# FATIGUE AND FLAW TOLERANCE DESIGN APPROACH FOR HELICOPTER AIRFRAME

**N. Galletti[1], P.Monticelli[2], and M. Tirelli[3]**

[1] AgustaWestland
Via G. Agusta 520, 21017 Cascina Costa di Samarate, Varese Italy
e-mail: n.galletti@it.agusta.com

[2] AgustaWestland
Via G. Agusta 520, 21017 Cascina Costa di Samarate, Varese Italy
e-mail: p.monticellli@it.agusta.com

[3] AgustaWestland
Via G. Agusta 520, 21017 Cascina Costa di Samarate, Varese Italy
e-mail: m.tirelli@it.agusta.com

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**Abstract:** The fuselage structure is classified into various parts according to the consequences of their failure to rotorcraft safety (FAR 29.571 / JAR 29.571). By the way are classified as "Primary Structural Elements (PSE)" the critical parts the failure of which could have a catastrophic effect upon the Rotorcraft.

Fatigue substantiation accomplishes by a combination of safe life and flaw tolerance approaches, supplemented by fail safe capabilities wherever applicable.

Through the safe life approach the retirement life is estimated for all of the Primary Structural Elements. For this reason the safe life is the capability of a pristine structure to withstand the operating loads without detectable cracks during the service life of the helicopter or during an established replacement time.
For the safe life evaluation the fundamental factors taken into account are the fatigue strength of the component (by test), the magnitude of loads or stresses in service (by flight load survey) and the operational usage spectra and loading frequencies (measured in operating ranges of the helicopter).

In addition each fatigue critical part has to be protected from realistic flaws and damages that could occur during the manufacturing, assembly or maintenance operation plus service usage. This is done by Flaw Tolerance and/or demonstrating Fail Safe capabilities in order to obtain the applicable inspection intervals for each critical part.
The fail safe design, generally recognized as flaw tolerance, is the capability of the helicopter structure to continue functioning without catastrophic failure after being subjected to fatigue damage, intrinsic flaws or accidental damage.

# 1 INTRODUCTION

The paper describes the philosophy for performing the Fatigue and Flaw Tolerant analysis of a helicopter airframe component in order to evaluate the fatigue strength and to provide the necessary data to the Fatigue Department for the definition of the fatigue lives and threshold inspections.
The guidelines illustrated are reported in accordance with J.A.R./F.A.R. 29.571 philosophy that was followed to achieve the certification of the AgustaWestland Helicopter.

## 1.1 General Philosophy for Fatigue Evaluation (durability and damage tolerance)

Durability evaluation of helicopter airframe structure is performed in accordance with JAR/FAR 29.571. The Safe Life (Fatigue) approach is defined as the property of a structure to meet the design goal life with out any detectable flaws / cracks during its service life.
The Flaw Tolerant approach (or Damage Tolerance defined as the property of a structure to sustain flaws/cracks safely) is considering the effect of detectable flaws / cracks on the structure fatigue capability (with or without flaw growth).
Metallic structures are evaluated using Linear Elastic Fracture Mechanics (LEFM). Composite structure is generally evaluated for damage tolerance by test (considering manufacturing and in-service damage). In general, testing will be performed on environmentally conditioned sub-components or specimens with imbedded defects and applied impact damage.
In general, an evaluation of the structure, under typical load and environmental spectra, must show that catastrophic failure due to fatigue, corrosion or accidental damage, will be avoided throughout the operational life of the rotorcraft. This can only be assured with an adequate inspection program and therefore the evaluation must result in inspection and maintenance procedures for each PSE whose failure, if remained undetected, would lead to loss of the rotorcraft.

The general philosophy is summarized in the following flowchart.

