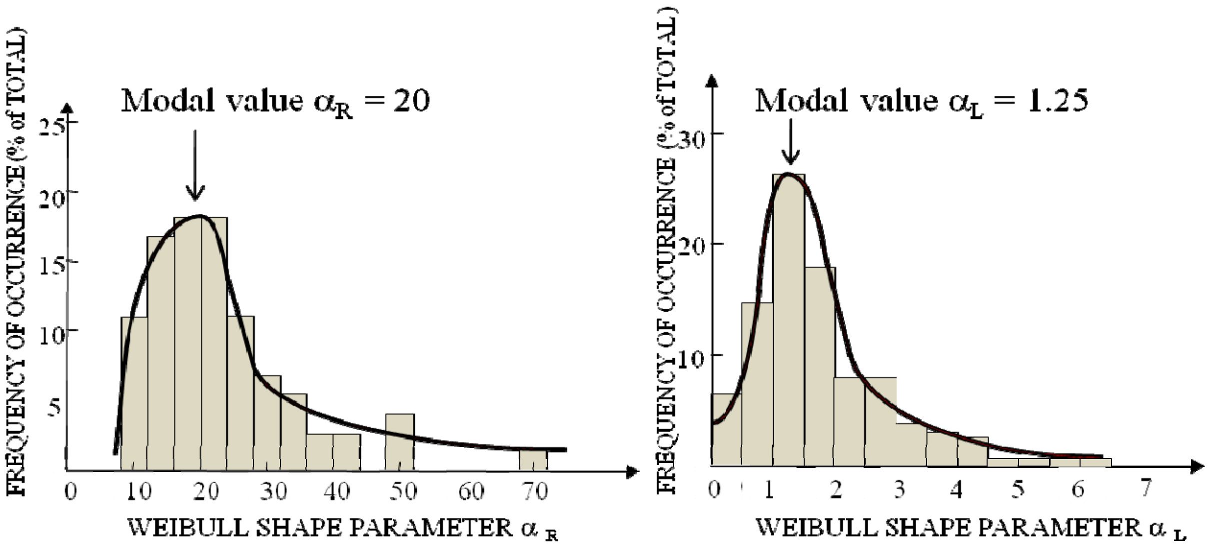

Figure 7: Population of scatter in the 'Whitehead' method

Now, it is interesting to try a first comparison with the methods currently used for metals. When reference [17], again, proposes α_L =1.25 as being representative of the modal value of the population of scatter in fatigue for composites, the authors state that this modal value for metals would be α_L =7.5. Introducing now this figure in the calculation of a factor on life (without any load enhancement factor) the result is Nf=1.47, which is lower than any current practice for the substantiation of metallic structures in fatigue. However, compared to other literature on the subject, this value α_L =7.5 for metals appears to be optimistic and figures ranging from 3 to 4 are more widely shared. Assuming now α_L =4 for an aluminum structure, the 'B' basis reliability calculated with the Whitehead formula could be achieved with a test factor equal to 2.09. Should the assumption α_L =7.5 be true, the first comment arising from the comparative study would be that the current method for metals provides more than covering the typical gap between the mean S-N curve (50% survivability) and the B-Basis curve (90% survivability), as it is the intent of the composite methodology.

Let us go deeper in this comparison with fixed wing structures in metal. Reference [19] is one of the numerous papers providing a comprehensive review of the philosophy developed to determine test factors for fatigue evaluation. Although this paper focuses on the historical factor of 3 1/3 implemented in the U.K. Defence Standards (formerly AvP. 970, now DEF-STAN 00-970) for aircraft of safe-life design, it contains quite useful information about the rationale behind the currently used safety factors.

Even if the first test factors which were used when fatigue emerged as a threat to aircraft structures during the 40's were quite empirical, it turned out that within the following 20 years a correspondence with a targeted risk of failure has been sought. A risk of fatigue failure of about 1/1000, should the fatigue damage be critical, seems to have been rapidly shared among the fatigue community and then postulated. Such risk can be achieved through the reduction of the true mean by around three times the standard deviation (exactly 3.09), assuming a normal or a lognormal population. Unfortunately, we can never know the true mean and the true standard deviation. So, in practice, the estimated mean is assumed to be the true mean and this is balanced by consideration of a conservative value of the standard deviation. The application of the Whitehead formula with a targeted risk of failure of 1/1000°

and a representative (typical) value of the scatter of metal in fatigue equal to 7.5 leads to a factor on life equal to 2.73. The same calculation shows that when the widely used test factors for composites (NF= 1.5 together with LEF=1.15) are applied to a hybrid structure, the composite parts are covered on a 'B' basis reliability, while the metal parts are covered on a 997/1000 reliability level, or 988/1000 with a less controversial value for α_L=4. Such reliability levels are much above the need and the current practice for a redundant part of a fail-safe damage tolerant metallic structure.

At the earliest time of fatigue evaluation pressurized fuselages were supposed to be the sole fatigue critical items and a factor on load (pressure) selected to cover scatter. As soon as the fatigue phenomenon was better understood, and the need for a fatigue evaluation extended to other parts of the structures, a factor on life has been be preferred to prevent the risk of invalidating the test by excessive stresses due to the retardation effects. Today, this is the general practice.

Looking at the current regulations, CS 25.571 with the parent Acceptable Means of Compliance, a factor on life ranging from 3 to 5 is clearly stated for safe-life structures in the European specifications, with no proposal for fail-safe and damage tolerant structures. However, a reduction by a factor of 2 of this safe-life factor is the usual practice, which means between 1.5 and 2.5 in this case. Nevertheless, we are not referring to a test factor any longer, but a scatter factor which means the life reduction factor used in the interpretation of fatigue analysis and test results. There is a test factor of 2 required by FAR 25.571 to be applied on the full-scale fatigue test, but the intent is to cover widespread fatigue damages.

It is noteworthy that the historical intent of the previously discussed test factors or safety coefficients was to cover scatter in the fatigue phenomena only and not in the usage spectrum. It is today a common practice for transport aircraft category to cover the uncertainty in the usage by an additional factor on loads, let us say 10%. This can cover a refinement of the load spectrum after the flight test campaign and/or the coverage of derivatives or stretched versions. However, for unmonitored fighter aircraft, the uncertainty associated to service usage is generally covered by an additional factor on life (3 1/3 x 1.5 in the UK Defence Standard 00-970 [20], from 3 to 5 in the French standard AIR 2004E [21]. But, this is military practice.

While the Airplane Industry could live around forty years before regulatory requirements covering fatigue were introduced, the situation could not be the same with the Helicopter industry where compliance with static strength requirements only would have been totally unable to safely cover a significant time of usage of the dynamic elements. At the start of Helicopter development in the 40's fatigue substantiations according to a safe-life principle were implemented. While a typical representation of the number of cycles supported by an airplane is in the range of 10^5 to 10^6, the number of cycles to be investigated with Helicopter dynamic elements may exceed 10^8. Applying a factor on life for testing purpose on this figure would lead to excessive test durations all the more so because S-N curves are mostly flat in this domain. For this reason, covering the scatter by a safety factor on stress only, was rapidly generalized. Figure 8 shows the illustration of this factor which represents the gap between what is called a mean curve and a working curve. More details about the practices of the Industry in this domain can be found in references [22] and [23].

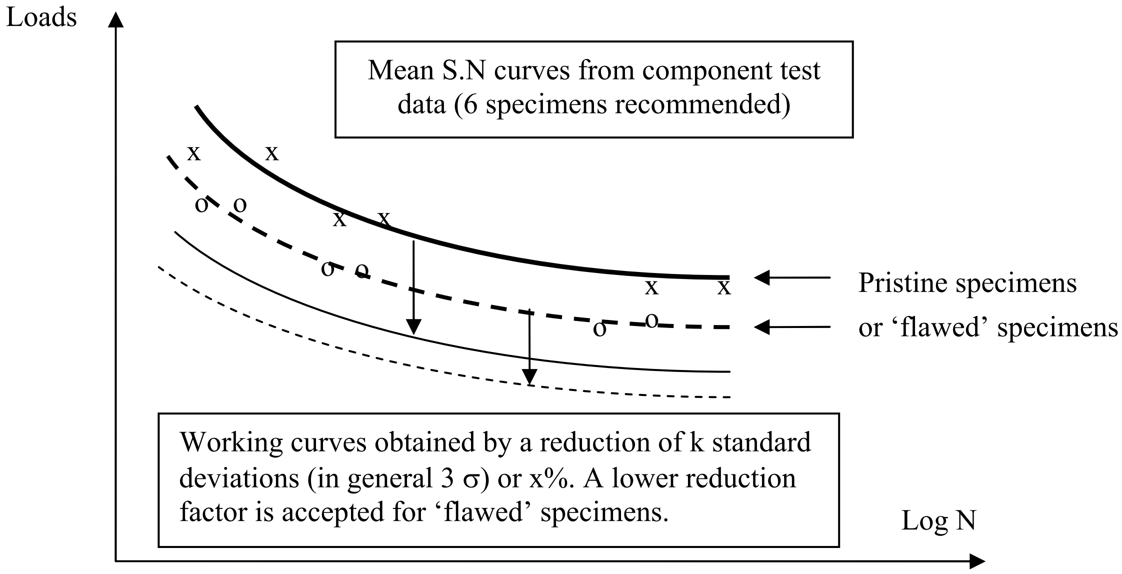

Figure 8: Fatigue substantiation principle for rotorcraft elements

As it was intended by fixed wing structure specialists, the coefficient implemented for the initial safe-life was determined in such a way to guarantee a reliability level here also set at 99.9%. This was the rationale behind the 3 σ value. Let us assume that we know the true mean and the true standard deviation, a (m-3σ) value represents a survival probability 99.865% for a normal population, but here also nobody knows the true mean and the true standard deviation. In the same way as it is done with fixed wing structures, the estimator of the mean is assumed to be the true value of the mean and a conservative value of the standard deviation is considered. When the mean S-N curve is plotted from test results, the working curve is used to establish the retirement life of the component, or the inspection threshold and interval for a damage tolerance evaluation. The parent advisory circulars of parts 27 and 29 certification specifications recommend the test of 6 specimens for the fatigue evaluation (e.g. a representative section of a blade, a rotor hub or a shaft).

