CERTIFICATION OF LARGE AIRPLANE COMPOSITE STRUCTURES, RECENT PROGRESS AND NEW TRENDS IN COMPLIANCE PHILOSOPHY

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Abstract

Because of the unconventional structural behaviour of composite materials when compared with metals, special attention is required in the certification procedures of aircraft structures involving these new materials. To cope with the first applications in the late seventies, acceptable means of compliance specific to composites were developed in the form of appropriate guidance material (Advisory Circulars) issued by airworthiness authorities (AA). The first advisory circular covering composite certification (AC 20 107) was issued by the Federal Aviation Administration (FAA) in 1978. That AC was intended to be periodically updated in order to reflect progress drawn from experience. Its latest issue, jointly prepared and agreed upon by FAA and the European member countries of the Joint Airworthiness Requirements (JAR) group, dates from 1984. French AA took part in the discussions and are willing to continue in such a way, at a time when updating this AC is going to be placed on the agenda.

From the experience gained through major programmes from the Falcon 10 carbon wing (V10F) to the latest AIRBUS aircraft and the ATR 72, an experience shared in most cases with other European agencies, this paper deals with three major key issues where compliance philosophy is evolving or should evolve in the next future : second source material qualification, conditions to simulate environmental effects, and damage tolerance demonstration for accidental impact damage.

I - Introduction
An overview of differences between composites and metals regarding certification

When composite materials were introduced in aircraft primary structures, existing certification procedures used for metallic airframes could no longer be applied without allowing for the differences between the two generations of material. Such differences may be obvious, from the material basic evaluation, or may be discovered after a significative in-service experience. The main inherent composite characteristics or properties having so far tailored the composite certification procedures currently used are the following :

- sensitivity to environment (temperature, humidity) within the aircraft operational envelope, at least for matrix dominated properties, due to their organic matrices
- extreme sensitivity to in-plane delamination, due to their laminated construction with a strongly weak direction. Practical consequences concern the susceptibility to low velocity accidental impact damage, and to the induced out-of-plane secondary loads in complex built-up structures
- less sensitivity to fatigue than metals, as far as thermoset matrices are concerned, because of their brittle behaviour. Static strength requirements become the dominating issue
- structural properties cannot be more process dependent, because the material does not exist before the part is built. Consequences are mainly seen on material qualification procedures and quality assurance
- poor electrical conductivity, direct and indirect effects of lightning are to be addressed.

Guidance material - Advisory Circulars - issued by airworthiness authorities (AA) to help in certification, provide acceptable means of compliance taking into account these differences.

From the experience gained in service or in laboratory evaluations, the interpretation of these inherent behaviours has been evolving from the beginning and is still likely to evolve in the future. The successive issues of the advisory circulars reflect the main steps in the understanding of the problem. Before developing in this paper some recent progress and new trends in compliance philosophy, the next section will go back over the steps having preceded the latest issue of the composite AC.

II - Composite Advisory Circular previous issues

The AC 20 107, 7 October 1978

This earliest guidance paper issued by FAA (Federal Aviation Administration) aimed at dealing with the first composite primary structure certification applications such as mainly the B737 horizontal stabilizer, the DC10 and L1011 vertical fins. All these parts were prototypes developed in the scope of a NASA supported programme. These early guidelines emphasized environmental sensitivity and the need for it to be allowed for in structural substantiation tests, either by appropriate simulation or by enhancing the loads to be demonstrated. Consequences of some composite inherent properties on fire resistance, lightning strike resistance and quality assurance were also addressed.

The French STPA 81/04 note, issue 2, 2 December 1981

This paper was the interpretation by the French airworthiness authorities of the a.m. AC 20 107 which they deemed insufficiently explicit in some respects. The main novelty provided by this paper was the differenciation between 'degradation' : alteration of intrinsec material properties due to ageing for instance, and defects : flaws or damage. Defects are assumed to be detectable by Non Destructive Testing (NDT), degradation not. Because it was suspected that repeated loading might contribute to degrade material properties, ultimate load carrying capability was required to be demonstrated after fatigue, as long as degradation was not detectable.

This paper served as the certification basis for the Falcon 10 carbon wing programme and the A310/300 vertical fin, and as a starting point for discussions on composite certification procedures, between the European members countries of the Joint Airworthiness Requirements (JAR) group in the early eighties.

The draft issue AC 20 107 A

This paper was submitted for comments to the JAR early in 1983. It was the first initiative to try to reach a FAA-JAR agreement upon an advisory circular before its implementation. Consistency with the FAR 25 45th amendment introducing damage tolerance was the main reason for updating this AC. Never addressed so far, inherent composite sensitivity to accidental impact damage was pointed out in that draft issue. The no-growth concept was proposed to comply with damage tolerance requirements.