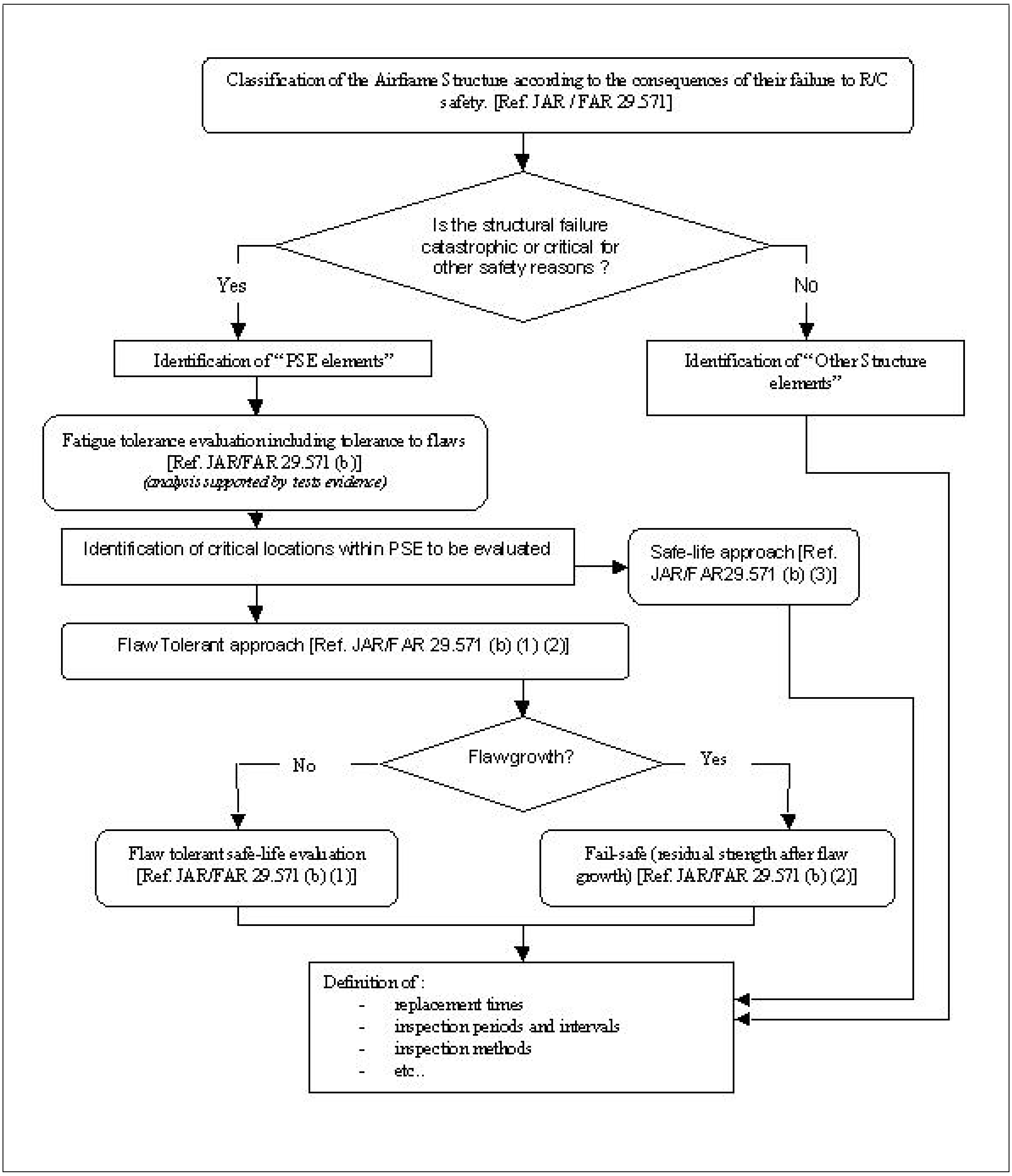


*Figure 1: General Philosophy for durability and damage tolerance evaluation based on Civil Certification (JAR/FAR)*

## 1.2 Airframe Structural Classification

Structure is classified into various parts according to the consequences of their failure to rotorcraft (R/C ) safety (Ref. [2], FAR 29.571 / JAR 29.571). A "Primary Structural Element (PSE)" is a critical part, the failure of which could have a catastrophic effect upon the R/C, and/or for which characteristics have been identified as critical for other safety reasons.
A PSE is any detail, element or assembly which significantly contributes to carrying flight, ground, pressure or control loads. PSE's are classified into three different levels of criticality as a function of the following:

- consequences of their failure;
- experience of the loading environment (high or low vibration, GAG and manoeuvre);
- stress level;
- in service failures of similar design structure;
- type of load path ( single or multiple load path);
- locations where flaw detection would be difficult;
- items/areas vulnerable to accidental damage;
- corrosion sensitive areas.

"Other Structure (OS)" is composed of those parts judged not to be a PSE.
Damage to these parts will not affect the structural integrity and the safety of the R/C (they may be omitted for structural strength and without them the R/C may be flown safely).