Here appears a strong difference between the demonstrations required for Airplanes and Helicopters regarding fatigue. For the latter, structural parts are systematically tested up to the occurrence of a fatigue failure by sufficiently increasing the loads, while for Airplanes, the demonstrations do not necessarily need to go up to the fatigue failure

Now, let us go back to Part 25 composite structures. A small sample size formula has been developed within this community to derive either 'A' or 'B' values, assuming a normal population and the coefficient of variation being known. This formula is as follows:

$$B\ value = \frac{1-(Kb.CF)}{1+\left(conf.\frac{CF}{\sqrt{n}}\right)}.\overline{X}$$

Where:

$\overline{X}$ = Estimator of the mean (test data average value).

CF is the assumed coefficient of variation (ratio σ/m).

Conf is a factor liked to the confidence limit (here 95%) and equal to 1.6449.

Kb is a coefficient linked to 90% survivability and is equal to 1.2816. When an ‘A’ value is sought, Kb must be replaced by Ka, which is equal to 2.3263.

n is the sample size.

As above, the value of the standard deviation (more exactly the coefficient of variation) is assumed to be known, but the estimator of the mean is not considered as being the true mean. Let us assume now a sample size of six specimens as it is in Helicopter practice and a usual coefficient of variation of 9% along the vertical axis. The rotorcraft methodology would say that the working curve is 27% below the mean curve, while the application of the small sample size formula says that the ‘A’ value (99% reliability) is 25.5% below the mean calculated with the small sample size formula says. This example shows that, introducing practical figures for the variability, considering the estimator of the mean as being the true mean reduces the reliability by about a factor of 10. As a result, the reliability level demonstrated by the ‘Rotorcraft’ method is much closer to an ‘A’ value than to the expected 99.9%.

Like the small sample size formula, the ‘Whitehead’ method is able to take the sample size into account to derive the Load/Life enhancement factor. Assuming the simulation of one lifetime and covering the scatter in fatigue by enhancing the loads, only, figure 9 compares the outcome of the two methods.

Method 1: Rotorcraft coefficients picked up from the NH90 specification (In this case, the typical coefficient of variation is assumed to be equal to 9% along the stress axis).

Method 2: ‘Whitehead’ method with several levels of reliability (α_R= 13.6 which corresponds to CV equal to 9%).

Even if the Helicopter method is less conservative than its intent (around 99% reliability instead of 99.9% by assuming the estimator of the mean representing the true mean), this method provides much higher coefficients than what is currently the result of the ‘Whitehead’ method applied to fixed wing structures. As a conclusion, despite Load/Life factors apparently high in the ‘Whitehead’ method, when they are compared with what is the current practice for Helicopter rotating elements, or part 25 metallic structures, the demonstrated reliability level is lower. While the coefficients used for metals (airplanes and rotorcraft) have been historically related to reliability objectives, the coefficients used for part 25 composite structures are based on an objective of significance, as it can be read in the relevant paragraphs of the advisory circular AC 20-107A:

§ 7 a (1) Structural details, elements, and subcomponents of critical areas should be tested under repeated loads to define the sensitivity of the structure to damage growth. This testing can form the basis to validating a no-growth approach to the damage tolerance requirements.

…..

The repeated load testing should be representative of anticipated service usage. The repeated load testing should include damage levels (including impact damage) typical of those that may occur during fabrication, assembly, and in-service, consistent with the inspection techniques employed.

Finally, the consequence is that the current method provision for composites is more a durability demonstration than a safety related one. This is true, as far as fatigue initiation is concerned, obviously.

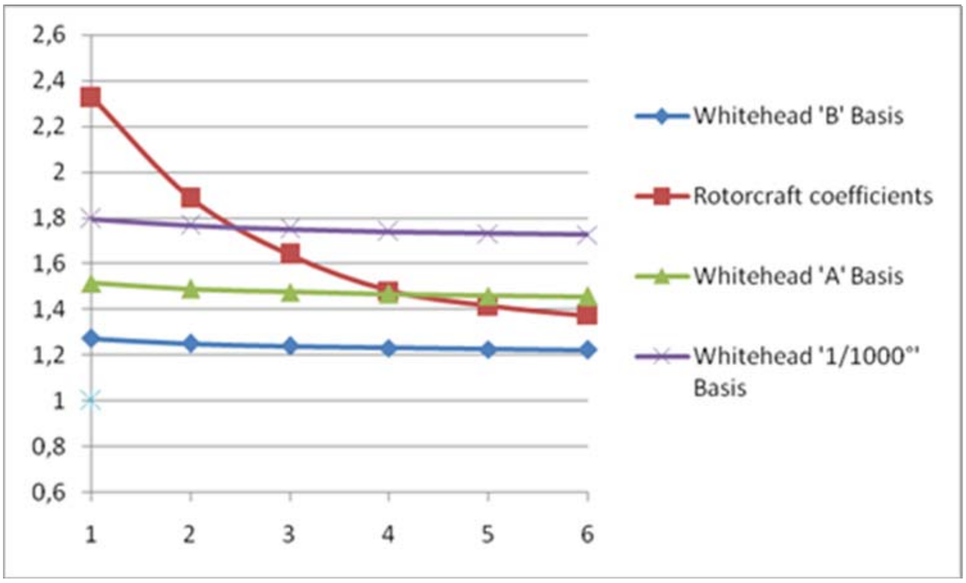

Figure 9: A comparison between the 'Whitehead method' and 'Rotorcraft' coefficients as a function of the sample size

A COMMON TEST PROTOCOL FOR STATIC AND FATIGUE AND ITS SIGNIFICANCE

While the fatigue demonstration and the damage tolerance evaluation for metal structures are based on analysis supported by test evidence, the situation with composites is essentially test evidence. Fatigue analysis, if any, is just a stress (strain) assessment to demonstrate that their level is significantly below the endurance limit of the material. This is mainly achieved by limiting the stain to already proven levels. At the end, any fatigue test performed on a composite structure is more a sort of proof testing than a substantiation of fatigue calculation regarding crack initiation or propagation phenomena. However, what looks as a simplification of the problem at the beginning may eventually become a real burden when there is a need to cope with a modification of the load spectrum. We know what to do with metals to recalculate inspection thresholds and intervals, but not with composites.

Since, unlike what happens with metals, there is no interaction between high level static stresses and fatigue behaviour with composite, it is a widespread practice to use a single test article to meet, at the full-scale test level, both static and fatigue/damage tolerance requirements. Figure 10 is a schematic representation of the resulting test protocol. The significance of the various phases of this test protocol will be discussed in the following part of this paragraph.

Phase 1: Is it a fatigue safe life or a durability demonstration?

Unlike what happens with metals, too, it is a widespread practice with composites to carry out substantiations by test articles representative of the minimum quality allowed by the process specification. For this purpose, they are provided with various tolerable manufacturing deviations (porosities or delaminations artificially simulated by teflon shims) together with low velocity accidental impact damages within the limits represented by their threshold of detectability (so-called Barely Visible Impact Damage) or a realistic threat (realistic cut-off energy), whichever comes first.

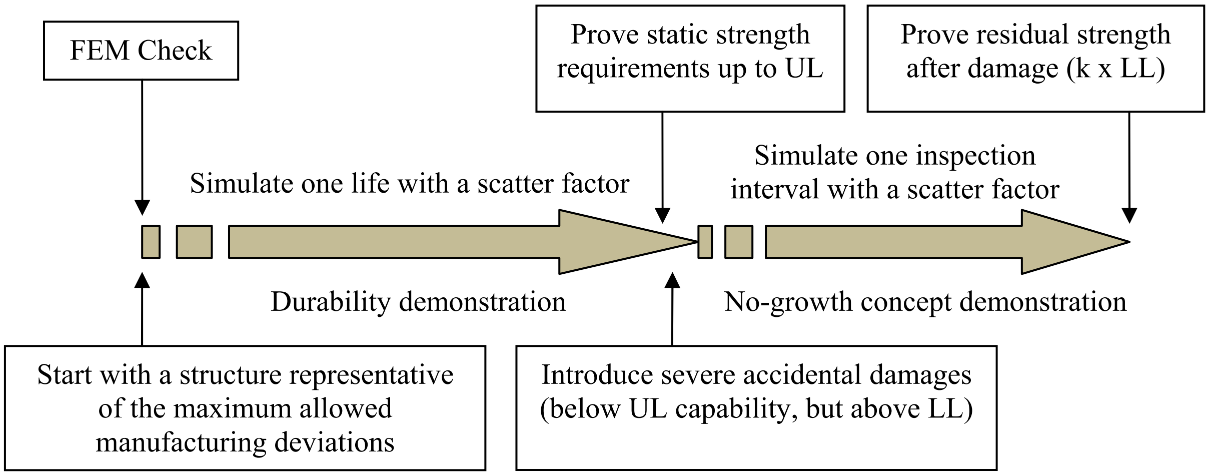

Figure 10: The full-scale test protocol

It is here interesting to go back to the historical reasons of this practice. In fact, they are three of them:

- The material is made at the same time as the structure, there will be built-in manufacturing flaws that we will have to live with,
- There will be likely accidental damages producing internal delamination without being detectable by practical inspection schemes,
- Calculation methods are often poor to demonstrate the no-growth of such manufacturing deviations or damages.

Applying a fatigue test spectrum on such test articles would correspond to a flaw tolerant safe life demonstration as defined in the Helicopter regulatory and parent guidance materials.

***Definition of 'Flaw tolerant safe-life (ref AC 29 MG8, amendment 29-42, dated 25 August 03):** *The capability of as-manufactured structure, with expected flaws, as shown by tests or analysis based on tests, not to initiate fatigue cracks or flaw/damage growth during the service life of the rotorcraft or before a replacement time.*

While in both cases, the test articles are intended to represent the lower bound of the quality of the production, the test factor used for helicopter elements, as shown in the former paragraph, is significantly higher than the one implemented for fixed wing structures. It is a shared understanding that the flaw tolerant safe life demonstration for rotorcraft elements is intended to be a real damage tolerance demonstration which is applicable to Principal Structural Elements as they are defined by § 29.571. On the other hand, the first phase of the test protocol applied to Airplane articles cannot claim more than providing a durability demonstration or a crack free life. This means no fatigue crack initiation either from design features (out-of-plan stresses) or from manufacturing deviations or damages.

Phase 2: Ultimate Loads after fatigue, what does it represent?

It is a common practice, too, to perform the Ultimate Loads demonstration after this durability phase and not before. The historical reason comes from a direct application of the guidance material which can be found in the original AC 20-107 issued in 1978, in its paragraph Static Strength.