The AC 20 107 A, 25 April 1984

This issue, the latest one is thus still applicable now; it reflects results of discussions and the agreement achieved between American and European specialists on the a.m. draft. The following JAR contribution to these new guidelines was far from negligible :

- impact damage that can be realistically expected from manufacturing and service, but no more than the established threshold of detectability for the selected inspection procedures should be assumed. Substantiations with test articles representative of the minimum quality according to the end-product control plan (i.e. provided with tolerable manufacturing defects, damage etc...) were then implicitly required.
- In the case of the no-growth concept, the selection of inspection intervals for those damage reducing static strength below ultimate loads (UL), should consider the residual strength level associated with the assumed damage. In other words, the more impact damage reduces strength, the sooner it should be detected.

The JAR ACJ 25 603, 16 June 1986

This is the European version of the AC 20 107 A, with some editorial changes for consistency with JAR rules. The main sections of this advisory circular are laid down figure 1.

The purpose of this paper is not to review the current certification procedures related to each of these sections. This has been widely developed in some recent outstanding references as [1], [2], [3]. This paper is only addressing three selected key issues where important discussions between manufacturers and airworthiness authorities took place in the latest large airplane type certifications in Europe, entailing a noteworthy evolution in compliance philosophy. These issues are:

- second source material qualification
- conditions to simulate environmental effects
- damage tolerance demonstration for accidental impact damage.

1 - Purpose
2 - Definitions
3 - General
4 - Material and fabrication development
5 - Proof of structure - Static
6 - Proof of structure - Fatigue/Damage tolerance
7 - Proof of structure - Flutter
8 - Additional considerations

Figure 1 - ACJ 25 603 major sections

III - Second source material qualification

The problem

Most European aircraft composite structures are certified with only one material supplying source. Several industrial problems arise from this situation, the most critical one being the entire subordination of the aircraft programme to the reliability of the selected source. Now at a time when the introduction of composites in aircraft structures is reaching a "plateau" in terms of components being involved (for large airplanes, applications are so far restricted to secondary structures, moveables, and sometimes stabilizers; the introduction of composites in large fuselages or wings is not foreseenable in the next future) efforts in composite development are mainly concentrated on economic issues and weighing on material prices by diversifying the supplying sources may be an important contribution to cost reductions. So, second source material qualification is now taking a large place in airworthiness authorities/manufacturers meeting agendas. Because this issue is not specifically discussed in applicable AC's, it was necessary to develop a compliance philosophy to address this subject. On account of the number of material change substantiation programmes which were to be submitted to approval in the scope of the A320 post-certification activities, this opportunity has been taken to try to come up with a coordinated approach between the AIRBUS partners and European airworthiness authorities involved in this aircraft programme. This approach which has also been extended to the ATR72 outer wing is outlined hereafter.

To understand the problem of material change with composites, it is necessary to go back over the inherent differences existing between those materials and metals. Composite materials can develop their mechanical properties only when the component is processed (or at least the resin cured). These properties are therefore, hardly less, process dependant and must be interrogated through specimens representative of all the critical details of the structure design. This is the main goal of the building block approach with its associated pyramid of tests (fig 2). This pyramid is the consistent way to prepare and present structural substantiations for airworthiness approval. From the elementary coupons to the full-scale component culminating at the top, the successive stages of this pyramid represent the progressive design complexity, allowing the validation, step by step, of the models used in structural calculations.

Because of the large amount of testing which is needed to cover all critical design features taking into account the effects of adverse environment on failure modes and variability, structural substantiations with composites are costly in time and money. This is the reason why in most cases it has been possible , within the budgetary and time constraints of the aircraft programme, to approve only one material. As ACJ 25 603 states (c.f. §§ 5.5 and 6.2.1) that the test articles are representative of production structure, those substantiations are only valid for this material and called in question again for any change in the material itself or in the processing route.

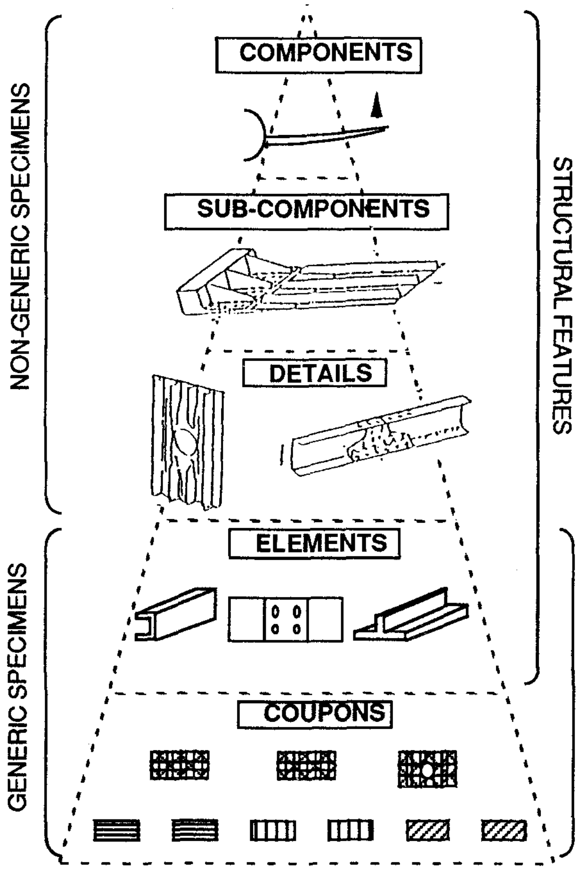

Figure 2 - The pyramid of tests

What is a material change with composites?

