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ROTORCRAFT FATIGUE AND DAMAGE TOLERANCE

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1. PURPOSE

This paper on rotorcraft fatigue and damage tolerance has been independently prepared at the request of the Technical Oversight Group on Aging Aircraft (TOGAA). The TOGAA mission is to review aging-related safety issues and make recommendations to implement corrective action. As part of this mission, the TOGAA has expressed concerns regarding the current FAR 29.571 (Fatigue Evaluation) and the associated advisory circular. They therefore requested that the industry provide them with a paper on fatigue and damage tolerance that would describe current and evolving practices within the international rotorcraft industry and perhaps influence future regulatory changes. Representatives from the major helicopter companies in the United States and Europe were appointed to a Rotorcraft Working Group (RWG) to facilitate communication with the TOGAA. This paper provides a description of industry practices and recommendations on fatigue and damage tolerance certification for rotorcraft structure, focusing on metals. It is recommended that an industry paper on composites be prepared in the future, perhaps using the 1985 industry AHS paper (Ref. 1) as a beginning point.

2. INTRODUCTION

Since the introduction of the helicopter to the civil market in the 1940's, there have been many technology developments within the rotorcraft industry related to structures and structural fatigue. The primary emphasis was to refine the safe-life process. In more recent times, the rotorcraft industry has been transitioning to damage/flaw tolerance. This trend will continue as new design features, fabrication techniques, and innovative approaches evolve. It will be necessary to continually review regulatory requirements and procedures. The most recent example of this process is the incorporation of damage/flaw tolerance requirements into FAR Part 29 at Amendment 28. Since the TOGAA has expressed concerns about the current FAR 29.571 and the associated advisory material, this paper has been prepared by the industry RWG to document industry practices and present recommendations on fatigue and damage tolerance certification which could influence future regulatory material. A roadmap of anticipated industry activity relative to fatigue and damage tolerance methodology is presented in Section 8.

Under the current FAR 29.571 and the recommendations of this paper, adherence to damage/flaw tolerant design and certification is required unless complications such as limitations in geometry, inspectability, or good design practice renders them impractical. Here good design practice includes consideration of component complexity, component weight, production methods, and component cost. Under these circumstances, a design that complies with the conventional rotorcraft safe-life design and certification requirements may be used. Typical examples of rotorcraft structure that might not be conducive to damage/flaw tolerant design are swashplates, main rotor shafts, push rods, small rotor head components (devices, bolts, etc.), landing gear, and gearbox internal parts including bearings (Ref. 2). The options for the fatigue methodology to be used and the selection criterion for each are as follows:

a. <u>Safe Life (SL)</u>
   - Use if DT or FT is limited by geometry, inspectability, or good design practice.
   - Miner's Rule used to retire component prior to crack initiation (*S–N*) for as-manufactured component.
   - No special inspection required.

b. <u>Damage Tolerance (DT)</u>
   - Based on fracture mechanics principles.
   - Inspection interval set based on crack propagation (da/dN, ΔK).
   - No special inspection for no/benign crack growth.
   - Component retirement can be based on durability/*S–N* approach.

c. <u>Flaw Tolerance (FT)</u>
   - Miner's Rule used to set inspection/retirement prior to crack initiation from clearly detectable flaws (dents, scratches, corrosion, etc.).
   - Inspection is for flaws based on *S–N* testing of flawed component.
   - Component can be inspected for flaws and returned to service if none found or repaired if flaws found.
   - Component can be retired based on crack initiation from barely detectable flaws.

In order to achieve the design objectives relative to fatigue and damage/flaw tolerance certification, a "building block" approach is recommended that involves analysis and coupon/element/full-scale testing. The appropriate combination will depend on factors such as the criticality of the structure, complexity of the structure and load path, and whether the structure is redundant.

To meet the design objectives of fatigue evaluation, an adequate test background must exist in the form of coupon, element, and/or full-scale data. This must include tests related to damage/flaw tolerance for design information and guidance purposes. The location, growth, and detection criteria for damage or flaws are part of this information data base, and must be considered when establishing an effective inspection program.

Unless it is determined from results of stress analyses, static and fatigue testing, load surveys, and service experience that normal operating loads or stresses are sufficiently low so as to preclude initiation of fatigue or serious damage growth, repeated load analyses and/or tests should be conducted. The structure should be representative of the component being evaluated. In addition, test fixtures should support the structure in such a way that the load paths are not altered and the boundary conditions are representative of the installed component. Any method used in the analyses should be supported when necessary by test or service experience.

Flight loads and usage (Section 4) is the common thread in all the fatigue and damage/flaw tolerance methodology. Whether calculating a safe-life or inspection interval, a complete and representative flight loads data base is essential. The usage of the helicopter is equally important. Knowledge of the types of missions and the mission characteristics are required to attain conservative lives or inspection intervals.

Damage tolerant or flaw tolerant design and certification (Sections 5 and 6) is required unless complications such as limitations in geometry, inspectability, or good design practice renders them impractical. In the damage tolerant approach, the damage is defined as a crack, and the structural behavior is characterized by fracture mechanics methods and fatigue crack growth analysis and testing. In flaw tolerance, the damage is defined as an intrinsic or induced flaw from which time is required to initiate a fully developed propagating crack. Such structure is characterized by the initiation time for a crack to

This is an abridged version of a document prepared for TOGAA. To request the unabridged version, email one of the following: bdickson@bellhelicopter.textron.com or u.mariani@agusta.it develop and is characterized by crack-initiation analysis and testing.

Conventional rotorcraft safe-life design and certification (Section 7) is applicable to structure in which damage/flaw tolerance is impractical. It is generally based on load or stress versus cycles (S/N) test data and Miner's cumulative damage analysis with sufficient factors of safety to provide a safe replacement time for the structure. A combination of damage/flaw tolerance and safe-life may be appropriate for some structures.

A few comments concerning the inclusion of flaw tolerant design and certification in the paper are in order at this point. There has been an ongoing debate both within the industry member working group and between the industry and the TOGAA concerning use of this method. Some of the industry members would not choose to use the flaw tolerant method. However, a majority of industry members feel strongly that it should be retained as a method for the immediate future with an orderly transition away from flaw tolerance (FT) and towards damage tolerance (DT), if warranted by experience. The proponents of FT believe that until the industry has come up to speed on DT, FT should be retained as an acceptable alternate. Other industry members feel strongly that it should be retained indefinitely. Since the industry working group agreed at the outset to include all viewpoints in the paper, FT has been included.

Close adherence to the procedures and methodology presented in this paper are recommended. However it is recognized that in such a complex field, individual company procedures and design/fabrication will require some variations. For example, information presented in this paper on topics such as factors of safety, crack sizes, and flaw types and sizes, are provided as general guidelines. Each manufacturer must decide what is appropriate based on quality assurance procedures, manufacturing capabilities, and history. Of course, specific information on these topics should be developed in a logical way and be accepted by the certifying agency (e.g., the FAA) before proceeding.

3. DEFINITIONS

Most of the definitions used in this paper have been extracted from Reference 3 with some editing to make them more accurately fit rotorcraft requirements. Definitions with specific application to rotorcraft have been defined in the sections of the report where they are used.

4. USAGE AND LOADS

The fatigue spectrum used for each principal structural element is based on the usage spectrum of the helicopter being certified and measured flight loads. This section describes how each of these is determined or measured. The resulting loading spectra for each PSE are used in both the damage/flaw tolerance and safe-life analyses. Multiple missions may be considered in the calculation of inspection intervals and retirement lives to account for severe usages (e.g., sling load operations).

4.1 USAGE SPECTRUM

Strength, loads, and usage are the three basic elements that go into the safe-life or damage/flaw tolerance analyses. The usage spectrum is the least known element. The loads in each PSE may be accurately measured for a particular maneuver. The number of occurrences of these maneuvers or the usage spectrum is derived from knowing or estimating the wide range of helicopter missions.

The usage spectra selected by each manufacturer are approved by the certification agency and are as severe as those expected in service considering all the different missions and mission mixes of the helicopter being certified. The loads used for each principal structural element are measured during the flight strain survey. Flight simulation loads may be used when sufficient correlation to flight test data is shown.

Due to the wide capabilities and varied usages of a helicopter (e.g., Transport, Police, EMS, heavy lift, training), many different missions, configurations, and "points-in-the-sky" maneuver conditions are considered. The usage spectrum defines the distribution of the flight conditions and maneuvers in terms of percent time or number of occurrences. The usage spectrum should conservatively and accurately represent the anticipated helicopter usage, considering all operators.

Suggested spectra are given in AC20-95 and FAA order 8110.9 (Refs. 4 and 5). Modifications to this spectrum should be made for the specific helicopter and anticipated mission profiles based on historical company data. Interviews with the operators also provide mission profile information that may be incorporated into the usage spectra. Recently, more emphasis has been placed on measuring helicopter usage information. The final spectrum often includes multiple mission profiles to cover the different users. The spectrum to be used for certification must be approved by the certifying agency.

The elements to be considered when compiling the usage spectrum include the following:

a. Speed.
b. Altitude.
c. Gross weight.
d. Landings.
e. Drive system power cycles.
f. Hover and low speed flight.
g. Autorotation.
h. Maneuvers.
i. Gust.
j. Reversals.
k. Special flight configurations and conditions.
l. Ground conditions.
m. Ground–air–ground and power/thrust cycles.
n. Environmental effects.

4.2 FLIGHT LOAD/STRAIN SURVEY

The purpose of the flight strain survey (FSS) is to measure either directly or indirectly the mean and cyclic loads for the maneuvers specified in the usage spectrum for each PSE and to demonstrate that the design limit loads are not exceeded during flight within the operating envelope of the helicopter. Data gathered during the FSS may also be used for correlation with the flight loads simulator. For some PSEs, load simulation may be used to analytically determine loads for maneuvers not flown during the FSS when sufficient correlation has been shown. Such cases may be interpolated from different weights, speeds, g, etc. This same procedure is sometimes used to predict loads in redundant airframe structure, but without necessarily correlating with flight test data. This correlation for airframe is most often accomplished using FEM techniques.

During recording of the FSS data, the recording frequency needs to be high enough to resolve all significant cyclic loads. Significant blade and drive system loading frequencies may exceed 100 Hz.