PSE's are classified as a function of their level of criticality and normally three different levels are defined:

Level 1: PSE's whose failure is critical for the R/C.
These PSE's include the areas where highest stresses due to flight and ground loads are expected, and where high cycle fatigue loading is experienced. They coincide substantially with the supporting structures of Main / Tail Rotor, Engines, and Tail Plane.
Fatigue Design goal criteria applied to these PSE's is "Small Damage with No Flaw Growth" [Flaw Tolerant Safe Life evaluation, FAR / JAR 29.571 (b), (1)].

Level 2: PSE's whose failure will reduce the safety level of the R/C.
These PSE's include the areas where significant flight and ground loads are expected.
Fatigue Design Goal criteria applied to these PSE's is "Small Damage with possible Slow Flaw Growth" [Fail Safe (residual strength after flaw growth) evaluation, FAR / JAR 29.571 (b), (2)].

Level 3: PSE's whose failure has a minor impact on structural integrity and safety level of the R/C.
These PSE's include the areas with low stress level and showing generally "Multiple Load Path" design.
Fatigue Design goal criteria applied to these PSE's is "Large Damage with possible Slow Flaw Growth" [Fail Safe (residual strength after flaw growth) evaluation, FAR / JAR 29.571 (b), (2)].

### 1.3 General Methodology and Related Assumption for Durability Analysis.

The durability evaluation must show that catastrophic failure due to fatigue considering the effects of intrinsic/ discrete flaws, or accidental damage will be avoided.

The fatigue evaluation is carried out using the following approach:
Safe-Life [see Ref. [2] : FAR/JAR 29.571 (b) (3) ] ;
For the Safe-Life approach, it must be shown that the pristine structure (without detectable flaws/cracks) is able to withstand fatigue loading without detectable cracks for the following time intervals:

- Life of the helicopter
- Within an established replacement time.

Life estimation for Safe-Life approach can be calculated using conventional fatigue analysis techniques incorporating stress distribution, stress concentration factors, S-N curves with appropriate reduction factors. The life will be calculated using the Miner's cumulative damage approach.

The general methodology and the related main assumptions to cover the Fatigue Design objectives are highlighted in the next pages.

<u>Selection of critical locations</u>
When selecting the critical locations to be analysed within a PSE for Fatigue Analysis, the following should be considered:

- Points of high stress concentration;
- Locations where static analyses shows low Margins of Safety;
- Locations of potential fatigue damage;
- Locations where inspection is difficult;
- Design details that service experience of similarly designed components indicates are prone to fatigue and other damages;
- Components fabricated from materials of potentially poor fatigue behaviour;
- Areas of probable damage from sources such as manufacturing, severe corrosive and/or fretting environment, maintenance, wear, etc
- Locations with high load spectrum severity (i.e. subjected to Rotor Induced Vibratory Loads).