5. PROOF OF STRUCTURE STATIC

b. Structural static testing of a component may be conducted on either new structure or structure previously submitted to repeated loads. If new structure is used to determine proof of compliance, coupon test data should be provided to assess the possible material property degradation of static strength after the application of repeated loads and should be accounted for in the results of static test of new structure.

In reference [3] again, the author explained the initial concern behind this paragraph which was a suspected degradation (i.e. without detectable evidence by inspection) of the composite mechanical properties under the combination of repeated loads and environment (cycled temperature and humidity). Another initial reason to demonstrate Ultimate Loads after fatigue was at that time the poorness of the calculation models for composites to allow for discontinuities as flaws. Today, both arguments supporting the need for a UL demonstration after the fatigue durability phase have considerably regressed and this cannot remain as a strong requirement any longer.

Phase 3: The damage tolerance evaluation.

Of major importance regarding safety, is this third phase of the test protocol intended to show the no-growth of in-service damages that would reduce static strength below Ultimate Loads capability. This is clearly a damage tolerance demonstration with a safety objective. The reason why a no-growth demonstration has so far been required for certification purpose with composites is the absence of a slow, stable and predictable process for crack growth with such materials. Even with the improvement of the calculation methods and the toughness properties of the most recent materials, this is a situation which has not improved sufficiently yet, not enough so to accept demonstrations based on a growth rate calculation as those commonly carried out with metals.

One must recognize that introducing damage growth calculations for the substantiation of composite structures is more and more becoming a matter of debate. There is first an inherent difficulty: a slow and stable growth of a damage is a very unlikely with composites, most of the time the delamination growth is totally unstable. Second, the material properties needed by the model present a lot of uncertainties, since largely influenced by the manufacturing process and environmental conditions. This is a point which has already been addressed in a former paragraph of this paper.

While the test factor is of less importance for a durability demonstration, attention should be paid on it whenever a safety related demonstration is concerned in a damage tolerance evaluation phase. At this stage, it is interesting to observe that a single test factor (or combination of a factor on loads with a factor on life) has always been used to cover three phenomena which are significantly different in their nature:

-Material degradation, if any.

-Fatigue crack initiation in undamaged material.

-No-growth of damage from existing flaws or crack tips (delamination onset).

On top of that, the data base used to derive the existing scatter factors (e.g. 1.15 on loads together with 1.5 on life) has been mostly built up from test data collected on design features which are not reputed to be fatigue sensitive. Today, there is no too much added value to test thousands of composite specimens which are not reputed to be fatigue sensitive in order to refine the currently used test factors, but there is still a lack of test data about the variability of the crack growth threshold from existing damages (e.g. a delaminated volume from an accidental impact damage for instance).

SUBSTANTIATION OF HYBRID STRUCTURES IN FATIGUE, WAYS TO CONVERGE TO A SINGLE FATIGUE TEST SPECTRUM

As composite materials usage has been limited to control surfaces or tailplanes, it has been a common practice to test these parts separately from the rest of the full-scale test airframe for certification purpose. Obviously, none of these components is a hundred per cent composite and then, for these structures including some critical large metal parts which cannot be easily segregated from the rest of that structure, there have been historically two options:

- Two different full-scale fatigue tests, one dedicated to the composite and the other one to the metal.
- One single full-scale test article where a composite, then a metal tailored spectrum, are successively applied with the replacement of the metal parts in between. While the metal parts are considered as tools in the first phase of this test, and then may need to be reinforced for this purpose, the composite ones become tools for the rest of the test.

The horizontal tailplane with a metallic center joint fitting is a good example of this situation.

In the current large commercial aircraft projects incorporating more than 50% of composites, introduced now in the wings and the fuselage, such testing strategy is no longer possible and there is a need to try to converge toward a single spectrum where none of the required demonstrations would be compromised. While fatigue tests performed so far on composite structures have shown very few findings, the situation is not the same with metals where such tests are continuously proving their efficiency in revealing numerous unexpected damages and in supporting the analysis for the Damage Tolerance Evaluation.

As a consequence, any common test protocol must not compromise the demonstrations required for metals and the question becomes:

- Applying a metal tailored test protocol on a hybrid structure, what can be missed for the composite demonstration?
- Is it possible to modify the metal tailored test protocol, to fill the gap, without compromising the demonstrations for the metal?

Figure 11 drawn from reference [3] of the author shows what the main discriminating features are, between a spectrum tailored to composite needs and to metal ones.

All tension loads being represented, a higher omission level and a more refined stepping, as we will find in a spectrum tailored to metals, cannot compromise the demonstrations required by the composite. The sole conflicting areas are the test factors to cover the material variability and the clipping (truncation) level. Application of Ultimate Loads after the fatigue durability demonstration has already been addressed in this paper and should not be a conflicting area anymore.

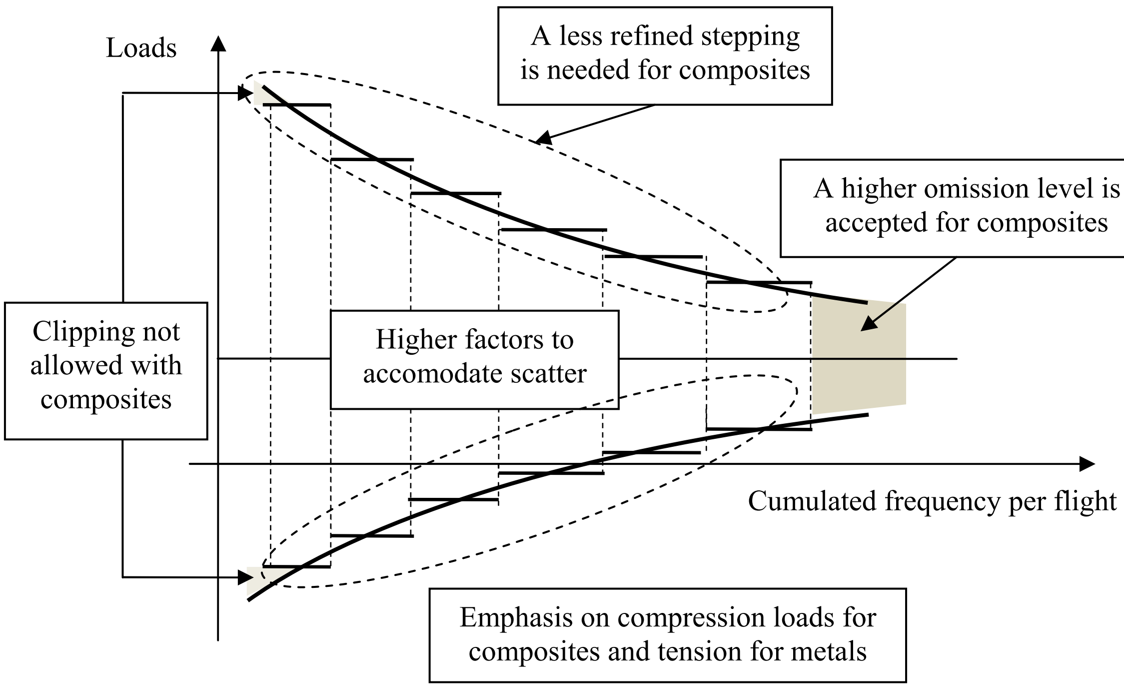

Figure 11: Main discriminating features between a composite and a metal tailored fatigue spectrum

Regarding the test factor issue, in reference [3] again, the author started to show to what extent this factor was governed by the assumptions on the material variability and that a limited shift on these assumptions would be sufficient to achieve a convergence. For instance, a combination of test factors with 1.1 on loads along with 2.5 on the life, as we can often find in metal structure tests, would be able to cover composite variability provided α_L is assumed to be equal to 1.4 instead of 1.25. Figure 12 illustrates what is the relative significance of this shift.

Such an observation is a tremendous incentive to launch new studies in order to revisit a more than twenty years old database in the light of today's materials and technologies. The first initiative in this domain is from EADS CASA in reference [24] where they came up with results only drawn from design features recognized as being fatigue sensitive, since generating out-of-plane stresses. Two materials have been tested with the same resin system: 8552/IM7 and 8552/AS4. Some of these specimens were representing stringers bonded on a skin tested either in tension or in shear, other ones were representing interlaminar shear stresses with double notch coupons tested in tension. While the typical value of the scatter in static was found to be quite close to the Whitehead value, α_R=19.63 instead of 20, an α_L modal value equal to 2.74 could be found.

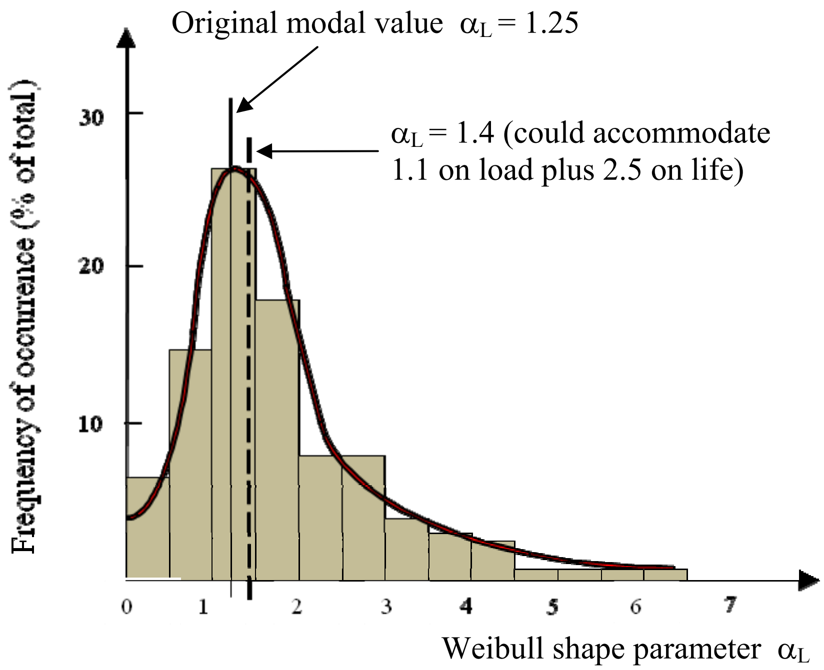

Figure 12: Illustration of the significance of the α_L assumption to accommodate a metal fatigue spectrum

A much more comprehensive investigation about composite variability and the consequences on test factors for fatigue is currently in progress at the NIAR Wichita, as can be found in reference [25]. Results have started to be released for both static and fatigue properties, with updated histograms representing the 'scatter of scatter' and their typical values. Even if the results of this programme are not yet completely released, there is a tendency to observe an improvement of the scatter when compared to the original Whitehead database.