As the usage spectrum represents accurately the anticipated helicopter usage, so should the measured loads represent as accurately as possible the anticipated helicopter loads. Due to the frequency of loading, it is imperative that all significant service loads be known and accounted for. Over conservatism by including unrealistic maneuvers or maneuvers flown in an unrealistic manner should be avoided.

The data selection and reduction must account for variability. This may be accomplished in a number of ways. A number of repetitions of the same flight condition may be flown and a statistical analysis conducted to select representative loads. When there are similar flight conditions that vary in altitude, weight, or speed, the maneuver that produces the highest loads may be used for all the similar maneuvers. The third method to account for loads variability is to fly the FSS maneuvers aggressively—using more rapid or larger control inputs, delaying recovery from the initial control input, using high power settings, maximum tolerance rotor imbalance and out of track, adverse cgs, worst combinations of weight/speed/altitude, etc.

All cycles within each maneuver must be accounted for. This can be accomplished by using the highest load for the entire maneuver or by cycle counting the time history data during data reduction.

4.3 SERVICE LOADS/ENVIRONMENTAL SURVEY

The amount of measured flight data from in-service helicopters is not as great as the data available for airplanes. Part of this is due to the much wider variety of usages for a helicopter and because the simple V-G-H information adequate for most airplane components will not fully describe the loads in a helicopter rotor and control system. Recorders used for helicopters can be aimed at three different goals: Maneuver Recognition, Health Monitoring, and the Flight Loads Data Recording. Recorders may overlap in these goals, but each goal is discussed below. Recorded service data may be used to modify the usage spectrum for the particular helicopter or type of operation.

4.3.1 Maneuver Recognition

This type of recorder determines the time spent in predefined flight conditions. The data would be useful for establishing usage spectra for similar helicopters that would replace or modify the original usage spectrum. The data may also be used to calculate retirement times or inspection intervals but is limited to the predefined maneuvers.

4.3.2 Health and Usage Monitoring Systems (HUMS)

The health monitoring recorder, in use for many years by the engine manufacturers and North Sea operators (Ref. 6), may be as simple as recording specific load (or any other parameter) exceedences, or may be as complex as monitoring the vibration levels of the component with an array of accelerometers. Usage tracking is now being included in some operational HUMS. The goal would be to obtain maintenance credits for dynamic component service lives and increase safety (Figure 4-1). If components remain in service for an extended time, the cost savings to the operator can significantly reduce the direct operating cost.

Engines and transmissions have characteristic vibration levels and patterns. These change when a part within the component cracks or wears excessively. The health monitor can therefore detect a potential failure before it becomes critical and warn the flight crew or maintenance personnel that replacement or overhaul of that component is required.

Other types of health monitors that have been used for over 35 years are the BIM (Blade Inspection Method) and chip detectors. The BIM is simply a pressure gage that monitors a pressurized section of the main rotor blade. If the blade cracks, the pressure is lost and a warning is given. This type of monitor is dependent upon the structure leaking before it fails. The chip detector is used in transmissions to detect the presence of metallic chips in the oil that indicate unacceptable wear in the gears, bearings, etc. The material found on the chip detector can be analyzed to determine a specific source.

Although not a true health monitor, one manufacturer has used a Cruise Guide Indicator to alert the pilot of low, medium, and high damaging flight conditions. The indicator is a display of a resultant algorithm of fixed system parameters.

All these systems are examples of monitoring the structure to either reduce the possibility of high loads, or warn the operator/pilot of required maintenance action. These systems can all be used to reduce the maintenance burden on the operator by requiring maintenance for need.

4.3.3 Loads Monitoring

A Flight Loads Data Recorder (FLDR) is an on-board recorded that monitors critical parameters. The instrumented locations may be processed in a number of ways.

a. Two or more parameters may be combined to form a resultant load parameter.
b. Fixed system parameters may be used to derive loads in the rotor system.
c. All time history data may be recorded.
d. The time history reversal points above a predetermined alternating threshold may be sequentially recorded.
e. The data may be cycle counted and stored as an array of mean and alternating cycles.
f. Fatigue damage fractions, or crack lengths (from an inspectable crack length) may be calculated directly and stored.

An FLDR typically would record all loads from switch-on to switch-off. This type of system is designed for individual aircraft tracking with little or no input required from the flight crew. Since the majority of helicopters do not experience the conservative loads used for substantiation, an FLDR has the potential to greatly increase the retirement times or inspection intervals of many components thus reducing their direct operating costs. A helicopter equipped with an FLDR that is used in a severe usage environment (i.e., logging) will also benefit from a safety of flight viewpoint by recording the rare high load that may occur.

Further development of FLDR provides the opportunity to better monitor loads in rotating system through telemetry (TM

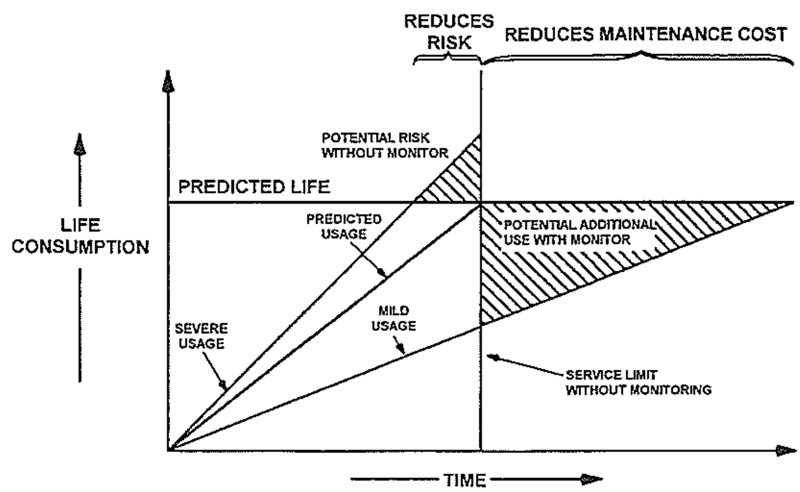


Figure 4-1. Potential benefits of usage monitoring.

from the rotating system to a location on the fixed system) or transfer functions. Both methods have had some success in the past, but more work is needed to perfect the transfer function methods and reduce the size, weight, and cost of the TM equipment.

5. DAMAGE TOLERANT DESIGN AND CERTIFICATION

Damage tolerance design and certification of structure is intended to ensure that should serious fatigue, corrosion, or accidental damage occur within the operational life of the helicopter, the remaining structure will not fail or experience excessive structural deformation until the damage is detected and repaired. The terms and approach in this section are consistent with the material presented in Refs. 3, 7, and 8.

The general requirement for a structure in this category is a maintenance of a slow growth of cracks that should not reach the critical size before being detected or before the replacement of the structure. This requirement is also met when cracks do not grow, or when the critical growth of a crack is contained to autonomous sections of the structure achieved by designing a structure as fail-safe (multiple load path structure, crack arrest structure, etc.).

The damage tolerance evaluation should encompass the following:

a. Establishing the components to be designed as damage-tolerant. This includes all principal structural elements.
b. Developing operational stress spectra for these areas based on the design usage spectra and flight loads.
c. Determining the maximum probable initial manufacturing and in-service crack sizes and the NDI detectable crack sizes for both initial and follow on in-service inspections (i.e., establishing rogue and/or detectable crack locations and sizes).
d. Determining the times to grow the initial cracks to the critical crack size and the resulting inspection intervals using fracture mechanics analysis and test.
e. Determining the service life and the resulting overhaul/replacement interval to ensure that the assumptions of the analysis (crack lengths, critical locations, environmental degradation) are not altered (e.g., widespread fatigue damage or no crack growth).

Design features that should be considered in attaining damage tolerant structure include the following:

a. Multiple load path construction and the use of crack stoppers.
b. Materials with high fracture toughness (i.e., resulting in large critical crack sizes) and with slow rates of crack propagation (i.e., low da/dN and da/dt).
c. Structural design which allows for required inspections.
d. Shielding or protective coatings and treatments that prevent and/or retard the growth of environmental and/or accidental mechanical damage.
e. Surface residual stresses such as from shot-peening, cold work, etc., that delay crack initiation and retard crack growth.
f. Provisions to prevent an occurrence of widespread fatigue damage during the service life.
g. Use of frozen planning to control the manufacturing processes.

Certification should be accomplished by a combination of analysis and supporting data from coupon/element/full-scale testing. Combination will depend on a number of factors including confidence in analysis, component criticality, structural complexity, and type of structure.

5.1 DEFINITION OF A CRACK

5.1.1 Cracks

Three types of cracks are considered:

a. Initial quality cracks that can exist as a result of normal (standard) manufacturing, maintenance, or service environment.
b. Rogue cracks are representative of the most severe crack resulting from manufacturing, maintenance, or service environment.
c. Detectable cracks can be detected during a defined inspection using the prescribed procedures and are the result of crack growth from either initial quality cracks or rogue cracks.

The assumed crack sizes for minimum quality cracks and rogue cracks vary, depending on the manufacturer, manufacturing process, usage, maintenance, material properties, etc. Typically, crack sizes in use for surface or corner cracks are depth of 0.005 inch (0.125 mm) for the initial quality cracks, and depth of 0.015 inch (0.380 mm) for the rogue cracks.

The size of a fatigue crack that can be detected is determined by the inspection method, location of the crack, material, etc.

5.1.2 Flaws

There is an interest in the rotorcraft community to consider the presence of flaws, such as corrosion, nicks, gouges, or scratches, rather than cracks.

Since the failure of structures with flaws could be caused by cracks originating at these flaws, the damage tolerance approach presented in this section can be used.

a. Considering flaws as cracks (i.e., the crack will have the size of the flaw it replaces).
b. Establishing equivalent cracks for the flaws in such a way that the equivalent crack growth to failure will be equal to or shorter than the crack initiation and growth from the flaw to failure.

Options (a) and (b) are illustrated in Figure 5-1.

Once a flaw is replaced by the crack using either Option (a) or Option (b), the damage tolerance method described in this section can be applied, including inspections for flaws.