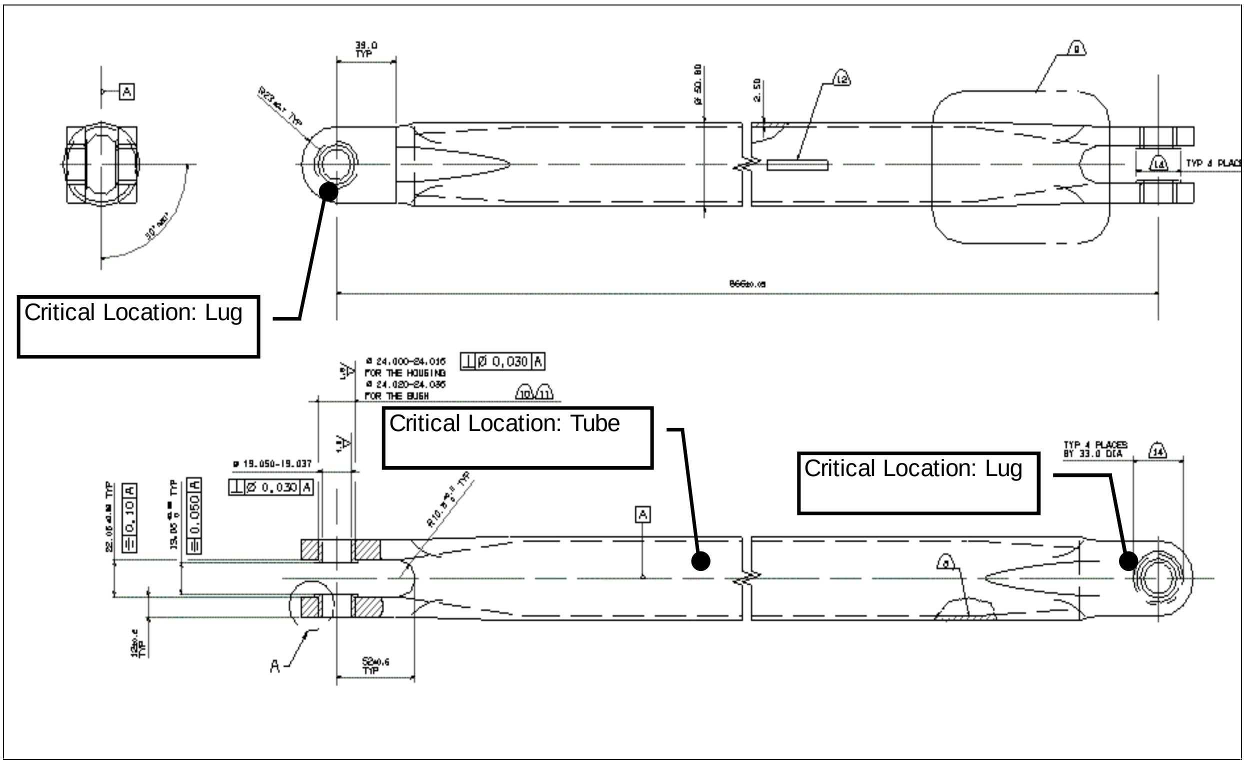


*Figure 2: Selection of Critical Locations*

Experience gained from past service history of similar structure shall be used in selecting locations for analysis.

Definition of Fatigue Spectrum :
The most critical location could correspond to the zone where the maximum principal stress ($\sigma_{Max\ Princ}$) is highest. The analysis fatigue spectrum at this location (for avoiding any further conservatism) should be built as follows :

- Identify the load condition showing the critical $\sigma_{MaxPrinc}$ (highest tensile stress value) and the related location;
- Calculate at this location the principal stresses for all other load cases in the fatigue spectrum.
- Alternatively if a more refined analysis is required calculate at this location the applied stress for all other load conditions (composing the fatigue spectrum) in accordance with the direction defined by $\sigma_{Max\ Princ}$ ;

Calculation of Stress Concentration Factor :
Use Ref.[3]

Calculation of Stress History :

As Fatigue analysis is based upon net section stress , the Fatigue Spectrum, whatever its basis, must be converted into stresses;

The transfer functions to transform loads into stresses to generate the applicable Stress History at a certain location, should be calculated avoiding any unnecessary conservatisms such as:

- Peak stresses from Global or Detailed FEM analysis must have realistic stress concentration effect, and any additional stress concentration factors should not be used.
- Calculate the correct reference net section stress before including the stress concentration factors.
- The reference stress is normally based on nominal dimensions. The load distribution on symmetric lugs is assumed to be 50%-50%; only for asymmetric lugs (asymmetry might be due to different thickness and diameter and /or large tolerances) a more detailed analysis should performed to evaluate the actual load distribution on lugs;

Material Database:

Material data are the taken from Ref. [7].

Inspections During Service Life

If inspections for fatigue cracks are to be carried out during the service life of the component, the following size cracks can be expected to be detected. Just for information, the following field detectable crack sizes are suggested with a probability of detection of 95% and confidence level of 90%:

- Magnetic particle 3.60 mm
- Dye penetrant 4.00 mm
- Ultrasonic 6.00 mm
- Eddy current 6.00 mm
- Visual inspection on glossy paint 15.0 mm
- Visual inspection on dull paint 30.0 mm

Calculation of Fatigue Life

- This calculation is performed using an dedicated software which uses the four parameter equation to define the SN curve and uses the fatigue stress spectrum (net section stresses) including any stress concentration factors to calculate the Fatigue Damage using Miner's Rule. The damage due to each load phase/cycle in the spectrum is summed up. The Fatigue Life is equal to the spectrum flight Hours / total fatigue damage.
- If the fatigue damage is >1.0, then the component has insufficient fatigue life and obviously does not meet the fatigue life goal requirements and the design has to be improved.

Definition of Required Inspections and Retirement Life :

The definition of the Required Inspections and the Retirement Life is a Fatigue Department responsibility. The information that must be provided to the Fatigue Department is normally the results of the fatigue analysis.