A standardized material classification, as can be seen with aluminium alloys, does not exist with composites, which are still called for through their manufacturer's tradenames. This means that no equivalence between two materials can be 'abinitio' assumed, except if both materials are made from the same fibre, the same resin system and the same prepregging machine. Therefore we are in the situation of a material change in any of the following cases:

a - A change in one or both of the basic components:
- resin
- fiber (including the sizing or the surface treatment alone).

b - same basic components, but change of the prepreg itself:
- prepregging process (e.g. from soaking to hot melt coating)
- tow size (3k, 6k, 12k) with the same fibre areal weight
- prepregging machine at the same supplier's
- supplier change for a same material (licensed supplier).

A classification is to be made between a new material which is intended to be a replica of the former one (case 'b') and a truly new material (case 'a'). So, two classes are proposed:
- **'Identical materials'** in case of a replica.
- **'Alternative materials'** for truly new materials.

Moreover, as far as the processing route governs eventual composite mechanical properties, using exactly the same prepreg but modifying either the curing cycle, or the tooling, or the lay-up method (e.g. hand-made to automatic) amounts to a material change.

Within the 'identical materials' class, a sub-classification can be made between a change of the prepregging machine alone at the same supplier's (extension of the output capacity), and a licensed material. It is more likely that the licensed one may be different from the reference.

Case 'a' (alternative materials) should always be considered as a change of major importance. It is not recommended to try a sub-classification according to the basic component which has been changed, as far as some behaviours (e.g. notch sensitivity) are partly governed by interfacial properties which may be affected either by a fiber or a resin change. This point is illustrated by figure 3 drawn from reference [4]. In this example, hole coefficients (ratio between holed and plain strength) have been measured for all the six possible combinations involving two types of fibers and three types of resin system (the same batch being used for every type of fiber and resin system). Results show that this coefficient significantly vary from one fiber/resin combination to another, without being specifically correlated to the fiber or the resin system alone. In other words, because of this compatibility problem, changing both fiber and resin may lead to less variations in mechanical properties than changing one basic component alone.

Resin / Fibre	914	N 5208	N 5245c
T300	0.70	0.76	0.60
AS4	0.67	0.64	0.63

Figure 3 - Hole coefficients

Substantiations to be provided

Substantiations are based on a comparability study between the structural performances of the material approved for type certification, and the second source material. The procedure should normally involve the following steps:

- identify the key material parameters governing performances
- define the appropriate tests able to measure these parameters
- define the pass/fail criteria for these tests.

But with composites, as long as we are not capable of accurately identifying the key material parameters governing processability, there will be a need for tests directly interrogating material performances through specimens representative of the actual design details of the structure. Standardized test programmes defining exactly what kind of tests are to be performed according to the nature of the material change alone cannot be issued, and the method then consists in selecting - within each pyramid of tests in the building block approach - those tests which are to be duplicated for the demonstration. The test programme must be discussed each case individually between airworthiness authorities and the manufacturer. Mainly based on the engineering judgment, the extent of testing required depends on the airworthiness significance of the structure, the complexity of the design with its associated process sensitivity, and obviously the nature of the material change.

Guidance papers (AC's) define five levels of testing: coupons, elements, details, sub-components, and component which are illustrated on figure 2. The lowest two levels (coupons and elements) referring to generic specimens constitute the data base, and may therefore be common to several substantiation programmes. The other three levels are non-generic specimens, specific to the project under consideration. For material change submitted in the scope of the A320 post-type certification, it has been agreed that the extent of testing could be restricted to generic specimens for replica materials, but should involve non-generic specimens for the case of alternative materials. Nevertheless, substantiations should always cover weak points of composites regarding their structural behaviour (e.g. environmental sensitivity, static notch sensitivity, impact damage susceptibility).

For this aircraft, six material change substantiation programmes, involving two replica materials and four alternative ones, as they are previously defined, have been discussed and approved along these guidelines.

IV - Conditions to simulate environmental effects

Composite ageing under environmental conditions was the major concern at the beginning of the composite era in the seventies. For no-bonded structures, the *experience now gained in service*, associated with a large amount of test results, have strongly cleared away our doubts. It does not mean that composite material ageing does not exist, but consequences are now considered to be limited and predictible, that is able to be allowed for in the design, dimensioning and certification of the structures.