Going back to the assumptions and the database behind the Whitehead method, the typical value selected to represent the variability in static is α_R equal to 20 which corresponds to a coefficient of variation equal to 6.5%. Not a lot of progress can be expected from an updated database with today materials regarding this value. So investigations, if any, should mainly focus on the actual variability in fatigue, obviously for those design features that could be sensitive to this kind of loading.

To summarize the test factor issue, even if most of the investigations in this domain are still to be carried out, there are reasonable chances to come up with an acceptable coverage of composite needs by existing test factors for metals, as far as a fatigue durability demonstration is concerned at least.

Regarding now the effects of truncation (clipping) and omission levels, the recent literature is very poor on this subject. So, the following early eighties concepts, as they are reflected in numerous references, as [26] and [27] for instance, are still followed. These concepts are:

- no truncation allowed,
- omission level up to 30% limit loads allowed.

In much more recent studies, as references [15] and [16] from FOI Sweden, it is shown that a higher omission level (reaching 50% of LL) could be accepted by composites. Unfortunately, there is nothing in the recent literature about a comprehensive investigation of the high load effects. However, as far as the conservatism associated with metals is due to the plastic deformation at the crack tip and its retardation effect, not a lot is to be expected from inherently brittle composite materials.

Since the general principle which is sought is not to compromise the substantiations required by the metal part of the hybrid structure, the spectrum will have to be truncated at the appropriate level, but this will create a gap to be filled for the substantiation of the composite part. Either a 'factor on life' will have to be applied on this high amplitude cycles, provided equivalence can be shown, or they will be simulated at the end of the test when the demonstrations required by the metal are fully completed. In both cases there is a need for much supporting analysis and tests at the coupon level.

THE ACCIDENTAL IMPACT DAMAGE IN THE STATIC STRENGTH AND 'FLAW TOLERANT SAFE LIFE' EVALUATION

Sensitivity to low velocity impact damage, such as those from tool drops during fabrication or maintenance operations, is at the forefront of the damage tolerance evaluation for composite structures. This issue comes from the laminated construction of currently used composite materials combining relatively poor through-the-thickness strength properties with high matrix brittleness. As a consequence, large internal delaminations can be created while the damage is poorly detectable from external inspection. This ratio between the amount of possible internal damage and detectability is illustrated by the picture figure 13 drawn from a laboratory test. The point is that, provided the impactor is blunt enough, drastic compression strength reductions may occur before damage becomes visually detectable. Consequences are less rogue for what concerns tensile strength since detrimental damages are generally visible.

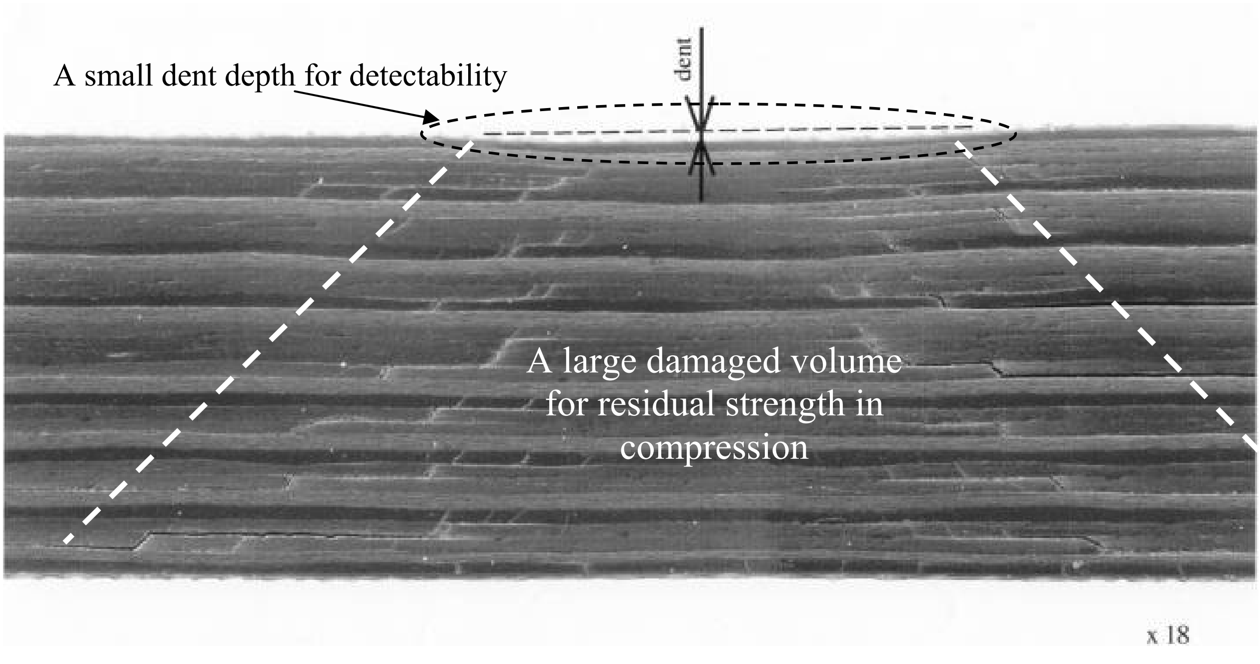

Figure 13: Low velocity impact damage micrography

Here, it is interesting to notice that low velocity impact damage, as a concern for composite structures loaded in compression, was completely ignored in the first issue of the Advisory Circular AC 20-107 in July 1978 and just addressed in the 1984 revision. Meanwhile, the NASA Langley Research Centre had carried out a comprehensive programme on composite sensitivity to accidental impact damages which has been published under several references since 1977 and on. Only one [28] among these references is mentioned in this paper. It is related to the date when these results were unveiled in Europe for the first time, on the occasion of the AGARD conference held in Athens in April 1980. Since then, thousands of other references have been published and are now available through the combined 'impact' and ‘composite’ keywords.

Before being a damage tolerance concern, low velocity impact damage is first a ‘Static Strength’ issue which is addressed as follows in the current advisory circular AC 20-107A (§6g):

It should be shown that impact damage that can be realistically expected from manufacturing and service, but no more than the established threshold of detectability for the selected inspection procedure, will not reduce the structural strength below ultimate load capability.

From this sentence, a domain limited by two cut-off thresholds can be defined and for any sort of accidental damage standing within this domain, see figure 14, the structure will have to retain its Ultimate Loads capability all along its service life since damage will never be detected. Beyond a simple ‘Static Strength’ requirement, it is a sort of ‘Flaw Tolerant Safe Life’ demonstration with Ultimate Loads capability shown at the end.

Certification needs an agreement about the metrics for the detectability thresholds and the realistic level of energy. This has been a matter of debate since the beginning and continues to be discussed nowadays. So, the rest of this chapter will tell the reader where we are now.

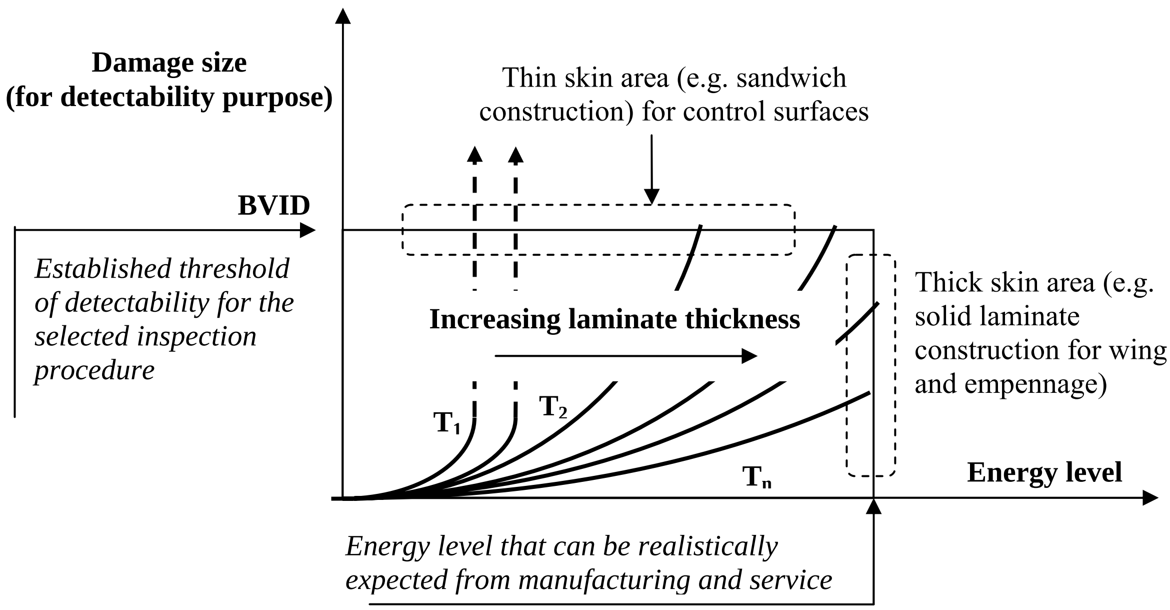

Figure 14: Illustration of the AC 20-107A, § 6g

<u>The detectability threshold:</u>

Regarding accidental impact damage detection, visual inspection only is achievable within a reasonable cost. This is the reason why it is the sole option envisaged by the aircraft manufacturers and operators, at the beginning.

However, a visual inspection comprises three levels, commonly defined as follows:

- Walkaround: conducted from the ground level to detect obvious discrepancies.

- General visual: performed at touching distance, may require some access (removal of fairings, access panels) and use of ladders or workstand to gain proximity.

- Detailed: Intensive visual examination requiring access, proximity, adequate lighting (grazing light) and any necessary inspection aids (such as mirror or hand lens).

As far as the detectability threshold is concerned, considering dent depth as damage metric is a widespread usage. This choice may be questionable for thin laminates behaving like a drum skin and where actual indentation can only be expected for a very narrow energy band just before through penetration. Figure 13 tries to represent more accurately the actual thickness effect on this damage metric. However, damage tolerance in the situation of low velocity impact damage is a matter of concern mainly for thicker laminates where large internal damages are to be associated with poor external evidences. This is a situation where indentation occurs for a broad energy band and, then, the dent depth is quite relevant as damage metric for detectability purpose.