5.2 SLOW CRACK GROWTH REQUIREMENTS

The airworthiness of the slow crack growth structures is assured by the inspection program and the replacement interval (structure's life). The frequency of inspection is determined by crack growth analysis, supported if necessary by testing, of the largest undetectable cracks for the proposed inspection method, assumed at the location, called the critical location, which yields

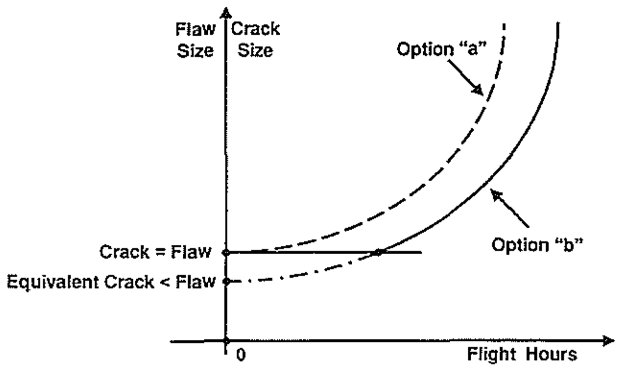


Figure 5-1. Option (a) and (b) concepts.

the shortest crack growth interval under the expected service loading and environment. Where appropriate, the interaction of the growing main crack with the growing initial quality cracks should be taken into account.

If the rogue crack assumed at the critical locations will not grow to critical size under the expected service loading and environment in the life of the structure, no inspections are required.

Such structures belong to the no/benign crack growth structures which can be considered as a subset of the slow crack growth structures.

Crack growth analysis determines crack growth by growing a crack under the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement. The residual strength requirement specifies the minimum required residual strength in terms of the maximum load (limit load) that the structure with cracks must withstand without affecting safety of flight.

5.3 FAIL-SAFE REQUIREMENTS

The airworthiness of the fail-safe structures, depicted in Figure 5-2, is assured by the design features that contain the critical cracks to the separate sections of the structure, and by the inspection/maintenance program which detect the cracking sections of structure and replace or repair them. The replacement/overhaul interval is also defined to guard against simultaneous failure of the otherwise autonomous sections. The frequency of inspections is determined by crack growth analysis of the structure, supported if necessary by testing, with the contained partial failure of its critical section and the initial quality cracks at all possible locations, under the expected service loading and environment. The replacement/overhaul interval is determined by crack growth analysis/testing of the initial quality cracks assumed at all possible locations, under the expected service loading and environment. Inspections can be determined either for (a) partial failure, or (b) a detectable crack prior to the partial failure.

Crack growth analysis determines the crack growth intervals by growing the initial quality cracks under the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement for the partial failure inspection, or by growing the detectable crack until the partial failure plus the concurrent growth of the initial quality cracks in the remaining structure for the less than partial failure inspection. The residual strength requirement specifies the minimum required residual strength in terms of the maximum load which the structure with cracks must withstand without affecting safety of flight. The initial quality crack size at the time of the partial failure should take into account its growth in the intact structure, and should be checked against the residual strength requirement during the partial failure of the structure taking into account the dynamic load increase and load redistribution due to this failure.

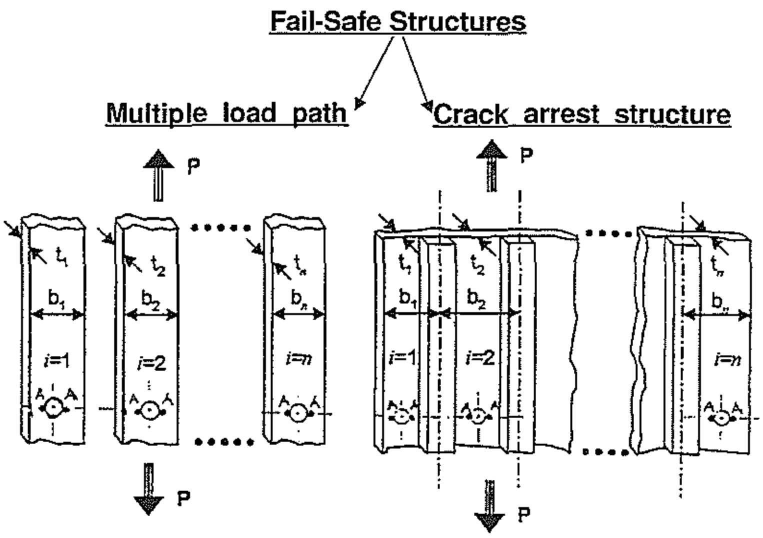
5.4 DURABILITY

Since damage tolerant design may be "on condition" based on inspection for a detectable crack, or the replacement time determined by crack growth time from a rogue crack, durability assessment is not necessarily a design requirement. It is, however, considered reasonable design practice to perform a durability assessment, and to assign the replacement/overhaul interval to ensure the required durability. Such intervals are determined either by crack growth analysis using the initial quality crack, or by testing of the as-manufactured structures.

5.5 CERTIFICATION METHODS

At the certification stage, the structure is fully defined in terms of material and geometry. This would include FEM or other stress analysis to determine stresses in the structure as a function of the external loads. The flight load survey data flown at the outer points of the rotorcraft flight envelope are also available. Rotorcraft usage should be reviewed to determine the expected usage which, combined with the flight load survey data, defines the load spectrum for each critical component of the certified rotorcraft. The load spectrum should be fully cycle counted for oscillatory and associated steady loads, since crack growth is very sensitive to both. The maximum measured load is defined as the limit load and is used for the minimum strength requirement. The number of ground–air–ground cycles, rotor start-stop cycles, and heavy-lift cycles should be defined and incorporated into the load spectrum for each critical component. The detailed description of the load spectrum and related issues are presented in Section 4. Other data needed for damage tolerance certification of any rotorcraft component are

a. Crack growth data comprised of the da/dN vs ΔK curves for applicable stress ratios, R.



A indicates locations of the minimum quality cracks

Total load-carrying area $= \sum_{i=1}^{n} b_i t_i$

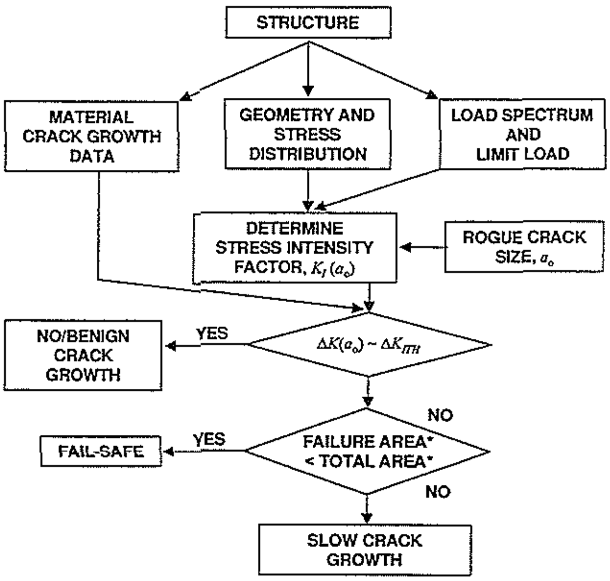
Failure area $= \sup * \{b_k t_k\}$ or $\{b_k t_k + b_{k+1} t_{k+1}\}\ k = 1 \ldots n$ if sections are overlapped and connected together

Figure 5-2. Fail-safe structure.

b. Fracture toughness, K_{IC}, and K_C.
c. Crack growth threshold, ΔK_{ITH}, for the specific material which the component is made, including its heat treatment, material form, grain orientation, and environment.

The geometry of the structure, stress distribution, load spectrum, and material crack growth data are basic input data for damage tolerance certification.

The diagram in Figure 5-3 shows the basic qualification of the damage tolerance structures, with the slow crack growth category dominant; i.e., each damage tolerance structure could be analyzed as slow crack growth.



*Load carrying area, see Figure 5-2

Fig. 5-3. Qualification procedure for damage tolerance structures.

The certification of damage tolerant rotorcraft structure can be accomplished analytically by (1) fracture-mechanics-based crack growth analysis using material crack growth data, (2) full-scale testing, or both. The amount of full-scale testing would vary for each component, dependent upon the structure's complexity, available data, and other factors such as the manufacturer's approach and certifying agency policies.

5.5.1 Crack Growth Analysis

Basic inputs to the crack growth analysis are load spectrum, stress intensity factor solution, and material crack growth data. The material crack growth data are presented as log-log da/dN versus ΔK plots, and come from coupon crack growth tests. Typical examples from Ref. 9 are presented as Figure 5-4. Similar material crack growth data can be found in Refs. 10 and 11. These data show scatter influenced by various factors. Therefore, it is very important to acquire data for the specific material including heat treatment, material form, grain orientation and environment. With the limited number of tested coupons, only average da/dN versus ΔK curves can be defined, whereas with a larger number (at least three) the conservative top of scatter can also be drawn. In the crack growth analysis either average or top-of-scatter curves could be used with the different reduction factors applied to determine the inspection intervals or the replacement interval.

The crack growth analysis concentrates on the cracks in the critical locations. The list of potential critical locations come from

a. The static stress analysis (FEM, etc.) as points of maximum stresses.
b. The test/maintenance data for similar components, considering the origins of cracks, fretting, corrosion and other damages.
c. The flight load survey as points of high measured strains.
d. The static strain survey of the component or other measurements as points of maximum strain, called "hot spots."