It has now been proved that epoxy matrix composite ageing for an aircraft is essentially due to the effect of moisture uptake, and it is a satisfactory approximation to assume a biunivocal relationship between the moisture content and the residual mechanical properties. This assumption is drawn from results of several test programmes where specimens going from coupons to full-scale parts have been submitted to hygrothermal profiles representative of transport aircraft or fighter operations. Examples of full-scale parts tested under such conditions in France (inCEAT) are summarized figure 4, while figures 5 and 6 show the typical hygrothermal cycles developed in this respect. These experiences have shown that the static strength reductions measured after such hygrothermal ageing were not significantly greater than what could have been expected from the knowledge of the eventual material moisture content.

Structure parts having been submitted to a fatigue test combined with a hygrothermal profile in CEAT
-Airbrake A300B
-Aileron F50
-Aileron Mirage F1
-Spar box V10F

Figure 4

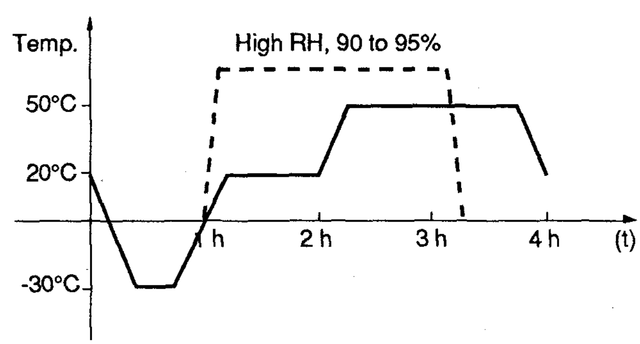

Figure 5 - Hygrothermal profile for a civil aircraft
(e.g. programme V10F)

Extending to any kind of composite material, the assumption that the effect of moisture content on the mechanical property degradation is independent of the hygrothermal history having led to that content would be unwary. At a time when quite chemically different new organic matrices are developed to improve toughness mainly, tests at the coupon level at least should be performed to check whether or not this assumption is still valid.

With epoxy matrices used so far, the problem is then transferred to the determination of the actual composite moisture content and temperature in the most adverse situation. While the maximum temperature to be considered is not difficult to assess and check in a short period of time (common values are 70°C - 80°C for subsonic transport), anticipating the end-of-lifetime moisture content is much more complex. This content is not only dependent on the aircraft operating conditions, but also on the material itself. According to their chemical formulation mainly, organic matrices are more or less water absorbent and a standardized value of that moisture content cannot be issued and applied to any system. Two methods are available to derive this value : i) analysis with a diffusion model, ii) equivalent steady conditions.

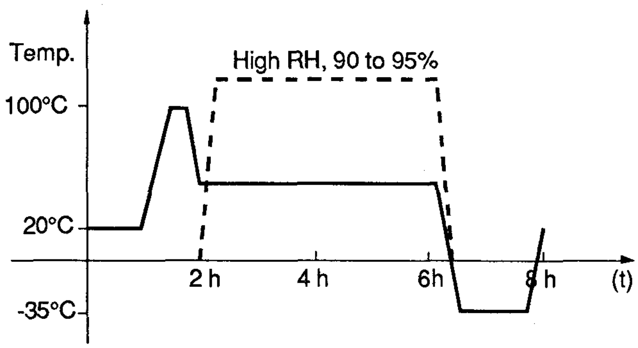

Figure 6 - Hygrothermal profile for a fighter aircraft (e.g. aileron Mirage F1)

Analysis from the hygrothermal history

Fick's law is widely used as a diffusion model for composite moisture uptake prediction. Limitations of this model are due to the strong dependence of the material parameters needed by the model on temperature, relative humidity, physical ageing, thickness, stacking sequence..., in such a way that accurate calculations in long term exposure are not practical.

Equivalent steady conditions

The second method consists in the definition of steady conditions equivalent, in term of eventual equilibrium moisture content, to the variable hygrothermal conditions encountered in service. Only the relative humidity is to be defined, temperature only controls the time needed to reach the equilibrium value.