Aerospatiale (now Airbus) in reference [29] has shown that a dent depth between 0.3 and 0.5 mm is detectable, through a detailed visual inspection, with a reliability better than 0.9 at 95% confidence. This will stand for the Barely Visible Impact Damage (BVID).

On the other side of the Atlantic, higher BVID values can be found in the military practices. The USAF requirement is 2.54 mm (0.1 inch) and the Navy suggest 0.05 inch (i.e. 1.27 mm) in reference [30]. For commercial airplanes, a detectable dent depth ranging from 0.01 to 0.02 inch, associated to a general visual inspection, has been published in reference [31]. As far as we stay in the commercial airplane world, we can assume that there is no significant divergence between both sides of the Atlantic.

An important point with accidental impact damage detection, revealed in the nineties, was the possible decay of the dent under the combination of fatigue and ageing. More details can be found in the following references [32] and [33] coming from France and Canada, respectively Since there is no reason that the delaminated plies stick together again while the dent decays the phenomenon is to be accounted for in the demonstration. With the materials and technologies available today, the initial dent depth must be above 1 mm to remain detectable by a visual detailed inspection after an ageing period covering the longest scheduled inspection interval.

<u>The energy cut-off issue :</u>

Defining the energy level that should correspond to the word *realistic,* as written down in the AC 20-107A §6, is much more difficult and controversial than achieving an agreement on the detectability threshold. In reference [3] of the author, it is explained how a realistic level of energy set at 50 joules was decided in the eighties in the scope of the A320 certification programme with no more rationale than being relatively representative of the energy of a tool box dropping from an operator waist level. In a dedicated area, the root of the horizontal tailplane, this figure was set at 140 joules to allow for the risk of tool drops from higher elevations when operating on the top of the fin. These figures remained an agreement extended to other programmes in Europe (military aircraft and rotorcraft fuselages for instance), until an alternative proposal, presented in details by the author in reference [33], could come. According to his opinion, nothing else but field surveys with records of real accidental damages was able to provide the data needed to identify the actual threat and then to build up such an alternative proposal.

Reference [30], already mentioned in this paper, is the first one known by the author providing comprehensive data in this domain. With the analysis of 1644 impacts, this survey can be considered as quite significant. Although, these records are representative of military aircraft from the US Navy Forces (F4, F 111, A 10 and F18), they can be extended to investigations on transport category aircraft since maintenance tool weights and operating conditions should not be very different. In this study, all the 1644 impact dents observed on the metallic structures have been converted into energy levels through a calibration curve obtained on a F15 wing.

Converted into joules, these results are now reported in figure 15. According to this reference, the upper limit impact energy for the aircraft surveyed was approximately 48 joules (i.e. 35 ft.lb).

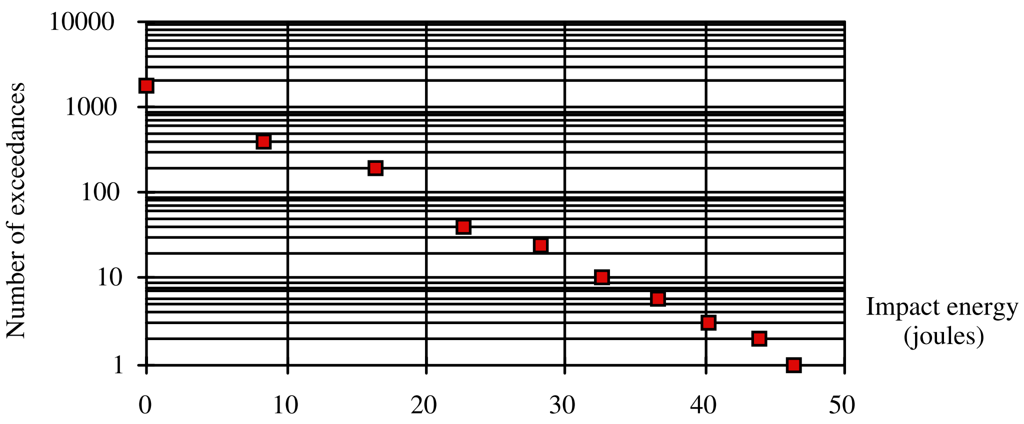

Figure 15: Results of the accidental impact damage survey, ref. [29]

Since this report does not mention the aircraft life before impact damage was identified, it is impossible to derive an impact hazard threat per flight hour (or flight cycle, or any unity representative of the usage). Nevertheless, this survey provides:

i) the order of magnitude of the expected energy, should an impact occur,

ii) the shape of the cumulative curve number of exceedance (Ne) versus energy.

The latter can be assumed as log-linear in this range of energy, with a slope of about -15 joules/Log Ne. Then, the probability (Pe) of exceeding a given level of energy x(j), should an accidental impact occur, can be then easily drawn from this curve through the following relationship:

$$\text{Log Pe} = -\,x(j) / 15$$

More rigorously, a two parameter Weibull distribution has been established from this field survey reference [30], with shape and scale parameters equal to 1.147 and 8.2 (5.98 for ft.lb energy units) respectively.

The probability that a structure in service encounters an accidental impact exceeding a given level of energy (Pa) is equal to the product of the probability of impact occurrence (Po) by the probability (Pe) to exceed this energy level. Unfortunately, the latter only is known from reference [30]. Then, in reference [34], the author proposed to assume the event to be 'reasonably probable' as such wording is defined by the regulations (CS 25.1309 with the parent Advisory Material). Figure 16 shows the result with various assumptions about the Po values, within the limits of reasonably probable, which are 10^{-3} and 10^{-5} per flight hour.

N.B. In fact the exposition to low velocity impact damage is not during the actual flight, but during the various operations associated with this flight, e.g. aircraft servicing and a shared part of the risk associated with the scheduled inspections.

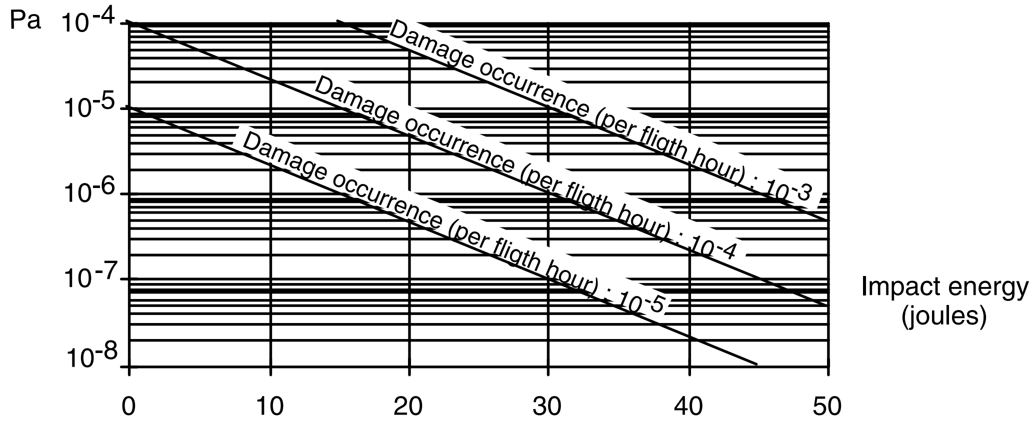

Figure 16: Resulting probability of accidental impact damage per flight hour

Now, it is time to try to associate a risk of accidental impact damage in service (probability of occurrence) to the word 'realistic'. For instance, the 'realistic level of energy' could be set at a value in such a way that a majority of the aircraft in a fleet (for instance 90%) should not have encountered an impact damage of a higher energy. Another proposal would be to state that this probability could be of the same magnitude as limit loads occurrences (why not?), etc., etc. It is not the purpose of this paper to go into the details of the calculations which can be found in reference [34] of the author, but the resulting realistic level of energy for static strength evaluation purpose turned out to be set at 35 joules in the absence of local specific threats. Field surveys carried out later on by Airbus on the A320 fleet, where the aircraft lifetime before inspection was properly recorded, could confirm this value which started being introduced in the certification programmes of the A340/600 and the A380 later on.

Because of different impact threats, for instance impact with ground vehicles, the above figure cannot be read across to new composite fuselages. In the same way, the openings, passenger and cargo doors, are likely to be much more exposed to accidental impact than a fin or a horizontal tail plane for instance. Eventually, a zoning principle, as we can find for lightning strike threats for instance, should be defined for the whole airframe, with different levels of realistic energy, depending on where we are on the airframe.

THE ACCIDENTAL IMPACT DAMAGE IN THE 'DAMAGE TOLERANCE EVALUATION'

Beyond a sort of 'flaw tolerant safe life' evaluation for those damage standing within the detectability threshold (BVID) and the realistic level of energy, whichever comes first, the damage tolerance evaluation will have to address the more severe damages which are standing outside this domain, as they are illustrated by figure 17. There will be damages detectable at scheduled inspection intervals and other ones still remaining undetectable. High velocity impact damages as, uncontained rotor burst failure, tire debris, bird strikes, etc. must be part of this evaluation. As far as the detectability (represented by the vertical axis) is concerned, the analysis will be limited to obvious or readily detectable damages that can be split into two categories:

i) Those easily detectable by regular pre-flight inspection and there is no need to associate a static strength requirement since taking off will not be permitted without further investigation,

ii) Those detectable in flight (known as discrete source damages) where a get home loads capability is required by the regulations.