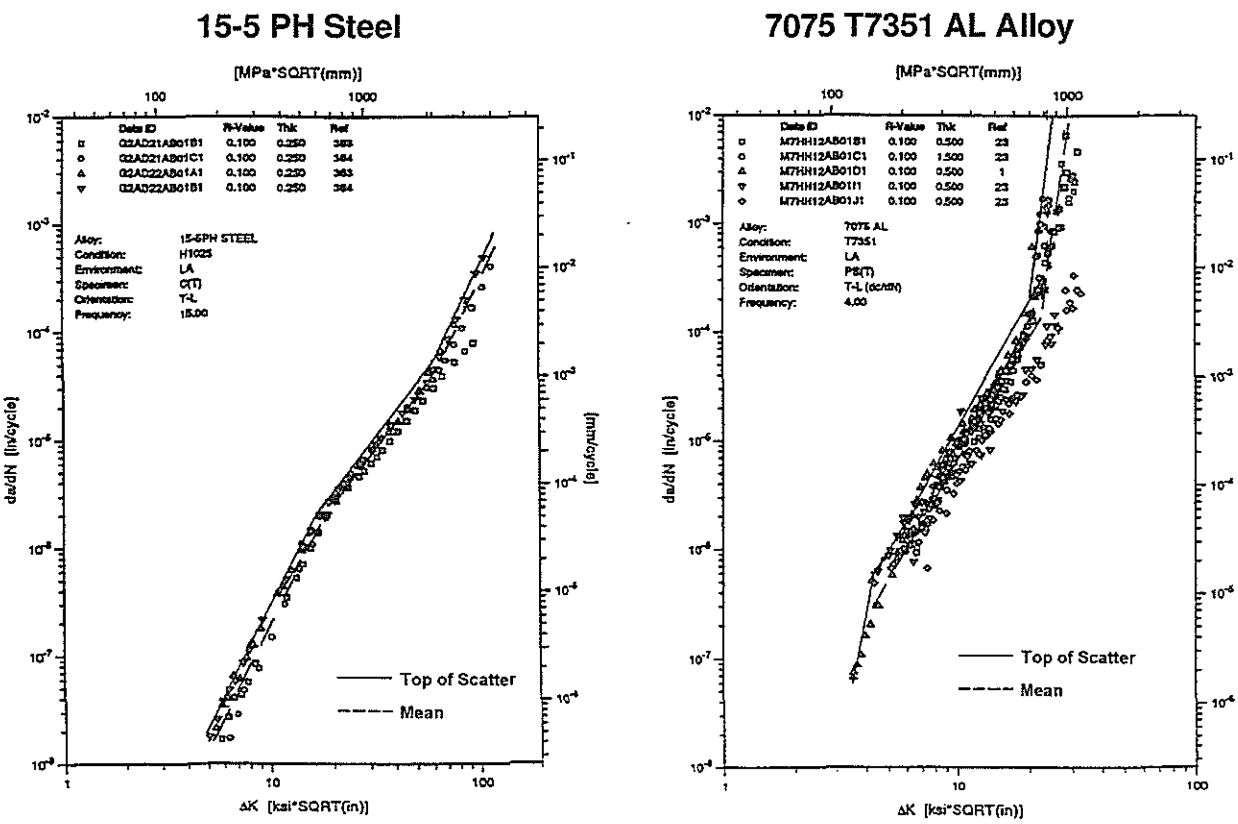


Figure 5-4. Typical da/dN versus ΔK curves.

e. The fatigue test of as-manufactured components considering the origin of cracks, fretting, etc.
f. Review of component inspectability to define the points least accessible for inspection.
g. Review of the possible sites for the widespread fatigue damage prior to reaching the service life.

5.5.1.1 Slow Crack Growth Structures

To substantiate a structure in the slow crack growth category involves consideration of residual strength and crack growth analyses and/or tests. The detectable crack for the inspection method to be used shall be assumed at a location, called the critical location, yielding the shortest crack growth interval under the expected service loading and environment. The factors used to set the inspection interval in the following discussion represent current industry trends. Of course, other factors may be appropriate when considering data quality, crack growth software, crack size, spectrum, design features, etc.

To determine the frequency of inspections, the detectable crack should be assumed at the critical location and the crack growth interval should be determined for the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement (Figure 5-5, curve "a"). The residual strength requirement specifies the minimum required residual strength in terms of the maximum load which the structure with crack(s) must withstand without affecting safety of flight. The frequency of inspections should be one-half of the detectable crack growth interval in cases when the conservative top-of-scatter crack growth data are used in the crack growth analysis, or one-quarter of the detectable crack growth interval when the average crack growth data are used in the crack growth analysis or when the detectable crack growth interval is obtained from crack growth test of one specimen (for two or more specimens, one-half of the shortest detectable crack growth interval can be used).

The inspection threshold should be (1) one-half of the rogue crack growth interval in the case where the conservative top-of-scatter crack growth data are used in the crack growth analysis, or (2) one-quarter of the rogue crack growth interval when the average crack growth data are used in the crack growth analysis, or when the rogue crack growth interval is obtained from the crack growth testing of one specimen. For two or more specimens, one-half of the shortest rogue crack growth interval can be used.

To determine the replacement interval, the initial quality cracks defined in 5.1 should be assumed at all possible locations, and the crack growth life should be determined for the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement. The replacement interval should be one-half of the crack growth life determined either by (1) the crack growth analysis using the average crack growth data or (2) the crack growth test, or should be based on the fatigue life determined by the safe-life testing and evaluation described in Section 7. For non-inspectable structures, the replacement interval is determined by the first inspection, which is defined by considering the rogue crack.

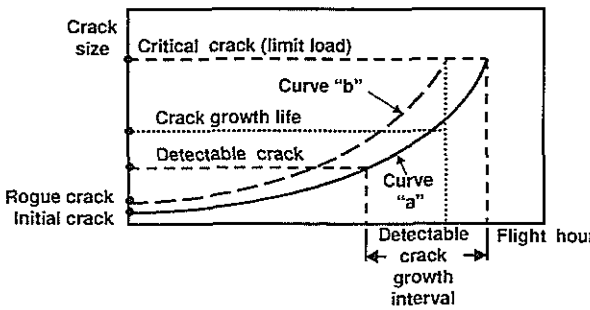


Figure 5-5. Crack growth for slow and no/benign crack growth structures.

5.5.1.2 No/Benign Crack Growth Structures

To substantiate a structure in the no/benign crack growth category requires demonstration either by analysis, testing, or both, that the rogue crack defined in 5.1 will not grow or will not grow critical under the service loading and environment before the structure removal. The crack should be assumed at the critical location, as defined by the largest stress intensity factor range under the expected service loading range including the ground–air–ground cycle.

To determine removal interval (service life), the rogue cracks defined in 5.1 should be assumed at the critical location and the crack growth life should be determined for the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement (Figure 5-5, curve "b").

The replacement/overhaul interval should be (1) one-half of the crack growth life in the case where the conservative top-of-scatter crack growth data are used in the crack growth analysis, or (2) one-quarter of the crack growth life when the average crack growth data are used in the crack growth analysis, or when the crack growth life is obtained from the crack growth test of one specimen (for two or more specimens, one-half of the shortest crack growth life can be used).

The use of the crack growth threshold, ΔK_{ITH}, to determine "no crack growth" structures should be addressed. The crack growth threshold, ΔK_{ITH}, is one of the crack growth parameters, which defines very slow (10^{-9} inch/cycle) or no crack growth conditions. The data currently available for ΔK_{ITH} show large variations. These variations can be attributed to the influence of test procedure, microstructure, crack size, loading conditions, environment, grain size and orientation, etc. The other sources of variations are the lack of an adequate standard for ΔK_{ITH} testing, and an arbitrary definition of ΔK_{ITH}. Therefore, to use ΔK_{ITH} to qualify a "no/benign crack growth" structure, its value should be verified against all available crack growth data in the slow growth region (10^{-7} to 10^{-10} inch/cycle, 10^{-6} to 10^{-9} mm/cycle) for the specific material, material form, heat treatment, environment, and crack sizes. These data should be reviewed and evaluated to establish appropriate value of ΔK_{ITH}. This ΔK_{ITH} should be used to determine weather a structure is in the no/benign crack growth category. In case there is not enough data to define such a ΔK_{ITH} value, a coupon testing program would be necessary. Otherwise, structures should be certified in the slow crack growth category.

5.5.1.3 Fail-Safe Structures

To substantiate a structure in the fail-safe category requires stress, residual strength, and crack growth analyses and or/tests. The structure should be assumed to fit one of the following options:

a. Fail at the critical or most highly loaded section, with the initial quality crack at the critical location in the remaining structure. That would result in the shortest crack growth interval after partial failure under the expected service loading and environment.
b. Grow to partial failure of the detectable crack in the critical location for the inspection method to be used, and the concurrent growth of the initial quality cracks in the other critical locations. That would result in the shortest combined crack growth interval under the expected service loading and environment.

To determine the frequency of inspections for Option (a), the initial quality cracks at the critical locations and their growth in the intact structure should be determined for the expected load/environment spectrum until the contained partial failure of the structure at the worst moment, i.e., at the end of the service life. At that point, the resulting crack should be checked against the residual strength requirement taking into account load redistribution and dynamic effects caused by the partial failure. If the resulting crack would meet the residual strength requirement, the frequency of inspections can be specified by determining the crack growth interval of this crack under the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement (Figure 5-6, curve "c").

The frequency of inspections for the partial failure, Option (a), should be one-half of the partial failure crack growth interval determined either by the crack growth analysis using the average crack growth data, or by the test. The inspection threshold for Option (a) should be one-half of the rogue crack growth interval determined either by the crack growth analysis using the average crack growth, or by the test.

To determine the frequency of inspections for Option (b), the largest undetected crack for the inspection method to be used should be assumed at the critical location, and its growth should be determined for the expected load/environment spectrum until the contained partial failure of the structure at the worst moment, i.e., at the end of the service life. At that point, the resulting crack should be checked against the residual strength requirement taking into account load redistribution and dynamic effects caused by the partial failure. If the resultant crack at the end of the service life would meet the residual strength requirement, its subsequent growth after the partial failure under the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement can be combined with the detectable crack growth interval before the partial failure to determine the frequency of inspections (Figure 5-6, curves "c" and "a"). The frequency of inspections for the detectable crack before the partial failure should be one half of the combined crack growth interval determined either by the crack growth analysis using the average crack growth data or by the test. The inspection threshold for Option (b) should be one-half of the minimal quality crack growth interval determined either by crack growth analysis using the average crack growth data or by the test.

To determine the replacement/overhaul interval, the minimum quality cracks defined in 5.1 should be assumed in all possible locations in the intact structure, and the crack growth life should be determined for the expected load/environment spectrum until reaching the critical size defined by the residual strength requirement. The replacement/overhaul interval should be one-half of the crack growth life determined either by (1) the crack growth analysis using the average crack growth data or (2) the crack growth test, or should be based on the fatigue life determined by the safe life testing and evaluation described in Section 7.

5.5.2 Component Test

5.5.2.1 As-Manufactured Components

The fatigue testing of as-manufactured components can be used to define fatigue critical locations and to determine the replacement interval (service life), i.e., the durability of the slow crack growth and fail-safe structures for which crack growth analysis was performed to establish an inspection interval. The testing could be performed as a constant amplitude S–N testing, following the safe life methodology described in Section 7.

5.5.2.2 Test of Pre-Cracked Components

The spectrum fatigue testing of pre-cracked components can be used to verify crack growth analysis results, to experimentally determine inspection intervals for the slow crack growth structures, and to experimentally determine replacement/overhaul intervals for the no crack growth structures. Where justified, precracked elements or coupons could be used in place of full-scale components if they adequately represent crack growth in the critical area of a component. The test load spectrum should be derived from the fully cycle counted flight load survey data as described in Section 4 with the maximum measured load as the limit load for the minimum strength requirement. The pre-cracking procedures should follow pre-cracking methods described in ASTM Standards.