In the early stage of composite development, secondary structures or control surfaces were only involved in applications for certification. Because of the widespread concern about composite ageing at that period, material saturation (that is equilibrium in a nearly 100% RH atmosphere) was sought in certification tests. This was feasible in a short period compatible with the programme time schedule, on account of the thin laminates involved in those structure design. Later on, from the mid eighties and on, when primary structures were introduced in Europe - outstanding examples of these structures in Europe were the A310/300 vertical tail, the A320 complete tail and the ATR 72 outer wing - laminate thicknesses in the area of the main junctions increased and it was no longer possible, in tests, to reach the saturation level within an acceptable time. So, the conservative assumption used so far was questionned, and a more representative value - likely to be lower and thus easier to reach in test - value of the equivalent steady relative humidity has been sought. Unfortunately in that period, we were short of comprehensive data about actual values of composite moisture content after a long term exposure. The most useful results provided by the literature in this respect, were those of a NASA supported programme, a synthesis of which is presented in reference [5]. In this programme, coupons of mainly T300/N5209 had been outdoor exposed all around the world, including in a wet tropical region. An experiment performed by an AIRBUS partner (MBB-UT) showed that the maximum moisture content which had been observed in that NASA programme (0.8% for the T300/N5209) was equivalent, in term of equilibrium value, to a steady conditioning at 70% RH for the same material. This figure was confirmed with another material : T300/913c by the weight monitoring of coupons mounted on some Thai Airways aircraft.

On the ground of these results, for the A320 type certification it was agreed that 70% RH should be the steady condition to be applied to any involved composite for the determination of the maximum in-service moisture content.

Another figure was to be determined : the maximum laminate thickness capable of reaching the equilibrium moisture content within the expected aircraft lifetime ($\neq$ 20 years for large airplanes). Strickly speaking, this value is material dependent because controlled by its diffusivity. Still in the scope of the A320 certification, it has been assumed that all laminate thicknesses will reach the equilibrium moisture content before end of lifetime, provided they are less than 8 mm when both faces are exposed, or 4 mm if only one face is exposed.

The latest evolution

Recent results published by CEAT question the approach which assumes that every composite material exposed to the same actual environment comply, in term of moisture uptake, to the same RH steady condition. Two materials widely used in European applications : the T300/N5208 from Narmco, and the T300/914 from Ciba Geigy, reputed to be very different regarding their sensitivity to water absorption have been simultaneously exposed to outdoor conditions in Toulouse where it is fairly wet (annual RH average value : 80%). Weight monitoring was performed by drying the coupons after exposure in order to discard the influence of surface erosion on the data. Results, up to five years, are shown figure 7. Surprisingly, despite what we knew about their respective moisture sensitivity, both materials have so far absorbed more or less the same amount of water (0.9%). Figure 8 shows the values analytically predicted (according to Fick's law) for both materials, from their characteristics and from the meteorological records during that period. Results show that theoretical values after 5 years are respectively 1.37 for the T300/914 and 0.83 for the T300/N5208. From the same analysis, the equilibrium level at 70% RH would be respectively 1.1% for the T300/914 and 0.69% for the T300/N5208. All these results are discussed in detail in reference [6]. From these results, directly extending the steady condition (70% RH) used for the A320 to the A330/340 certification has been questionned, and now the new requirement for that programme is 85% RH, unless results from long term exposure are available for the concerned material. This new figure is already commonly used in Europe for composite fighter structures.

According to reference [3],FAA also recommends a steady condition for RH which corresponds to the 95th percentile high monthly average relative humidity which might occur in the hot humid climates of the world.

		After 2 years 1/2	After 5 years
MATERIAL	T300 N5208	**0.68%**	**0.90%**
	T300 Ciba 914	**0.71%**	**0.88%**

Figure 7 - Moisture content measured by drying

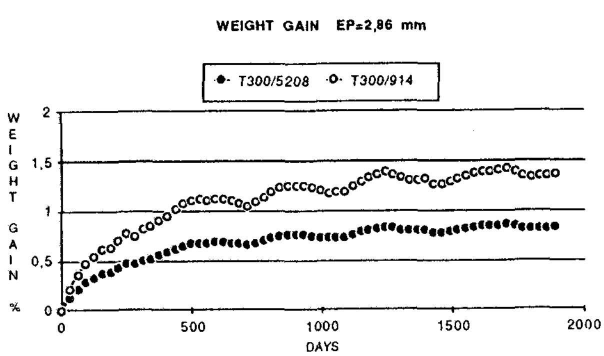

Figure 8 - Moisture uptake predicted by analysis

V - Damage tolerance demonstration for accidental impact damage

Developed first with the in-service behaviour of metallic structure background, damage tolerance analysis must take three sources of damage into account:

- fatigue
- corrosion
- accidental.

From the experience now gained with composite structures, no additional damage source specific to these materials has been found, but the relative importance of each of them must be changed.

Fatigue damage

It is well known that composite materials with thermoset resin systems are insensitive to fatigue. More exactly, their mechanical properties are such that static strength requirements normally cover fatigue strength ones. This is true for a laminate submitted to in-plane stresses, but still remains to be fully demonstrated for an actual structure with a complex design where locally, out-of-plane stresses prone to the development of interlaminar cracks, may be developed : e.g. stringer runnouts, areas of sharp thickness variation, edge delaminations, etc...