## 1.4 General Methodology and Related Assumption for Damage Tolerance Analysis.

The Damage Tolerance analysis has to determine mainly :

- the effect of the cracks on the static strength;
- the crack growth (if any) as a function of time.

The general methodology and the related main assumptions to cover these two Damage Tolerance objectives are highlighted in the following pages.

<u>Selection of critical locations</u>

When selecting the critical locations of cracks / flaws to be analysed within a PSE for Damage Tolerance, the following should be considered:

- Points of high stress concentration;
- Locations where static analyses shows low Margins of Safety;
- Locations of potential fatigue damage identified by the conventional fatigue evaluation;
- Locations where flaw detection is difficult ( no or difficult inspectability);
- Design details that service experience of similarly designed components indicates are prone to fatigue and other damages;
- Components fabricated from materials of potentially low fracture toughness or high flaw growth rate;
- Areas of probable damage from sources such as manufacturing, severe corrosive and/or fretting environment, maintenance, wear, etc
- Locations with high load spectrum severity (i.e. subjected to Rotor Induced Vibratory Loads).

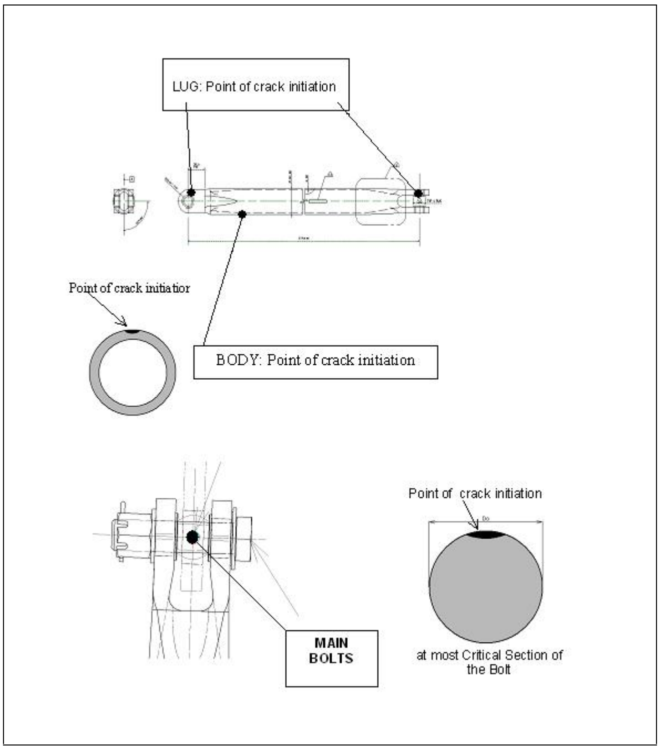


*Figure 3: Selection of Critical Locations*

Experience gained from past service history of similar structure is invaluable in selecting damage locations.
The general philosophy to be used for the flaw tolerance analysis is to consider damage at multiple locations without crack growth interaction.

Definition of initial crack sizes
The initial crack sizes (**$a_0$**) are established in accordance with JAR / FAR requirements ( Ref. [2] and [6]).
The initial crack sizes assumed to be detectable and representing a maximum which could exist as a result of abnormal manufacturing, maintenance actions or service environment.

There is a certain crack size below which detection is almost impossible. The basic approach is to establish the extent of **$a_0$** in terms of detectability in accordance with the inspection techniques used.

Initial cracks are assumed to be present from the beginning of the rotorcraft life.

Definition of Fatigue Spectrum :
Crack growth retardation due to high tension loads and acceleration due to low compression are not accounted. If retardation is not an issue and the loading history is essentially random ( normally based on 100 Flight Hours block), sequencing of loads/stresses of the fatigue spectrum is rather irrelevant (see § 6.5 of Ref. [1]). In this case the Fatigue crack growth is virtually insensitive to the loads sequence.

If a structural component is subjected to a tension-compression cycle (i.e. $R < 0.0$) the crack closes during the compression cycle; the compressive part of the cycle has no effect upon flaw propagation but it modifies the da/dN curve decreasing the value of threshold stress intensity factor as shown in following Figure 2. For this reason the actual stress ratio $R= \sigma_{MIN} / \sigma_{MAX}$ corresponding to the applied loading should be used (it is automatically done so by AFGROW/NASGRO program, *Ref. [4] and [5]*). This approach is valid for all the structural details apart from lugs where the compressive part of the Fatigue Spectrum should be removed. Lugs are a “unique” configuration based on test results showing that the compressive part of the cycle can be ignored and assume a cut-off R value equal to 0.0.