The crack growth test of one or more specimens can be used to verify the inspection method and to determine an inspection interval. The inspection interval should be one-quarter of the test flight hours for one specimen tested, and one-half of the shortest test flight hours for two or more specimens tested.

The spectrum crack growth test of a component with the rogue crack described in 5.1 can be used to define the replacement/overhaul interval for the no/benign crack growth structures. The replacement/overhaul interval should be one-quarter of the test flight hours for one specimen and one-half of the shortest test flight hours for two or more specimens. In special cases, the test loads could be increased to account for load and material variability and to shorten the test to the one replacement/overhaul interval.

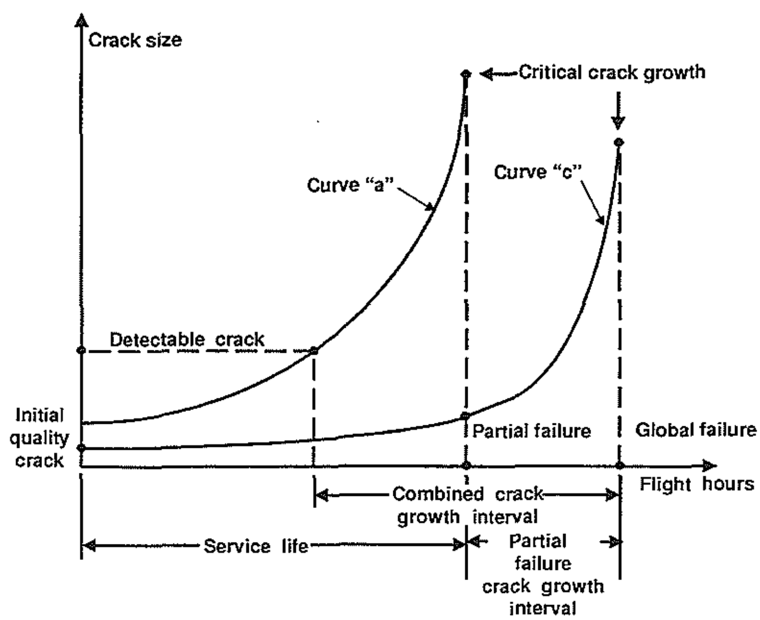


Figure 5-6. **Options a and b – crack growth for fail-safe structures.**

5.6 INSPECTION/OVERHAUL/REPLACEMENTS (MAINTENANCE)

5.6.1 Slow Crack Growth Structures

The maintenance action should ensure removal or repair of cracked or flawed structures by scheduled inspections defined in 5.5.1.1. The maintenance action should also ensure that the criteria of the initial quality cracks of 5.1 and the critical locations of 5.5.1 used to determine the replacement interval are maintained and that the structure is replaced at the interval specified in 5.5.1.1.

5.6.2 No/Benign Crack Growth Structures

The maintenance action should ensure replacement/overhaul of structure at the interval specified in 5.5.1.2. The maintenance action should also ensure that the criteria for the rogue crack specified in 5.1 and the critical locations of 5.5.1 used in establishing the no/benign crack growth are maintained, i.e., that there are no cracks induced by either maintenance or service and environment that would be larger than the crack size used for substantiation.

5.6.3 Fail-Safe Structures

The maintenance action should ensure removal or repair of cracked or flawed structures by schedule inspections defined in 5.5.1.3. The maintenance action should also ensure that the criteria of the initial quality cracks specified in 5.1 and the critical locations of 5.5.2 used to determine the inspection and replacement/overhaul intervals are maintained and that the structure is replaced/overhauled at the interval specified in 5.5.1.3.

5.7 REPAIR/ALTERATION

Structure must be reevaluated in accordance with the requirements and methods of subsections 5.1 through 5.5, with consideration of any structural changes resulting from the repair or alteration. In addition, the crack types and sizes as specified in 5.1 must be reevaluated.

6. FLAW TOLERANCE

Flaw tolerant design and certification of structure is an alternate to damage tolerant design and certification that uses crack initiation methods. It is intended to ensure that should serious corrosion, accidental damage, or manufacturing/maintenance flaws occur within the specified retirement time and/or inspection intervals of the component, the structure will not fail.

The flaw tolerance method may not be valid for the case where the flaw being considered is a true crack, since "crack initiation" has already occurred. In this event, an analytical verification of no growth of this crack under the projected flight/ground loading spectrum is conducted.

This method provides component management methods based on the assumption of the existence of flaws in the component's critical areas. Two sizes of flaws are considered: (1) "Barely Detectable Flaws" are used to conservatively represent the largest probable undetectable manufacturing or service-related flaws; (2) "Clearly Detectable Flaws" have a high probability of detection by the prescribed inspection method. The sizes considered in the flaw tolerance evaluation are limited by the probable maximum size of flaw that would not be detected in a routine inspection.

The approach to flaw tolerant design of principal structural elements depends on the type of structure. The approach for single load path structure has two requirements: (1) A barely detectable flaw will not initiate a propagating crack within the retirement time of the component; (2) a clearly detectable flaw will not initiate a propagating crack within an inspection interval, inspecting for the presence of the flaw. The approach for multiple load path or fail-safe structure also has two requirements: (1) A barely detectable flaw will not initiate a propagating crack within the retirement time of the component; (2) a barely detectable flaw in a second load path, after the first load path failure, will not initiate a propagating crack within an inspection interval, inspecting for first load path failure.

The flaw tolerance evaluation is accomplished by (1) establishing which components/areas are to be designed and substantiated as flaw tolerant; (2) developing operational stress spectra for these areas based on the design usage spectra and flight loads; (3) determining the maximum probable undetectable and clearly detectable flaw sizes, and critical locations, based on a review of historical data and manufacturing processes; and (4) determining life limits and inspection intervals using crack initiation/cumulative damage analysis and test.

Design features which should be considered in achieving a successful flaw tolerant structure include multiple load path construction; structural design that allows for required inspections; shielding or protective coatings and treatments that prevent and/or reduce the severity of environmental and/or mechanical damage; and surface residual stress processes, such as shot peening and cold working, that inhibit crack initiation.

6.1 FLAW DEFINITION

Flaw types and sizes to be imposed on each component being substantiated by flaw tolerance are defined, and are submitted with accompanying rationale to the certifying agency for approval. The first element of this process is a systematic evaluation of the types and sizes of flaws to be considered for each component. The types of flaws considered should include nicks, dents, scratches, inclusions, corrosion, fretting, and wear. Other factors which may influence the flaw tolerance approach are loss of mechanical joint preload and bolt torque.

The systematic evaluation should include a compilation of historical experience with similar parts and materials, including field service reports, overhaul and repair reports, metallurgical evaluations, manufacturing records, and accident/incident investigations. The design, manufacturing, and maintenance practices that could result in errors or defects should also be evaluated. Planned inspection methods and practices also define what are the sizes and locations of flaws. A coupon program is valuable in indicating the strength-reducing effects of various types of flaws, *S–N* curve shape, and statistical scatter for flawed parts, and, if needed, determination of "equivalent" flaw types and sizes that may be used on full-scale test specimens. This is illustrated in Figure 6-1.

Consideration should also be given to factors that reduce the chance of error, such as "frozen processes," Flight Critical Parts programs, and material selection to avoid inclusions and defects, and sensitivity to manufacturing errors. Another possibility is to limit the flaws considered if the design includes surface treatments that protect against environmental and/or accidental mechanical damage. In addition, it may be appropriate to show by means of a joint probability analysis that some flaws

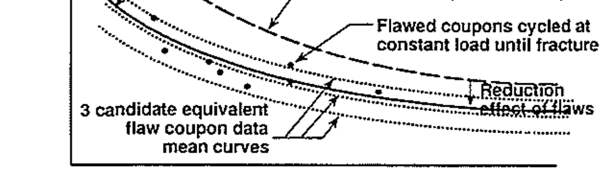


Figure 6-1. ***S–N*** **evaluation of flawed coupons.**

may be eliminated from consideration because they have an extremely remote chance of occurrence. This analysis combines the distribution of likely flaw sizes, the criticality of location and orientation, and the likelihood of being missed in an inspection.

If the evaluation above determines that a possible flaw is a true crack, the flaw tolerance method may not be valid. Cracks of this sort could be related to manufacturing errors in plating or surface treatments, heat treatment, or cold working. For these specific defects, an analytical evaluation should be conducted, using the methods of Section 5, to verify that these cracks will not grow under the expected spectrum of flight/ground loads.

6.1.1 Barely Detectable Flaws

Flaws in this category are intended to represent a conservative "worst case" of undetectable flaws, i.e., those that do not have a high probability of detection by the prescribed inspection methods. The flaws to be considered include nicks, dents, scratches, fretting, or corrosion that may occur in the manufacturing or service life of the structure.

6.1.2 Clearly Detectable Flaws

Flaws in this category have a high probability of detection by the prescribed inspection method. Flaws to be considered include nicks, dents, scratches, fretting, corrosion, and mechanical joint preload and/or bolt torque.

The maximum size of clearly detectable flaws to be considered may be limited by the smaller of either of the following: (1) the flaw which is obvious and readily detectable by routine visual inspection, which means that it would not be expected to remain in place without corrective action for any significant period of time; or (2) the flaw determined by an evaluation of the maximum probable flaw size.

6.2 SINGLE LOAD PATH STRUCTURE REQUIREMENT

For single load path helicopter structure it is especially important to show tolerance to flaws and defects that could lead to crack initiation and failure. The assumption is therefore made that significant flaws and defects are present in the structure at critical locations. Since any undetectable flaws would remain in place for the life of the structure, a high margin upper limit on time in service is determined by analysis and/or test so that no crack will initiate from these flaws. Barely detectable flaws are used in analysis and test to represent a conservative worst case of these flaws.

In addition, an inspection program is established to protect against failure for structure with larger flaws—flaws that can be readily detected by inspection. Analysis and/or test results for structure with clearly detectable flaws are used to calculate a high margin inspection interval for the structure, inspecting for the presence of the clearly detectable flaw. This position assures that no cracks will initiate from clearly detectable flaws for the inspection interval. If the inspection shows that the flaw is not present, the component may be returned to service for another inspection interval, up to the retirement time.