Except for a previously substantiated design, fatigue substantiations supported by a full-scale test, are still required in Europe for any new structure. The test article should display the minimum quality according to the control specification. In other words, the specimen should be provided with artificial manufacturing defects representative of their maximum allowed sizes. These defects simulating porosities, delaminations, should be judiciously located in areas of concern such as those above mentioned.

Even if the growth rate and the critical size of fatigue damage in composites was compatible with a damage tolerance substantiation based on the slow growth concept, this concept would not be practical as long as analytical tools, able to predict the phenomenon, are not available. Therefore a safe life substantiation showing on the one hand the no-growth of maximum tolerable size of defects and damage, and on the other hand the non-initiation of fatigue damage, can be accepted. It is currently favoured to cover variability by a factor on load rather than a factor on life.

Corrosion

Carbon fibres reinforced plastics are as such insensitive to corrosion. However, they could form with metallic parts in direct contact with them a galvanic couple able to develop corrosion in metals. Special care should be taken to prevent the phenomenon by adequate insulation (sealant, glass fibers). This is to be taken into account in the preparation of the maintenance manual.

Accidental damage

Among the three sources of damage to be considered, accidental damage is the one most likely to perceptibly reduce the strength capability of a composite structure. Effects of low velocity impacts on composites have been widely described in the literature. It is recognized that if the impactor (e.g. dropping tools) is blunt enough, large static strength reductions in compressively loaded structures can be expected without a visible indication of the damage. The typical shape of ultimate static strain versus impact energy curves shows a nearly horizontal asymptotic branch therefore, the limitation of the strain level in areas of concern is the most important precaution to be taken at the design level to comply with damage tolerance requirements. Experience has shown that this maximum strain level not only depends on the material itself, but on the design (laminate thickness, lay-up, boundary conditions). So, ultimate strain typical values cannot be recommended, and test data are needed for any new concept or material.

Damage tolerance demonstration

As far as accidental sources are concerned, damage tolerance substantiation of composite structures relies firstly upon a precaution at the design level (strain limitation) and secondly upon the no-growth concept. As for already mentioned fatigue damage, the slow growth concept cannot be allowed as long as analytical models able to predict damage growth and critical size are not available.

For a substantiation purpose, two categories of damage are to be differentiated :

- those which do not reduce the structure static strength below ultimate load, and thus, do not need to be detected
- those reducing the structure static strength below ultimate load, and which must be detected because flying under such condition is not permitted.

Concerning the first category, the no-growth concept (or more exactly the crack stabilization at a certain level, because the setting of the damage final size may need some load cycles after accidental impact) should be demonstrated by test.

Even if ultimate load carrying capability has already been demonstrated for such damage before fatigue (to comply with static strength requirements), the same demonstration should be performed after fatigue (cf ACJ 25 603, end of § 6-2-3 : *for the no-growth concept, residual strength testing should be performed after repeated load cycling*). In other words, it is not yet allowed to rely on NDT data alone to demonstrate the no-growth concept. Nevertheless, it is currently admitted that the ultimate load test after fatigue can take place after type certification. If performed before that time, it can be used to comply with static strength requirements. Such a procedure has already been used by AIRBUS INDUSTRIE for the A310/300 and A320 vertical fin, where only one component test was planned for all substantiations at the full-scale level.

The number of flights to be simulated to validate the no-growth concept should be statistically significant (cf ACJ 25 603 § 6-2-2). However, demonstrating this concept for existing impact damage and the safe life of the design as regards pure fatigue damage call for two different phenomenons. 'More' flights should be normally required for safe life (durability) demonstration. The safety factors to cover the scatter should be derived from tests at the coupon level investigating the same phenomenon, either the growth of existing cracks for the validation of the no-growth concept, or the initiation of fatigue cracks for the durability demonstration.

The case of impact damage reducing static strength capability below ultimate load, but above limit load

If their no-growth can be demonstrated, this damage will never reach a critical size, that is the size corresponding to the limit load carrying capability, according to the usual damage tolerance concept applied to metallic structures. In other words, whatever the scheduled inspection programme, this damage will be implicitly detected before its critical size, and the basic requirement of the damage tolerance will be automatically met. The graph fig. 9 drawn from reference [7] shows that sticking to such a principle could lead to composite structures less safe than metallic ones, an evolution obviously hardly acceptable.