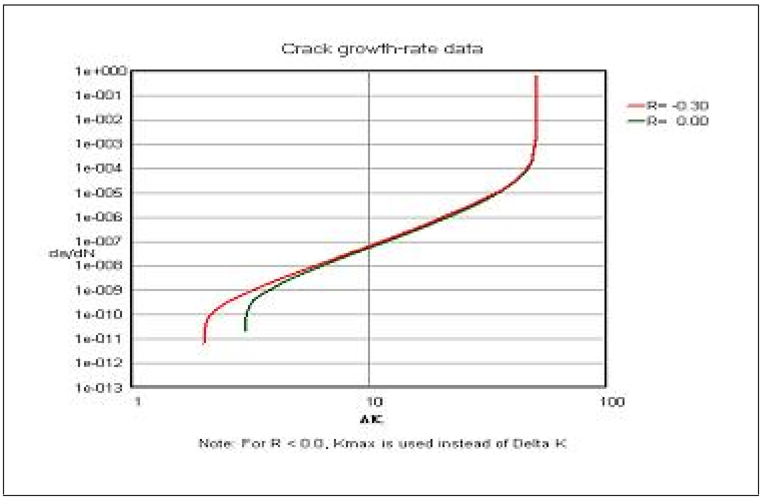


*Figure 4: Example of da/dN curve as function of R value*

The most critical location could correspond to the zone where the maximum principal stress ($\sigma_{Max\ Princ}$) is highest. The crack growth analysis fatigue spectrum at this location (for avoiding any further conservatism) should be built as follows :

- Identify the load condition showing the critical $\sigma_{MaxPrinc}$ (highest tension stress value) and the related location;

- Calculate at this location the principal stresses for all other load cases in the fatigue spectrum, assuming the crack growing perpendicular to the direction of the highest principal stress.

- Alternatively if a more refined analysis is required calculate at this location the applied stress for all other load conditions (composing the fatigue spectrum) in accordance with the direction defined by $\sigma_{Max\ Princ}$ ;

Calculation of Stress History :
As Fatigue crack growth analysis is based upon stress , the Fatigue Spectrum must be converted into stresses. The transfer functions to transform loads into stresses to generate the applicable Stress History at a certain location, should be calculated avoiding any unnecessary conservatisms such as :

- Peak stresses from Global or Detailed FEM analysis must not be used in crack growth analysis;
- Calculate the correct reference stress consistent with the Fracture Mechanics model used for crack growth analysis (see Ref. [4] and [5]);
- The reference stress is normally based on nominal dimensions.

Material Database:
Material data are the taken from Ref. [4] and Ref. [5]. For Threshold Intensity Factor calculation the primary source is the AgustaWestland Data Base; for those materials not covered by this report use the material data contained in AFGROW (Ref. [4]) and NASGRO (Ref. [5]) database after the Fatigue Department approval only;

Calculation of Critical Crack Length:
Critical crack length must be calculated in accordance with JAR/FAR applicable requirements where for justifying the "Residual Strength after Flaw Growth" it is required to show that the structure remaining is able to withstand Static Design Limit Loads without failure [JAR 29.571 (b) (2) ]. Thus it is clearly stated that the calculation of Critical Crack Length must be based on the maximum Static Design Limit Load and not on the maximum Fatigue Spectrum Load; this means that for the crack growth analysis the following approach must be applied:

- The critical crack length calculated by the Fracture Mechanics software (AFGROW and NASGRO) is normally based on the maximum Fatigue Spectrum Load, thus it represents only an additional information that cannot be used for the Inspection Intervals definition;
- The critical crack length to be used for the Inspection Intervals definition must be based on the maximum Static Design Limit Load. This maximum Static Design Limit Load is normally derived from all Flight and Ground "Normal conditions", excluding the "Special conditions" such as Fail-safe, Crash, etc..;

When calculating the Critical Crack Length check if "reasonable " sizes are obtained: if not a redesign of the component should be performed. Just for information, the following field detectable crack sizes are suggested with a probability of detection of 95% and confidence level of 90%:

| | | |
|---|---|---|
| Magnetic particle | 3.60 | mm |
| Dye penetrant | 4.00 | mm |
| Ultrasonic | 6.00 | mm |
| Eddy current | 6.00 | mm |
| Visual inspection on glossy paint | 15.0 | mm |
| Visual inspection on dull paint | 30.0 | mm |

Calculation of $\Delta K$ for each load condition of the fatigue spectrum:
This calculation is performed automatically by the dedicated software AFGROW and NASGRO (Ref. [4] and [5]), where the predicted flaw growth is based on their own materials database. The material's data used by these programs are normally recognised by the relevant Authority and by the Fatigue Department.