6.3 MULTIPLE LOAD PATH STRUCTURE REQUIREMENT

Multiple load path structure is subjected to the same requirement for the establishment of a retirement time as single load path structure, i.e., barely detectable flaws in critical locations in all load paths are shown to not cause crack initiation within the retirement time, with margin.

In addition, an inspection program is established to protect against complete failure in the event of the failure or disablement of one load path. No assumption needs to be made as to the cause of the first load path failure; however, barely detectable flaws are assumed to be present at critical locations in the remaining load paths. Additionally, it should be verified that the remaining load paths have a full limit load capability. The inspection interval is determined so that no cracks will initiate from these flaws, with the failed load path, with a high margin. The inspection is for the failure of the first load path. If the inspection shows that no load paths have failed, the component may be returned to service for another inspection interval, up to the retirement time.

6.4 DURABILITY

For flaw tolerant design, the durability requirement is satisfied, since it must be demonstrated that a barely detectable flaw will not initiate a crack in the life of all principal structural elements (6.2 and 6.3). This is a more severe durability requirement than for conventional safe life (Section 7).

6.5 DESIGN AND CERTIFICATION METHODS

Analysis and test are used together to accomplish the certification.

6.5.1 Analysis Methods

The design process begins with the specification of materials to be used. These specifications control the processing and quality of the material, thereby allowing the use of average properties as a basis for stress allowable in fatigue during the design process, and the compensation for the variability in fatigue strength of a material by a standardized reduction in allowable stress from the mean stress.

The quantification of stress as a function of applied loads is fundamental to the design process. The primary structure is analyzed using finite-element or other validated techniques to determine the magnitude of stress within the component for various loading conditions. This provides accurate insight into both stress concentration magnitude and the size of the stress concentration zone. In accordance with standardized fatigue methodology, the working curve is reduced to account for the size effect that is observed between material coupons and full-scale components.

Each potential fatigue crack initiation zone is also analyzed to determine conditions that uniquely affect the fatigue strength of the zone, such as surface finish and the possibility of fretting. The fatigue allowables are adjusted accordingly for these factors.

The structural analysis of metallic components includes a reduction factor in the calculation of a working stress allowable to account for the physical damage during the manufacture and service life of a component. The factors used during the design are based on preliminary estimates of the effect of physical damage, consistent with the levels observed during service for existing designs. A test program using material coupons with physical damage is used to validate the factors for the design.

Predictions of component life with barely detectable damage are made using the above procedures. Predictions of inspection intervals are also made using clearly detectable damage in single load path structure, and using a failed load path in multiple load path structure (with barely detectable damage in remaining load paths).

6.5.2 Coupon Testing

A coupon test program is essential in any safe-life design to provide the basic *S–N* data for the specific materials selected. This includes the *S–N* curve shape and basic material scatter. These characteristics may vary with the specific alloy, manufacturing method, heat treatment, surface treatment, and stress concentration.

For a flaw tolerant design, a survey is recommended of coupons with representative types and sizes of flaws to provide reduction factors for use in the design evaluation. This is illustrated in Figure 6-1. This data may also help decide what specific materials, manufacturing methods, and inspection criteria are required to achieve the design goals.

In addition, a flawed coupon program can be used to determine equivalent flaws for use in the full-scale test program, where each critical location may need to be evaluated for multiple types of flaws. Another issue is that it may be difficult to apply exactly the desired flaw at the desired location on the full-scale parts (a corrosion pit of a certain size in a certain radius, for example). Once the characteristics and reduction factors due to representative flaws are known, a coupon program can be conducted to determine a set of flaws that have a equivalent effects but are easy to apply and easy to control. These flaws could be sharp file notches or specific milled notches, for example. This is illustrated in Figure 6-1. A single "worst case" equivalent flaw can then be easily imposed on each critical area of the full-scale parts.

6.5.3 Full-Scale Testing – As-Manufactured Parts

At least one full-scale strain survey and fatigue test specimen in an as-manufactured condition is recommended. The full-scale test automatically accounts for factors assumed in the design—stress concentration, surface finish, residual stress, fretting, and load sharing. It provides correlation with the design stress analysis, serves as a baseline for comparison to flawed parts, verifies the predicted location of fatigue-critical areas, and validates the baseline cumulative damage analysis.

The strain survey and the fatigue test are conducted in simulated flight load setups. Several setups may be required for complex parts, and for multiple loading conditions (in-flight and ground–air–ground, for example). Instrumentation is provided to control applied loads, to survey load and strain distributions, and to provide master parameters for correlation to flight test conditions.

The fatigue test determines crack initiation characteristics by means of conventional *S–N* testing as described in Section 7. Constant-amplitude or spectrum accelerated-load cycling is conducted and continues until a crack initiation is detected by the best laboratory means available. The test may be terminated at this point; however, it may be useful to continue and obtain crack propagation data under spectrum loading as described in Section 5. This data is always useful, could help manage a future problem with the part, and could provide a slow crack growth or crack arrest damage tolerant position for this area of the component.

A mean *S–N* curve is established based on the loads and cycles to crack initiation, as illustrated in Figure 6-2. Multiple specimens provide increased confidence in the location of the mean curve and help validate the predetermined curve shape. A working curve is established from the mean curve using scatter factors based on either historical data or a statistical evaluation of the full-scale test data for the specific part or similar parts.

6.5.4 Full-Scale Testing – Parts with Flaws

Conventional *S–N* testing is conducted on the flawed parts as described in 6.5.3 and Section 7. Load level choice for the initial flawed specimen may be somewhat lower than for the as-manufactured parts based on the predicted strength reduction. Load levels for subsequent specimens can be chosen to fill out the flawed mean *S–N* curve in the conventional manner.

Multiple specimens should be tested, as approved by the certifying agency. Each specimen should have flaws imposed at each critical location, as determined by the flaw evaluation and analysis conducted as described above. In the event that multiple types of flaws should be evaluated at one location, or if the imposition of the exact type of flaw needed is difficult (such as a specific corrosion pit), an "equivalent" flaw, as determined by the coupon program described above, is recommended. The equivalent flaw could be a sharp file notch of a certain size or a specific milled notch. The flaw evaluation and implementation plan should also be approved by the certifying agency.

Initial testing of flawed specimens should be for the barely detectable flaw case. New flawed specimens may be fabricated for the clearly detectable flaw test program. However, it may be possible to use any runout (non-fracture) specimens from the barely detectable flaw test program by imposing the clearly detectable flaws on these specimens.

Multiple load path structure is first subjected to *S–N* testing with barely detectable flaws in place at the critical locations in all load paths on the part. This will establish the retirement time of the part as described in 6.2. The second load path testing to determine an inspection interval can be conducted on this same part by allowing the first crack initiation to propagate to complete failure of that load path. This method ensures that any fatigue damage that occurs in the second load path prior to complete failure of the first load path is included in the test specimen. Alternatively, the first load path can be completely disabled by removal of critical fasteners, or by mechanically severing it. In this case, especially if the test is conducted on a new part, the damage that would occur in service to the second load path prior to the complete fracture of the first load path must be accounted for in the damage calculation described below.

If any of the flawed specimen testing produces cracking at a point that was not included as a flaw location, a re-evaluation of the flaw selection and location program should be conducted. This evaluation could include full-scale test specimens with imposed flaws at the new location, and/or a combination of full-scale and coupon test results addressing the potential effect of flaws at the new location.

Mean *S–N* curves for the crack initiation data from these tests are derived using the procedures of Section 7, and are illustrated in Figure 6-3. One possible variation from the methods in Section 7 is that the *S–N* curve shape chosen for the analysis of the flawed data should reflect a "High K_t" shape or a "fretting" shape if appropriate for the particular flaw and material.

Full scale testing is not often conducted for highly redundant airframe structure. Flaw tolerance assessment for these structures may use coupon or element testing and analysis complemented by flight measured loads/stresses in the most critical areas.

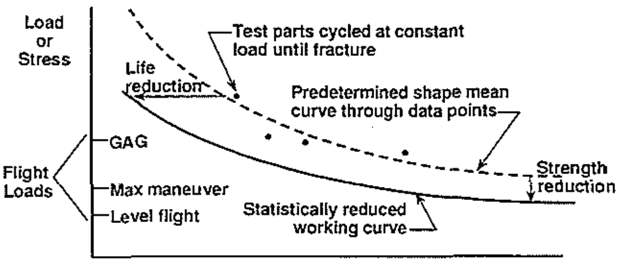


Figure 6-2. *S–N* curve for as-manufactured components.

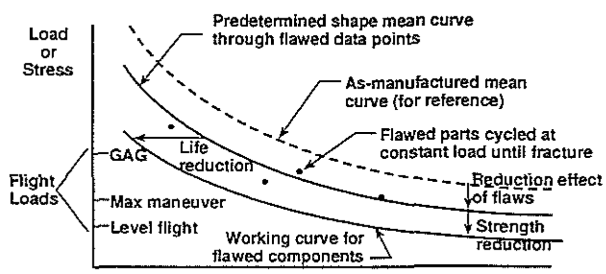


Figure 6-3. *S–N* curve for flawed components.

6.5.5 Combination of Methods

Mean crack initiation S–N curves may also be derived by a combination of methods. Providing a complete set of full-scale test specimens for each of the three types of components—as-manufactured, barely detectable flaws, and clearly detectable flaws—is not necessary if the effects of flaws have been specifically addressed by analysis, experience with similar parts, and/or a coupon program comparing the as-manufactured and flawed conditions. For example, mean curves for barely detectable flaws could be derived from as-manufactured full-scale test data reduced by a factor determined in an analytical or coupon evaluation. This is illustrated in Figure 6-3, where the "reduction effect of flaws" could be determined by analytical or coupon evaluations.

6.6 REDUCTIONS FOR SCATTER

Reductions for scatter are applied to the S–N mean curve to establish a working curve for each of the as-manufactured and flawed cases described above. For spectrum testing, factors of safety are normally achieved by testing for more than one lifetime and/or by load acceleration.

Conventional safe-life reductions as described in 7.4 are appropriate for the as-manufactured specimens, illustrated in Figure 6-2.