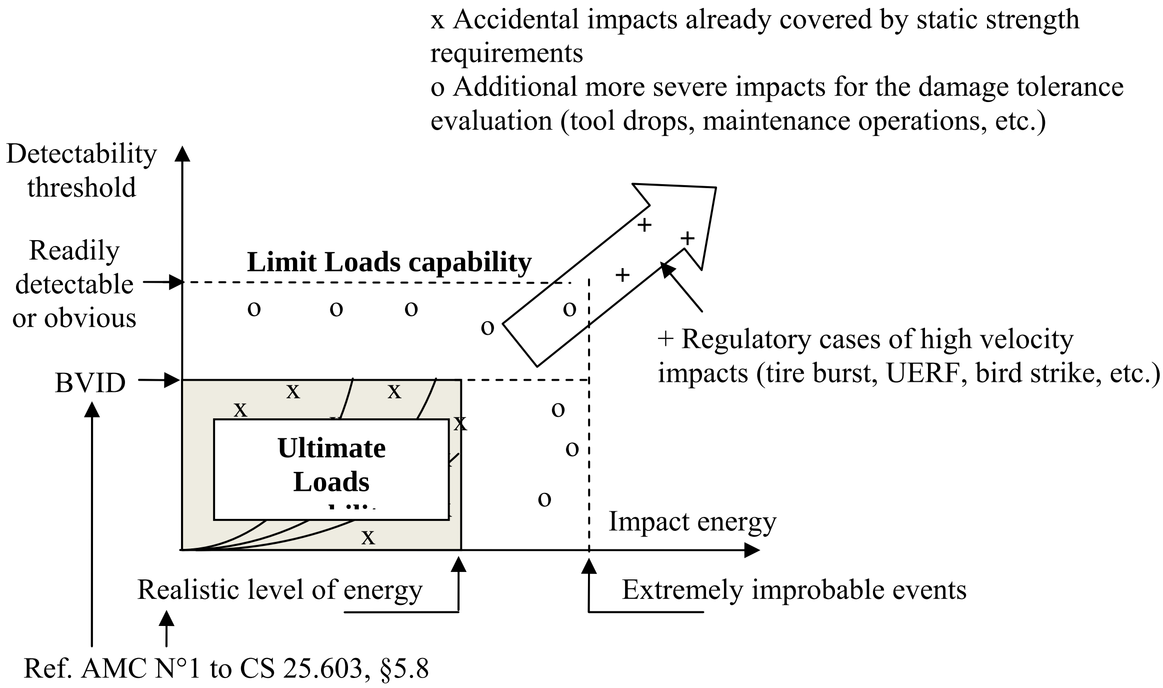

Figure 17, the domain of the damage tolerance evaluation

As it was the case for the static strength evaluation, developed in the paragraph here above, much more difficult is the definition of a limit of investigation regarding the energy level. Is it reasonable and really necessary for safety to assume accidental impacts of thousands joules?

Here, two options can be found in the existing practices. In Europe, the evaluation is limited to energy levels that are assumed to be extremely improbable, and then set at a risk of 10^{-9} per flight hour to correspond to the definitions that can be found in the rules (again in §25.1309 with its parent Advisory Material). A figure can be easily extrapolated with the assumption used just before, i.e. a Log-linear relationship between the probability of occurrence and the energy, with a slope of 15 joules per 10^{-1} increment of probability. The resulting figure is 95 joules in the absence of specific threats. While higher figures are obviously to be expected in some parts of an aircraft (door surroundings for instance), a lower cut-off threshold can be accepted for inner parts. The demonstration is completed by a large damage capability based on design precautions.

A different option seems to prevail in the United States. This option, which is illustrated figure 18, is more developed in the MIL-HDBK 17-3F, [35]. The origin of this method is to be found in reference [36].

Here, no limit of energy is specified. The principle is to investigate a range of energy broad enough to cross all the detectability levels represented on the damage size axis and to show that regulatory static strength requirements are met for each of them. For instance, at least the

Ultimate Loads capability must be shown at the BVID level. Limit Loads capability is to be shown for obvious damages (detectable before next flight) and Get Home Loads for discrete source damages. The full exercise is to be carried out, whatever the energy level required to attaining the last detectability level. Here, we can imagine that several thousands of joules may be needed to achieve an obvious damage on a self-stiffened and very thick panel.

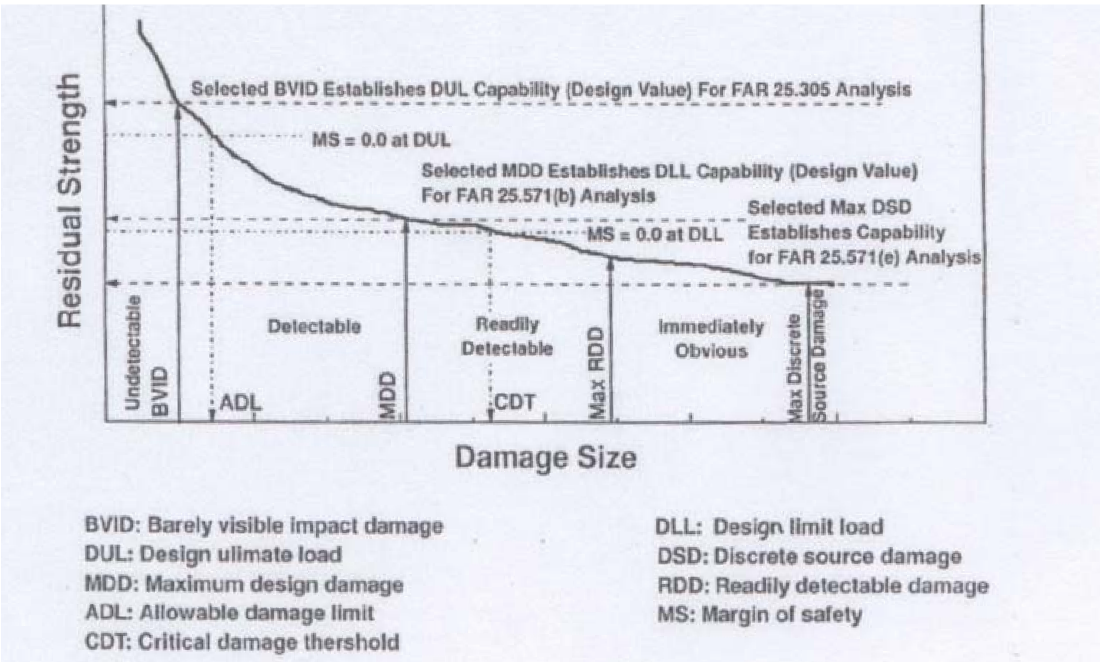

Figure 18: Residual strength requirements versus damage size ‘as per MIL-HDBK-17, vol 3F

An important advantage of this method is its inherent capability to investigate all possible ‘cliff effects’ in the behaviour of a structure that could be due for instance to the sudden failure of a substructure without visible evidence from outside the skin. However, when very high energy levels have to be investigated, the nature of the threat must be taken into account. It cannot be representative of a realistic threat to simulate a thousand joules by a drop weight test. At such an energy level, the threat cannot be much different from an impact with, let us say, a ground vehicle, provided such a situation may happen.

CONTINUED AIRWORTHINESS AND THE DERIVATION OF INSPECTION INTERVALS

The application of damage tolerance with the stable and slow growth principles establishes rules to derive the inspection thresholds and intervals based on a calculated critical damage size corresponding to a Limit Loads capability (cf. § 25.571). In the no-growth concept situation, the residual static strength curve never intercepts the Limit Loads capability level and there is no similar reference point provided to address the ALI’s (Airworthiness Limitation Items) as per the regulatory paragraph 25.1529. The main concern would be to fly a long time with a residual strength too much below Ultimate Loads capability, as illustrated figure 19, leading to a less safe situation than the outcome from metallic structures. Such a concern was captured as follows when writing the AC 20-107A: §7 PROOF OF

STRUCTURE FATIGUE/DAMAGE TOLERANCE, bullet a (4): *For the case of the no-growth design concept, inspection intervals should be established as part of the maintenance program. In selecting such intervals the residual strength level associated with the assumed damages should be considered.*

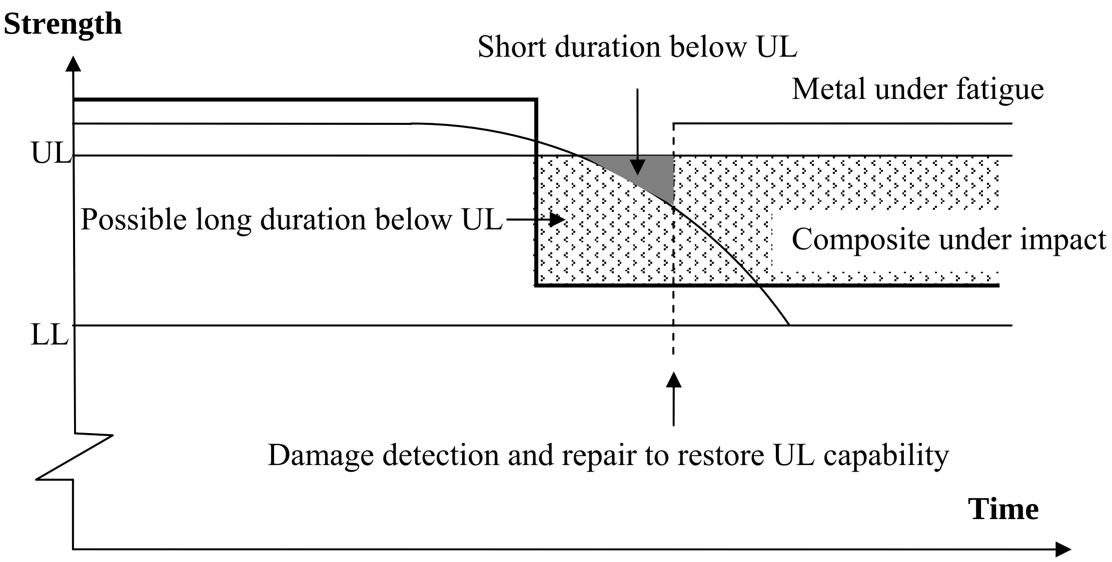

Figure 19: illustration of the issue associated with the no-growth concept

The interpretation is: the more severe the damage, the earlier it must be detected. However, when trying to implement such principles in the mid eighties it was also noticed that for a given strength reduction, the more likely the occurrence, the earlier damage should be detected. Figure 20 illustrates how the inspection interval has to be tailored by both the static strength reduction due to the assumed damage and its probability of occurrence.