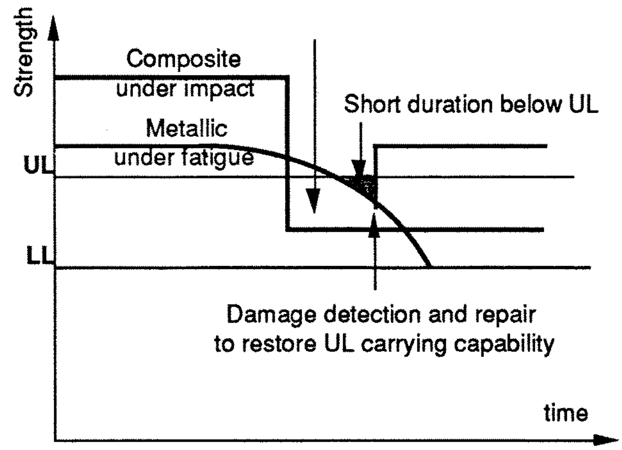

Figure 9 - Damage tolerance philosophy

A scheduled inspection programme should therefore be established and, as it can be read in ACJ 25-603 : *In selecting the intervals, the residual strength level associated with the assumed damage should be considered .*

Selection of inspection intervals.

Basically, applicable means of compliance provided by ACJ 25 603, require that the larger the strength reduction below ultimate load is, the sooner the damage should be detected. Nevertheless, accidental damage is a chancy phenomenon to which a probability of occurrence can be associated, depending on the structure part involved. For instance, the inner skin of a flap is more likely to be struck by a runway debris than one of the vertical fin skins. Consequently, not only should the static strength reduction associated with the assumed damage be considered, but its probability of occurrence as well. In other words, for a given static strength reduction below ultimate loads, the more likely damage may occur, the sooner it should be detected.

The need for that probabilistic approach had not been realized when the latest issue of the engineering advisory circular was prepared, but later it was required for the certification of the A320 composite parts and the ATR72 outer wing box.

Airworthiness authorities involved in the A320 joint certification proposed that the issue should be addressed by analogy with the failure of a system interacting with structural performances as the load alleviation system (LAS) of the A320. The LAS is a new improvement introduced in the A320, in which a computerized system provides alleviation of the wing bending moment induced by a gust, through the rapid coordinated movement in the same direction of the ailerons and the outer spoilers. In case of a failure of that system, the load carrying capability is subsequently reduced until the failure is detected and the system repaired, as it is the case for accidental damage on a composite structure.

If the strength reduction leads the structure to a load carrying capability below ultimate loads and because flying is not allowed under such conditions, an inspection programme able to detect the failure of the system (or by analogy accidental impact damage) should be established.

Tolerable reductions of the static strength safety factor versus the probability of flying under failed conditions have then been agreed for the certification. One approach consisted in establishing a relationship between the safety factor to be validated and a parameter called 'proportion of flight time' which is equal to the probability of failure occurrence per hour multiplied by the average time in failed condition which has been arbitrarily considered equal to half an interval of inspection.

Figure 10 drawn from reference [8] illustrates the use of the probabilistic approach based on the parameter 'proportion of flight time' to the selection of inspection intervals.

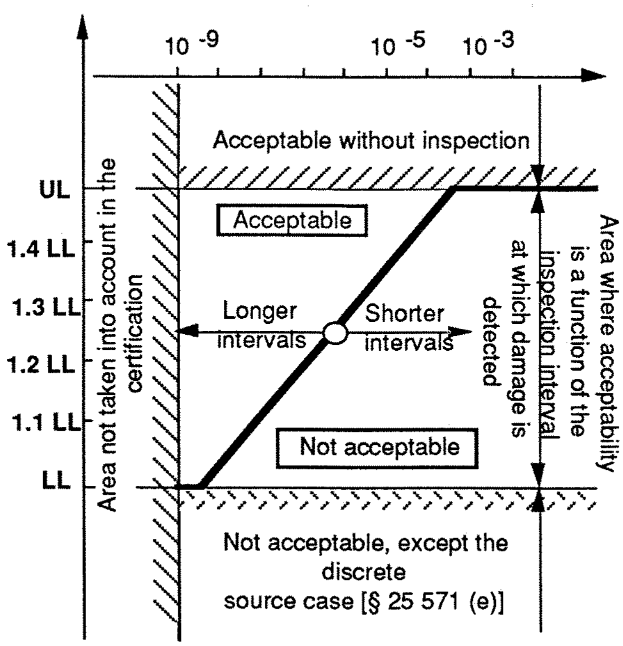

Figure 10 - Illustration of a probabilistic approach

The probability of accidental damage per flight hour is represented on the upper horizontal axis, and the safety factor on the vertical one. Only the area where both probability of damage occurrence is higher than 10E-9 and residual static strength is below ultimate loads is to be considered for this approach. A moveable segment whose position depends on the inspection interval expressed in hours splits this area into two zones : the acceptable one on the left hand side, and the non-acceptable one on the right hand one. According to the duration of the inspection interval, this segment moves parallel to itself, leftwards for longer intervals, rightwards for shorter ones.

The position of that segment is such that the probability of having the combination of a damage reducing the structure strength down to k.LL and encountering a gust of the same intensity, is extremely remote (≤ 10 E-9). A log/linear relationship between probability of gust occurrence and intensity is assumed between LL and UL.