Calculation of crack growth using material *da/dN* curve:
This calculation is performed automatically by the dedicated software AFGROW and NASGRO (Ref. [4] and [5]), where the crack growth is based on their own materials database. The *da/dN* curves used by these programs are normally recognised by the relevant Authority and by the Fatigue Department.
The crack growth is calculated without considering the Retardation effect and considering the Linear Elastic Fracture Mechanics approach. The majority of the materials used in the aerospace industry (ductile materials) with nominal applied Limit stresses less than about 0.80 yield strength can be analysed on the basis of elastic concepts (see Ref.[1], § 3).

Failure check based on $K_{max}$ or Net Section Yield Criteria:
This calculation is performed automatically by the dedicated software AFGROW and NAS-GRO (Ref. [4] and [5]). The calculation is stopped when either of the following is reached :

- *'$K_{max}$ Failure Criteria'* : the crack depth and length has reached a sufficient size such the stress intensity at the crack tip for the maximum load ($K_{max}$ ) has reached plain strain fracture toughness ($K_{Ic}$) or plain stress fracture toughness ($K_c$) *;*
- *'Net Section Yield Failure Criteria'*: the stress on the net section remaining has reached the yield stress (Fty);

The life (normally in terms of Flight Hours) that must be taken as reference for the Inspection Intervals definition is that based on "*$K_{max}$ Failure Criteria"* only; as mentioned in previous, the AFGROW and NASGRO calculation is based on the Fatigue Spectrum Load and not on maximum Static Design Limit Load as required by the Authority for the *'Net Section Yield Failure Criteria".*

Definition of Required Inspections and Retirement Life:
The definition of the Required Inspections and the Retirement Life is a Fatigue Department responsibility. The information that must be provided to the Fatigue Department is normally the Crack Growth (Plots and numerical format) for all locations showing flaw growth.

Hereafter the Fail Safe analysis approach to be applied to all PSE levels and Other Structure is summarised:

**Level 1 PSE**
**Design Goal:** Small Damage with No Flaw Growth
**Fail Safe Analysis Approach:** For No Growth condition Fail Safe analysis is not required
**Failure Type**: Small damage Not allowed, Large Damage Not allowed
**Remarks:** Slow Crack Growth due to small damage could be allowed only for those locations where the level of structural redundancy is such that any flaw growth can be justified easily and safely.

**Level 2 PSE**
**Design Goal:** Small Damage with No Flaw Growth or Slow Flaw Growth covered by Residual Strength
**Fail Safe Analysis Approach:** Fail Safe analysis is performed in case of Flaw Growth only
**Failure Type:** Small damage allowed, Large Damage Not allowed
**Remarks:** Large Damage could be allowed only for those locations where the level of structural redundancy is such that any flaw growth can be justified easily and safely.

**Level 3 PSE**
**Design Goal:** Large Damage with Slow or Fast Growth covered by Residual Strength
**Fail Safe Analysis Approach:** Fail Safe analysis is performed in case of Flaw Growth only
**Failure Type:** Small damage allowed, Large Damage allowed
**Remarks:** Larger crack lengths might be accepted (i.e. simulating the complete failure of secondary frames) if an appropriate analytical justification is performed.

**Other Structure**
**Design Goal:** Large Damage covered by Residual Strength
**Fail Safe Analysis Approach:** Flaw Growth analysis not requested
**Failure Type:** Small Damage not applicable, Large Damaged allowed
**Remarks:** none

## 1.5 Conclusion

For PSE level 1 items it is always the aim of the analysis to achieve “No Crack Growth” result. However crack growth can be allowed for PSE level 1 structure if it can be shown to be slow stable crack growth with a large critical crack length that is easily detectable. In addition the time to failure must be sufficiently large that economically acceptable inspection intervals can be defined.

Fatigue substantiation accomplishes by a combination of safe life and flaw tolerance approaches, supplemented by fail safe capabilities wherever applicable.

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