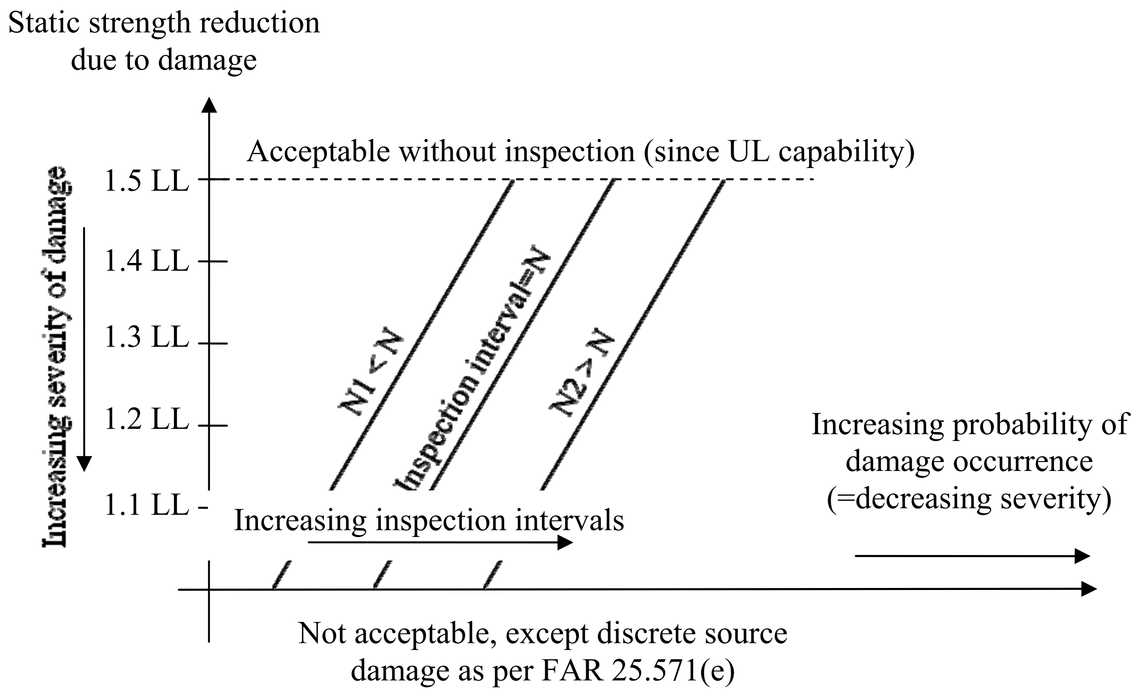

Figure 20: How the inspection interval is tailored by both damage severity and likelihood

Because the risk of accidental damage is very dependent on the structure zone under concern, probabilistic approaches have been developed, requiring an assessment of impact damage threats in terms of energy level versus the probability of occurrence. These probabilistic approaches can be more or less complex and difficult to handle, depending on the number of random variables allowed for in the analysis. However, their principle remains the same and consists in the calculation of the risk to be in the situation illustrated figure 21, which is the combination of a residual strength after damage at a given level with a load of the same intensity. An allowable value of the risk is assumed and inspection intervals are calculated to guarantee the target is met.

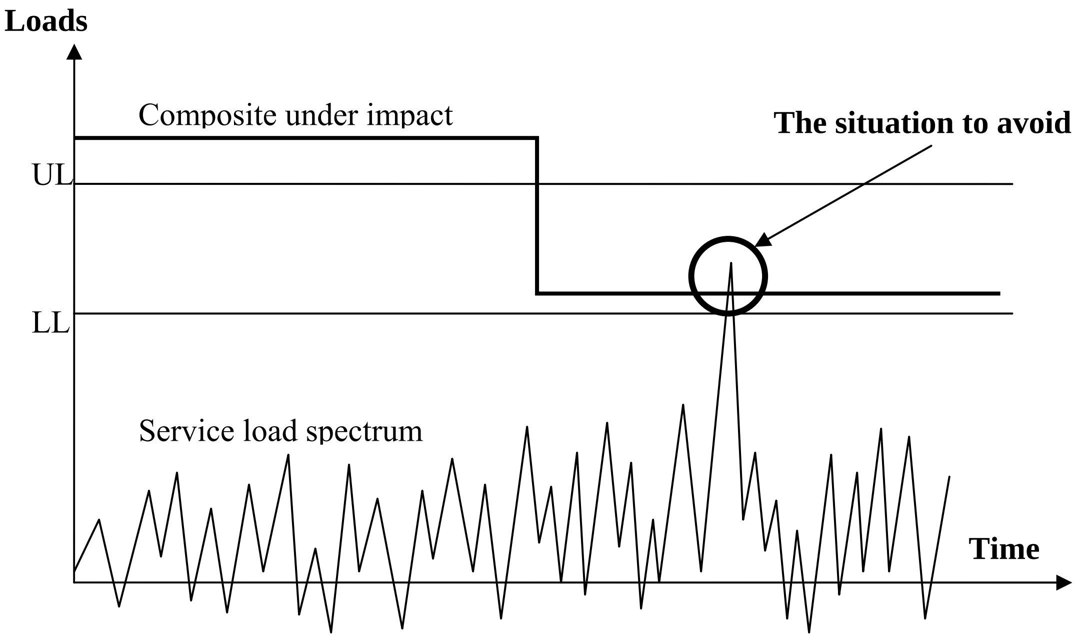

Figure 21: Illustration of the probabilistic approach target

As far as damage tolerance addresses Principal Structural Element (PSE) the failure of which would be catastrophic, this maximum risk (allowable quantitative probability) must be set at 10^{-9}/flight hours. This figure corresponds to the definition of 'Extremely Improbable' as specified by the regulation (CF § 25.1309 and its parent advisory circular) which applies to equipments, systems and installations.

There has been an abundant literature on probabilistic approaches applied to composite design and a selection of them is compared in the FAA document [37]. As part of this selection, a simplified semi-probabilistic approach proposed by the author is described in more details in [34] and has been incorporated in the MIL-HDBK-17-3F.

The general principles of the probabilistic approaches which can be implemented to address this issue are as follows:

If 'B' is the probability to have an accidental impact damage reducing the structure static strength down to k.LL capability and 'A' the probability to have a load exceeding that level, the probability of both events in conjunction is noted $P(A \cap B)$, with the following relationship:

$$P(A|B) = \frac{P(A \cap B)}{P(B)}$$

$P(A|B)$ is the conditional probability, which means the probability of A, assuming that B has occurred.

The problem can be addressed in a more sophisticated way by incorporating also in the relationship the probability of damage detection, and splitting the probability B into two parameters, which are the probability of impact damage occurrence and the probability that a given level of energy is exceeded. The latter has already been done in the former paragraph of this paper, dealing with the accidental impact damage in the static strength evaluation.

The problem can also be expressed under the form of a joint density function, as follows, for the purpose to apply reliability analysis methods:

$$P_f = \iint_{r-s\leq 0} f_{R,S}(r,s)drds$$

Where P_f is the probability of failure, f_R the density function of the residual strength after damage and f_s the density function of the service loads.

This paper will describe another method intended to largely simplify the probabilistic assessment, while providing practicable tools for design. This method is based on a similarity with a problem already addressed in the certification of aircraft structures which is the interaction of systems and structures. When an aircraft is provided with a load alleviation system (either for gusts or maneuvers), in case of failure of this system, and until detection and repair, the structure may have a lower Factor of Safety (FS). Certification specifications from EASA allow such reduction of this Factor of Safety with a level depending on the probability of being in failure condition. This is illustrated by figure 22 drawn from the regulatory paragraph CS 25.302 and its appendix K. In this similarity, the accidental impact damage will be assumed as the hidden failure of a system interacting with the structure and the same Factor of Safety reductions will be applied for continuation of flight. The reason why this method is said to be semi-probabilistic is because all the parameters used as inputs are not of a probabilistic nature, i.e. the residual static strength capability after damage is deterministic.

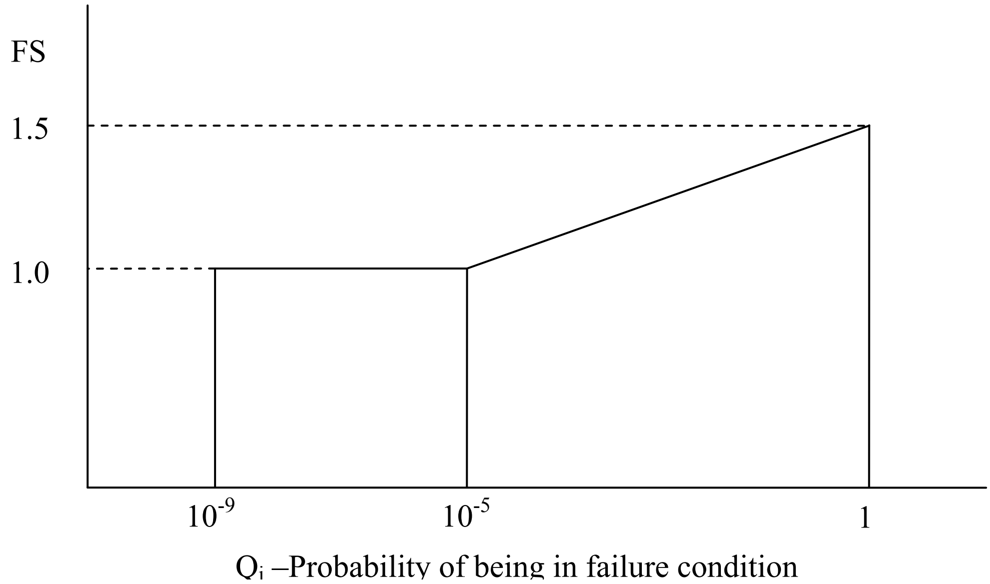

Figure 22: factor of safety for continuation of flight (cf. CS 25.302 and appendix K

$Q_j = (T_j)(P_j)$ where:

T_j = Average time spent in failure condition j (in hours).

P_j = Probability of occurrence of failure mode j (per hour).

In practice, the scheduled inspection programme that is implemented in service is not derived from the calculations made in the damage tolerance evaluation of the structure, but the structure is sized in order to meet the safety requirements with an achievable inspection programme. Achievable means respectful of operational and economical conditions. In these respects, a structural inspection programme with a threshold at 0.5 Design Service Goal (DSG) followed by 0.25 DSG inspection intervals is the general intent at the design level. Figure 23 illustrates the sizing criteria derived from the application of such a simplified semi-probabilistic approach. This graph shows where the curve residual static strength capability of the structure versus impact energy must stand to meet the reliability objectives.

Let us try now a numerical application of this method assuming a log-linear relationship between impact energy and the probability of damage occurrence (as seen in a former paragraph) and an aircraft design service goal (DSG) equal to 80,000 Flight Hours. The longest time spent without structural inspection will correspond to the intended inspection threshold that will be set at 0.5 DSG, which means 40,000 FH.

NB: In the application of the original method for the interaction of systems and structures, the average time spent in failed condition is generally assumed to be equal to half the inspection interval, here 20,000 FH.