Reductions in flaw tolerant design may be less than that used for conventional safe-life, illustrated in Figure 6-3. This is based on the observation that the occurrence of a significant flaw at a precise critical location on a component that is also at the absolute low bound of statistical strength is extremely unlikely. The use of conventional factors would essentially be an assumption that every single part in service was critically flawed at every critical location for its entire service history, which is excessively conservative.

In addition to use of the scatter factors described above, working curves will be further reduced if necessary to encompass all of the test fracture points.

The working curves described here are used in a conventional safe-life analysis to determine component retirement times and inspection intervals. The basic methods of Section 7 are used, as approved by the certifying agency. The flight loads, usage spectrum, and damage calculation procedures should be the same for each calculation.

6.7 INSPECTION

Inspection methods and intervals based on flaw tolerant structural substantiation are described below. In addition to these inspections, conventional inspection procedures for helicopter structure are still required. These include general inspections for condition, specific inspections of known critical areas, monitoring of condition by crew or by automated monitor systems, special inspections associated with overloads such as a rotor overspeed, and structural inspections associated with component overhaul and repair.

6.7.1 Single Load Path Components

The inspection is for the clearly detectable flaw, and the inspection interval cannot exceed the safe life calculated using the initiation data for the clearly detectable flaw as described in 6.6.

For the case where flaws of multiple types are considered and/or where multiple flaw locations are considered on one component, each should be evaluated independently. The simplest approach would then be to impose the lowest calculated inspection interval on the part, on all critical areas of the part, for all of the substantiated flaws. Alternatively, each mode considered could result in a special inspection interval and a special inspection method for the specific flaw at the specific location.

If no flaw is found at the time of inspection, the component may be returned to service for another inspection interval, up to the limit of the component retirement time. If a flaw is found the component may be retired, or, if a repair procedure for a detected flaw has been substantiated to the certifying agency to return the part to its original strength, it may be repaired and returned to service.

6.7.2 Multiple Load Path Components

The inspection is for the failed or disabled load path, and the inspection interval cannot exceed the safe life calculated using the initiation data for the barely detectable flaw in the second load path following first load path failure, using the method described in 6.6.

An alternate method for inspection interval calculation may be used if the result is more conservative, namely using an as-manufactured condition of the second load path as the strength basis, and the conventional safe-life reduction factors described in 6.6. This choice could be appropriate if the effects of barely detectable flaws are known to be small.

Cumulative damage that may occur in the remaining load paths prior to complete failure of the first load path must be accounted for in the inspection interval safe-life calculation. This effect is already included in full-scale test programs where all load paths are subjected to the accelerated fatigue loading and the failure of the first load path results from the propagation of a crack. However, in those substantiations where the first load path is mechanically severed or disabled, it should be assumed that this event occurs at the end of the established life of the component. The damage existing on the remaining load path at this time should be estimated based on its mean strength (i.e., no strength or life reductions), with all load paths intact. The inspection interval safe-life damage calculation, using S–N data and loads for the failed load path configuration, would be summed to 1.0 reduced by the amount of this pre-existing damage.

If the failed load path is not found, the structure may be returned to service for another inspection interval, up to the limit of the component retirement time.

6.7.3 No Special Inspection Required

If the calculated inspection interval exceeds the design life of the structure, no special inspection is required. The retirement time for the component may also be set equal to the calculated inspection interval, resulting in no special inspection required.

6.8 REPLACEMENT TIME

The upper limit of time in service for a flaw tolerant component cannot exceed the lower of (1) retirement time based on parts with barely detectable flaws and the reduced scatter factors described in 6.6; or (2) retirement time based on as-manufactured parts and the conventional scatter factors described in 6.6.

The inspection interval calculated in 6.7.1 or 6.7.2 may be substituted for the above retirement time if lower, resulting in no special inspection required.

6.9 REPAIRS AND ALTERATIONS

Any structure subjected to a repair, which could possibly affect its structural reliability, must be re-evaluated in accordance with the requirements and methods of 6.1 through 6.8. In addition, the flaw types and sizes as specified in 6.1 must be reevaluated.

7. SAFE LIFE

The original and most commonly used methodology to establish the retirement life of a Principal Structural Element (PSE) is the conventional safe life approach. This approach to the design and certification of the helicopter establishes a replacement time for a component during which time the structure can withstand the design loading spectrum without the occurrence of detectable cracks. This methodology calculates a retirement time with a high reliability.

The safe life approach is built on the three basic elements of strength, loads, and usage as previously discussed (see Figure 7-1). This methodology establishes the mean fatigue *S–N* strength curve, a conservative working *S–N* curve, and a service or retirement life using the service loads and usage spectrum discussed in detail in Section 4. The mean *S–N* curve or life is reduced using scatter factors based on the material and manufacturing variabilities and test parameters. Example scatter factors are presented in this discussion, but should be considered only as guidelines.

Each manufacturer has an historical data base for the different components, materials, joint types, etc. that they are comfortable using in the design and certification process to produce reliable parts. The safe life approach will probably always be used in some degree even for the components that are certified as damage tolerant.

The following subsections discuss the first basic element of strength and how all three basic elements are used together to determine the service life of a component. The discussions go into detail for *S–N* testing, spectrum testing, scatter factors, and the determination of unlimited life.

7.1 SAFE-LIFE REQUIREMENTS

Analysis and or test shall be used to determine that a detectable crack will not occur in the life of the structure. Each portion of the flight structure, the failure of which could be catastrophic, must be identified and evaluated. The locations and modes of probable failure must be determined. For the replacement time evaluation, it must be shown that the probability of catastrophic fatigue failure is extremely remote within the replacement time.

The material strength properties used to establish the *S–N* curve must be based on enough tests to establish design values on a statistical basis. The formulation of the *S–N* curve must include the effects of the various mean values in the loading spectrum.

7.2 CERTIFICATION METHODS

Fatigue is the progressive process of crack initiation (or failure) of a part due to the repeated application of varying amplitude loads, any one of which will not produce failure. The relationship between the applied loads and the number of cycles to a standard crack size (or to failure) needs to be established and is defined as the *S–N* curve. The loading spectrum applied to the part also needs to be established to cover the worst anticipated usage. See Section 4 for a more detailed discussion on the loads and usage spectrum. The loads or stresses used in the analysis are generally measured during flight.

In general, the linear cumulative damage fatigue hypothesis developed by Palmgren-Miner is used. Miner's theory states that the fatigue damage introduced by a given stress level is proportional to the number of applied cycles divided by the total number of allowable cycles at the same stress level (Ref. 12). The total damage is the sum of all the different cycle ratios. The fatigue life is determined when the sum of the all the cycle ratios equals one. This method has been successful in the helicopter industry in establishing high-margin retirement lives when the part stresses are known, the *S–N* curve is known, and the applied service loads have been properly determined. No accident has ever been blamed on Miner's theory. Since there is variability in the material strength, the *S–N* curve, and the service spectrum; scatter factors are applied to one or all of the above elements to ensure that the probability of in-service failure is extremely remote.

Other cumulative damage theories are used on a limited basis. The local strain approach that accounts for the elastic damage, plastic damage, and local residual stresses is used for parts with high stress concentrations or with local stresses above the yield stress. This methodology requires material data that defines the cyclic stress-strain curve, plastic strain *S–N* curve, and elastic strain *S–N* curve. Probabilistic analysis considers the distribution of static and fatigue strength and the loading distributions. Because the loading distributions for all the maneuvers considered in the service spectrum are not well defined (both pilot and air quality variabilities), the probabilistic approach has been used only on a limited basis. Since the local strain and the probabilistic methods are specialized and not universally used in the industry, they will not be discussed further in this paper.

7.3 *S–N* TEST

Constant amplitude or *S–N* testing is the most commonly used method to establish the appropriate *S–N* curve for parts that are now in service. A number of parts are tested at various load levels to determine the mean *S–N* curve (Figure 7-2). A suitable reduction factor is then applied to the mean curve to determine the working *S–N* curve. A sufficient number of data points are needed to be confident in the curve shape and mean stress effects.

7.3.1 *S–N* Curve Shape

All of the helicopter manufacturers use standardized *S–N* curves to some extent. These curves are based on published data (as in MIL-HDBK-5, Ref. 13) or from company generated coupon data. The company data are usually comprised of multiple constant amplitude coupon tests that are reduced to form standardized *S–N* curves. These curves may be normalized to the endurance limit or to the ultimate strength when no endurance limit is appropriate. These curves take into account the material type, temper, surface preparation, environment, and K_t. Many company curves are also constructed for joints that take into account load transfer and fretting. The standardized curves

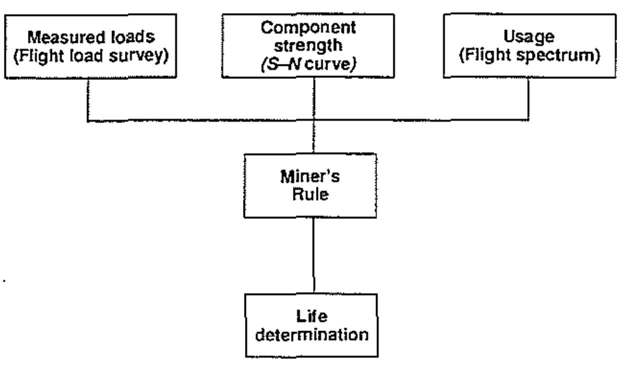


Figure 7-1. Elements of safe-life determination.

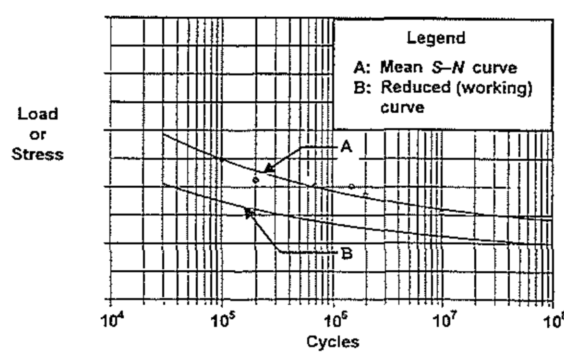


Figure 7-2. Typical *S–N* curve.

are mean curves, and the mean shapes are determined as best fit curves through the test data. The S–N curve may be represented in terms of alternating or maximum load (or stress).

Mean or steady stress effects are accounted for in four different ways. One method is to interpolate between different mean value or R-ratio curves. The method of equivalent stress as presented in MIL-HDBK-5 multiplies the stresses by a $(1-R)^{\gamma}$ factor to normalize to an $R = 0$ curve. Normalizing all mean stress (or load) curves using a Goodman curve is also commonly used. The final method is to conservatively use a high-percentile mean stress curve as the basis for all loads.