Obviously, to implement such an approach, a comprehensive data base about the probability of impact damage on a structure, allowing for the component involved, the aircraft operating conditions, etc... should be available. Everybody, mainly airlines, is then requested to contribute to the constitution of that data base. Because that background was not available in time for the A320 type certification, this method could only be applied in a very conservative way, where all accidental damage not immediately detectable is assumed not to reduce static strength below ultimate loads. More recently, a probabilistic approach has been submitted to approval for the ATR 72. Such an approach is also expected for the A330/340 programme.

Limitations in impact damage to be assumed

The impact severity to be assumed in structural substantiations is limited by two parameters : the threshold of detectability, and the realistic level of energy. In fact, two thresholds must be considered for detectability (figure 11). The lower one corresponds to the acceptability level according to the selected inspection procedure : all damage up to this level is demonstrated to be able to withstand UL before and after fatigue. The higher one corresponds to immediately detectable damage where only limit load carrying capability is required. Damage sizes ranging between these two thresholds can only be detected within the scheduled inspection programme and are then subjected to the above described procedure for interval selection.

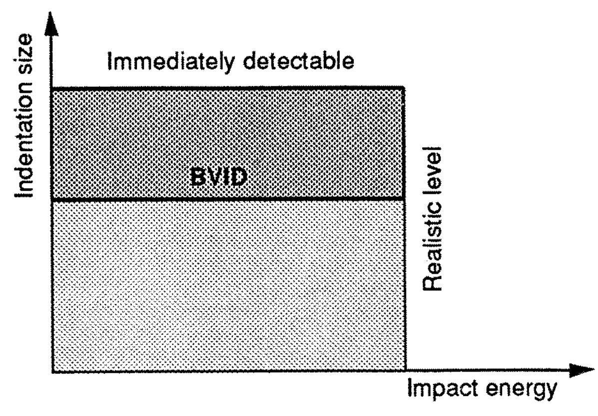

Figure 11 - Damage severity limitations

An agreement exists to select the Barely Visible Impact Damage (BVID) as the first threshold of detectability. The BVID corresponds to the indentation size detectable at the 'detailed' inspection level, which means the most favourable conditions for a visual inspection.

It is recognized that most of the compression strength reduction due to accidental impact occurs before BVID is reached. consequently, if that threshold of detectability is chosen to demonstrate UL capability before and after fatigue, the additional strength reduction due to a more severe damage should not lead to a residual strength far below ultimate loads.

As standardized minimum fatigue crack lengths, visually detectable, are now defined for metallic parts (cf MSG3 procedures) It is urgently needed to find an agreement about what should be the detectable size of an indentation. This size should correspond to the maximum size that might escape the common inspector notice, and not the minimum size that can be noticed by the most skillful of them.

Defining the energy cut-offs is even more subjective. Various possible scenarios of accidental impact during composite manufacturing (handling and assembling), aircraft maintenance or servicing must be investigated. Energy levels of dropping tools, contacts with rigs in realistic situations, should be calculated. Values ranging from 32 to 140 joules have been determined in the case of the ATR72 and the A320 type certification.

Supporting tests for validation

As already mentioned, it is still a widespread practice in Europe to perform a full-scale fatige test in the certification programme of a large airplane composite structure. Apart from substantiating the metal parts of this structure if any, the purpose of this test is twofold : i) to demonstrate the durability of the composite design, that is no evidence of crack initiation in fatigue sensitive areas (zones where there may be out-of-plane stresses are still suspected to be such), ii) to validate the no-growth concept for accidental impact damage. The fatigue test programme is devided into two successive sequences. The first one aimed at the demonstration of the durability and at the no-growth of tolerable impact damage. Damage severity introduced in the test article is represented by the light grey zone of the figure 11. One lifetime with a safety factor (commonly a load/life factor) is simulated in this first sequence, ultimate load carrying capability in the most adverse conditions is demonstrated at its end. In the second sequence, damages whose severity is represented by the dark grey zone of the figure 11 are added in the structure. Inspection intervals are simulated with a safety factor, then, a certain load carrying capability - between UL and LL - depending on the probability of damage occurrence and the inspection interval is demonstrated. For all the test, load cycling is performed at room temperature, with an 'aged' structure, that is having absorbed 60% of the maximum expected moisture content at least.

VI - Conclusion

From the experience gained with the latest AIRBUS and ATR aircraft, this paper has reviewed three selected key issues in the certification procedures used for large airplane composite structures, where recent noteworthy advances have been observed in Europe.

- A compliance philosophy has been developed to qualify a second source material for an already certified part.
- A recommended steady condition for the relative humidity equivalent, in term of material equilibrium moisture content, to the hygrothermal history encountered by the aircraft, is finalized.
- The need for a probalistic approach to select the inspection intervals for accidental impacts in damage tolerance justifications, has been shown.

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