When Q_j=1 and T_j=20,000 hours, P_j= 5.10^{-5}/FH

When Q_j=10^{-5} and T_j=20,000 hours, P_j= 5.10^{-10}/FH

The impact energy level E_U will then correspond to a probability of occurrence allocated to a Flight Hour equal to: P_j= 5.10^{-5}

The impact energy level E_L will then correspond to a probability of occurrence allocated to a Flight Hour equal to: P_j= 5.10^{-10}

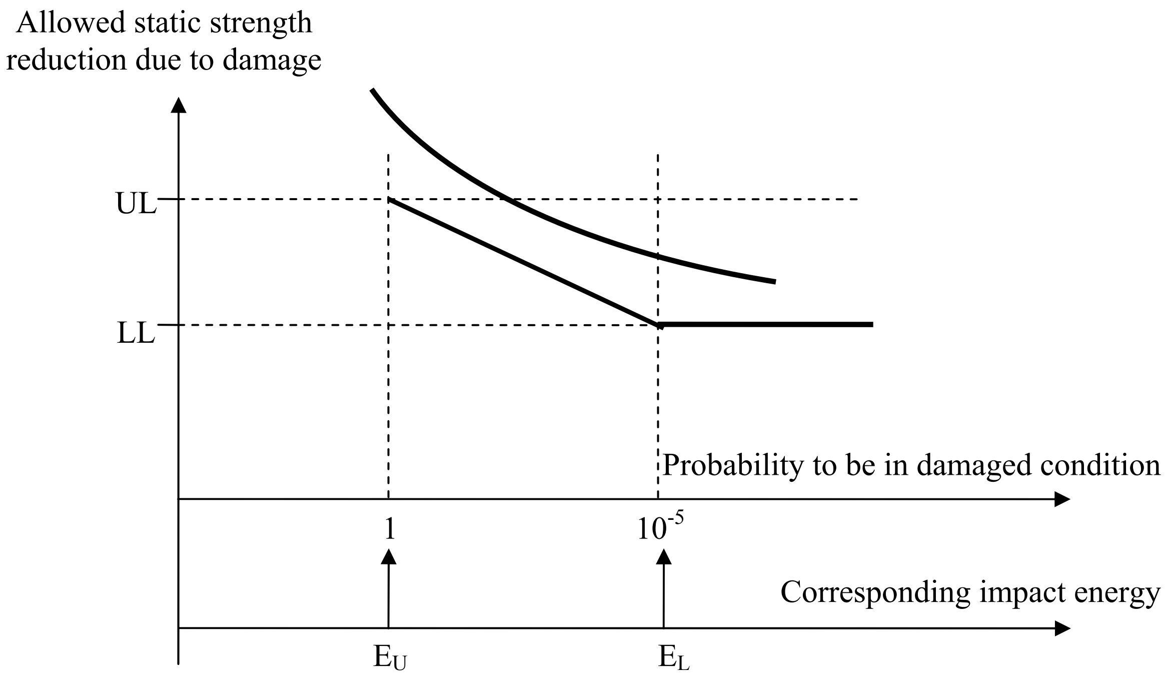

Figure 23: sizing criteria vs. impact energy

Let us assume now a probability occurrence of 10^{-5}/FH for a 35 joule impact and a 15 joules slope per 10^{-1}, E_U will then be equal to 24.5 joules and E_L equal to 99.5 joules. The resulting sizing criteria will be a residual Static Strength capability after damage standing above a segment linking the point UL capability for 24.5 joules and LL capability for 99.5 joules.

Now, let us make the calculation with Tj=80,000 FH (DSG).

When Q_j=1 and T_j=80,000 hours, P_j= $1.25.10^{-5}$/FH

When Q_j=10^{-5} and T_j=80,000 hours, P_j= $1.25.10^{-10}$/FH

For DSG, the E_U value then becomes 33.5 joules and E_L 108.5 joules.

What is interesting to notice, now, is how this segment moves when increasing the inspection interval up to the design service goal (DSG) that would mean no need of structural inspection. Moreover, in this exercise, the accidental impact will be assumed to occur very early in the aircraft usage that means an average time spent in failure condition equal to 80,000 FH. The result illustrated figure 24 shows that setting a scheduled inspection programme instead of no inspection all along the DSG seems to present a fairly poor advantage.

Again, this method requires a good assessment of accidental impact damage threats and highlights the need for carrying out field surveys to support the data that will be used.

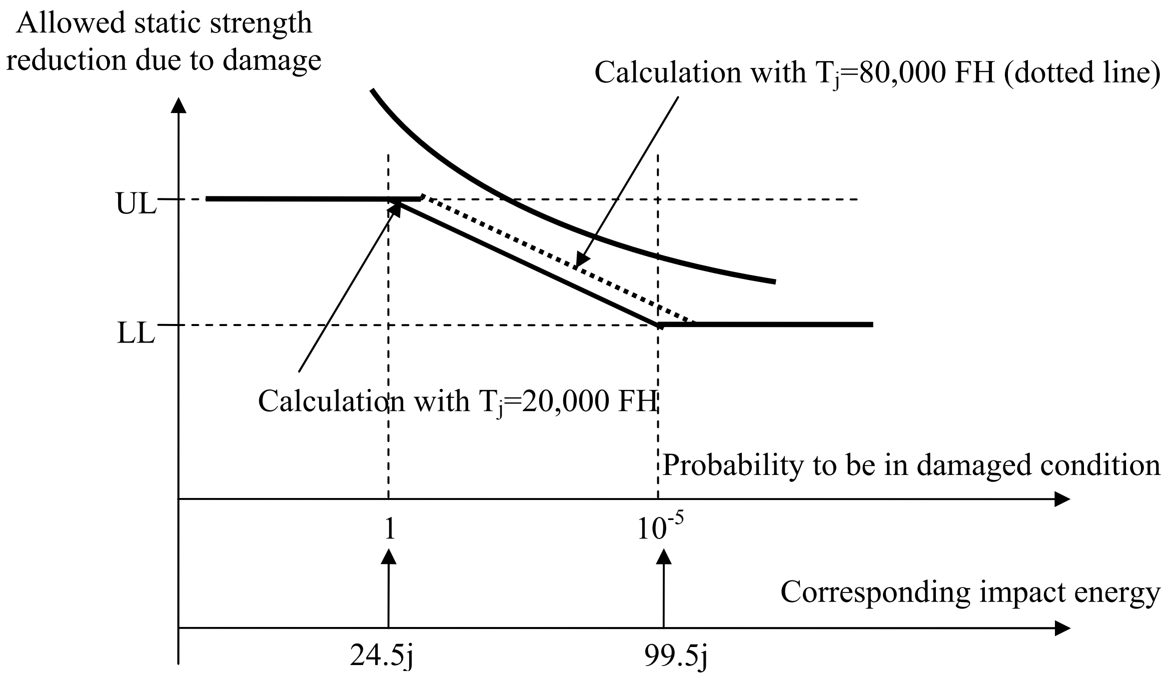

Figure 24: Result of the numerical application

SUMMARY AND CONCLUSION

Composites will now represent more than 50% of the airframe weight in the forthcoming wide-body commercial aircraft programmes. In this respect, and as predicted in the ICAF 1977 Plantema memorial lecture, composites have proved to be the 'structures of the future'. While weight and cost savings have always been the prior arguments to promote their introduction in past programmes, one must admit that the former is often difficult to assess and the latter rarely achieved. Then, it is the opinion of the author that the actual reasons for their success are to be sought elsewhere. One of these reasons is the constant willingness of the aircraft industry to show that they are tirelessly exploring the perspectives offered by the new technologies and able to offering the expected benefits ahead of their competitors. As important as this argument of a more 'marketing' nature, is the very good service behavior so far demonstrated by a hundred million flight hours today largely exceeded by commercial aircraft comprising one or more primary composite structures. Incomparable fatigue and corrosion material behaviors, as well as damage tolerant design principles inherently provided by mechanically fastened constructions, and a very low exposition to accidental damage occurrences, are the reasons for this success. A different opinion may prevail for thin-walled construction, as sandwich structures for instance, where a lack of robustness or some bonding line deficiencies can have been found in a few circumstances.

The application of composites to commercial aircraft fuselages raises several challenges with respect to various disciplines, not all related to 'structural integrity'. One of the structure related challenges concerns damage tolerance in accidental impact occurrences. In a paper [38] presented at a FAA workshop in Chicago, 2006, Airbus mentioned the results of an IATA survey where it was shown that more than 35% of the structure damages was due to ground handling operations, plus another survey, performed on the A320 fleet, where it was shown that around 70% of the structure repair needs concerned the fuselage. This is very clear evidence of a much higher exposition of the fuselages to accidental damage occurrences and a much higher demand for robust and damage tolerant design principles to accommodate this issue. Sizing will have to be made against threats which are to be much more accurately known. In this respect, field surveys to assess the real threat on fuselages cannot be avoided. On top of that, adequate design principles will have to be implemented to accommodate very rare but severe events. This is called Structural Damage Capability as already developed in many papers, as for instance reference [39] presented at the last ICAF symposium in 2007.

As far as fatigue is concerned, increasing the operational loads in the structures by reducing the static strength margins down to their minimum values should not yet make fatigue critical for composite structures. However, this will likely lead to new situations where more fatigue cracks will develop in areas where out-of-plane stresses may be found. Because of the unstable fatigue crack growth phenomenon always observed with composites, wherever such damages may occur, structural integrity will have always to be maintained by containment features that will significantly retard or arrest the cracks. If we try to translate this in terms of research topics for academia, needs are mainly focusing now on a better modeling of 3D stresses in the laminates, a better prediction of delamination onset allowing for the inherent material variability, and a better calculation of residual strength in the presence of damages. Still in the fatigue domain, the development of unified full-scale fatigue tests procedures for hybrid structures is also a very important topic for further investigations.

Another point to be raised today concerns the future of metals in aircraft structures with, in this particular ICAF forum, the associated need for expertise, development works and academia research in fatigue, corrosion and fracture mechanics. It is the opinion of the author that metals will not give in more ground in the future than today. When 50% of an aircraft structure is made out of composite, most metal applications are concentrated to 'structural knots', heavily loaded and of such complexity in their design that so anisotropic composites will never be able to accommodate. Applying very ambitious weight saving sizing policies to these structurally significant parts, with their inherently complex design, has never been so much demanding in modeling and material characterization. As a consequence, younger generations of engineers must remain convinced that the opportunities still offered by fatigue and fracture mechanics sciences remain numerous, before thinking about their reorientation and mass migrating from metals to composites.

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