Since the standardized S–N curves are constructed from many data points, a well defined variability is known for the specific curve. The data from many curves may be pooled together to gain an even greater confidence in the material variability. This high confidence variability, which is usually expressed as the coefficient of variation (COV), is often used as the lower bound COV for subsequent component fatigue tests.

7.3.2 Mean Curve Determination

Normally, a standardized curve shape is applicable for a component, requiring only a limited amount of testing (typically, four to six specimens) to establish the mean curve location. A lesser number of specimens may be used if a high strength is indicated. Testing is usually conducted to verify the endurance limit plus the most damaging loads expected in service. Testing may be conducted only near the endurance limit, if all flight loads including the ground–air–ground (GAG) cycle are expected to be below this level. The goal of the S–N testing is to adequately define the endurance limit. The highest test load level should include the highest expected GAG cycle. These loads are often increased (enhanced) to account for the reduction factors that are required for the working curve and to identify any additional failure modes. Care shall be taken when using enhanced loads to verify that the failure mode is not changed due to the load enhancement. To account for the interaction effects of large shifts in the mean loads of main rotor blades, periodic GAG cycles may be applied during the S–N testing. For some components, separate GAG testing may be required.

The number of data points at each alternating level is from one to three or more specimens. Testing is usually conducted to establish the endurance limit plus a load or stress high enough to encompass the highest load expected in service. This is to permit interpolation instead of extrapolation of the S–N curve for all flight cycles during the fatigue damage calculation. The endurance limit is defined as at least 10^7 cycles for steel and titanium and 5×10^7 cycles for aluminum.

Special consideration is taken to handle runout data. It is generally conservative to consider a significant runout data point a failure point and to use it in the calculation of the mean curve. A low runout point falling well below the S–N curve is usually considered an insignificant data point and ignored. If all the data are runout points, then the mean curve may be defined by either conservatively considering the highest runout point as a fracture point or performing a statistical analysis on all the data.

7.3.3 Data Scatter

After the mean S–N curve is defined, a working S–N curve is established to account for material and manufacturing variabilities plus any other company specific conservatism. This subsection presents one method that is used to determine the test data scatter on strength. This method is illustrated in Figure 7-3. The basic curve shape is assumed to be appropriate for all test data points. The curve is passed through each data point by adjusting the endurance limit, or other strength parameter, so that points may be established for all data at a common N (cycles-to-fail) value, which is usually the endurance limit.

The variability of all the "projected" data points is determined. The Normal distribution is usually assumed although the

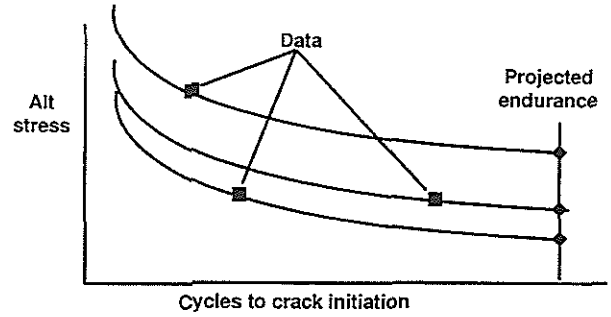


Figure 7-3 Data scatter on strength.

Weibull and Log normal distributions are also used. The working curve is then found by reducing the mean curve by a factor discussed in 7.5. The working S–N curve shall always be set at least as low as all the test data. Thus at least a minimum reduction will always be used. The factors that are used are justified by each manufacturer to the certifying agency.

7.4 SPECTRUM TEST

A spectrum fatigue test is a test conducted on a component or assembly that has more than one load level applied. The simplest spectrum is a test that has a runout at one stress (load), and a second higher stress is then applied. The spectrum may also be designed to directly relate the test cycles to flight hours. This representative type of spectrum test is often the method of choice for damage tolerance or crack growth testing or for relatively large components. A spectrum test may also be used to define the mean S–N curve if the general curve shape of the part is known. Once the S–N curve has been established for a component with a spectrum fatigue test, a different service spectrum may by used in the service life calculation. The method for defining the S–N curve is explained below.

A spectrum fatigue test is especially useful when a part is loaded by more than one type of load. A tailboom loaded by vertical bending, lateral bending, and torsion is one example. The phasing of the different loads for the different maneuvers or loading cases are properly applied to the spectrum fatigue test article. A properly defined spectrum test will also correctly load a number of different locations on the part, component, or assembly.

7.4.1 Spectrum Derivation

The fatigue test spectrum is based on measured flight test data from the flight strain survey (FSS) if that data is available. For tests performed before the FSS data is available, a conservative analysis is used such that there is an extremely low probability that the load spectrum will be exceeded in service. As a minimum, the spectrum includes the maximum and minimum loads expected in the service spectrum of the component. Thus, the GAG cycle is naturally included as well as the most damaging service cycles. Spectrum loads that are near the expected endurance limit of the part are also often applied to demonstrate the high cycle (endurance limit) capability.

The representative test spectrum is broken into repeatable time blocks. Each time block is designed to be repeated at least 10 and preferably a minimum of 50 times during the fatigue test. The time block should have a minimum of six different load levels unless the component loads indicate that fewer load levels are appropriate. The load that is exceeded at least 10 times during the service life of the part is the highest load incorporated in the time block. The high load is also evaluated to ensure that beneficial residual stresses are not introduced into the part. If beneficial residual stresses are introduced, then a lower maximum load level (clipping) may be selected. The number of low load cycles are limited (truncated) to those loads that are expected to produce a significant amount of fatigue damage. The load levels are often reduced by using the method of equivalent damage. Care is always taken to ensure that enough cycles are included to account for the effects of fretting during the test. The GAG cycles, which are often the most significant cycles in the spectrum, are included in the test spectrum. In some cases, the GAG cycle is included as a specific cycle within the spectrum. In other cases when the spectrum simulates a flight-by-flight ordered spectrum, the GAG cycle is inherently included by properly ordering the test cycles.

Once the test spectrum is defined, the complete time history of the spectrum is cycle counted to determine the equivalent applied cycles. These cycles are then compared to the predicted service loading history. The comparison consists of predicted fatigue damage, alternating exceedance, maximum load exceedance, and minimum load exceedance.

There are operations (e.g., logging) that are considered special operations. These special operations are considered on an individual basis. In most cases, the differences can be handled analytically by considering a different usage spectrum in the analysis.

7.4.2 Determining Mean Curve Location

If the basic curve shape is known, the *S–N* curve may be determined from the spectrum fatigue test. The procedure to find the correct placement of the curve is to first determine all the cycles applied to the test specimen. The cycles due to shutting the test machine down at the end of the shift, all pretest calibration cycles, and all inadvertent cycles should be cycle counted and included in the analysis.

A fatigue analysis is then conducted using Miner's theory and varying the *S–N* curve endurance limit (or other curve location parameters as appropriate) until a test damage of 1.0 is calculated using the previously defined test cycles. This curve is then the appropriate *S–N* curve for the particular failure mode produced by the test. Multiple specimen tests are handled the same way, and the data scatter is determined the same way as explained above. The same reduction factors presented in 7.5 are applied to the mean *S–N* curve to yield the working *S–N* curve.

It is obvious that if the test spectrum loads are enhanced by a factor, the test working stress levels can be made to approach the service stress levels. The test time can therefore be reduced to match more closely the safe life period desired.

7.4.3 Service Life

The results of the spectrum fatigue test are analyzed in one of two ways. If the applied test spectrum is equivalent to or exceeds the expected service spectrum, the test time is directly related to the service life. A reduction factor is then applied to the test time or applied cycles. The other option is to apply a reduction to the alternating stress as is done for the *S–N* testing. More detail on these reduction factors is given in 7.5.

7.5 SCATTER FACTORS

A scatter factor is applied to the mean curve to establish the working *S–N* curve. The factor may be applied to the strength, the life, or a combination of the two. These factors can be based on historical data or on a statistical approach, but must take into account the number of specimens. Some of the manufacturers have established factors based on large historical data bases, including material and manufacturing variabilities and the test parameters that have proven to be conservative.

The subsections below describe in more detail the calculated scatter factors used on strength and life, and how combinations of the two are possible. The final factors used are justified by each manufacturer and approved by the certifying agency.

7.5.1 Factor on Strength

The working factor for strength is usually applied only to the alternating strength values. The working *S–N* curve is therefore straight below the mean *S–N* by a fixed percentage (working factor). Typical mean and working curves are illustrated in Figure 7-2. The working factor can be based on historical data bases. The most common scatter factor in the helicopter industry has been to reduce the mean alternating strength value by three standard deviations (3σ). This factor assumes a normal (or log normal) distribution on strength. This is equivalent to a reliability of 99.865%. When a distribution other than the normal distribution is used, the working curve may be set to the same 99.865% reliability. Since both the composite analysis and the new AC 25.571-1C (Reference 3) use A-Basis (99% reliability with 95% confidence), some manufacturers may choose to use the reduction to establish an A-Basis curve. In any case, the historical reduction factor should be used if it is more conservative. The working factor is lowered, if required, to at least encompass all the valid data points.

7.5.2 Factor on Life

For metallic components, the working *S–N* curve life is calculated by applying an appropriate reduction factor. Typically, this value ranges between 3 and 9, depending on confidence in the test loads, the number of specimens, complexity of the structure, etc. The higher, more conservative factor should be used when using only analysis. Rotating components may use higher factors.

7.5.3 Combined Factor

The reduction to the strength level is usually more severe than the life reduction in the area of normal operational loads (see Figure 7-4). Since the life reduction is dominant at the low life end of the *S–N* curve, a change in the reduction method is appropriate if the part is highly loaded. This change-over point needs to be determined. If it falls within the normal spectrum loads, it needs to be accounted for in the working curve.

7.6 UNLIMITED LIFE

Many components on a helicopter are loaded below their respective working or reduced endurance limits during all phases of operation including GAG. For these components, the calculated fatigue damage is zero and the safe life is infinite. The manufacturer would not specify a retirement life for these parts. Normal maintenance inspections are still defined to look for corrosion and unusual wear. The part is essentially on-condition.

Other components may accumulate some fatigue damage, but the damage is small; and the predicted service life is long. If the predicted life is long enough, these components can also be

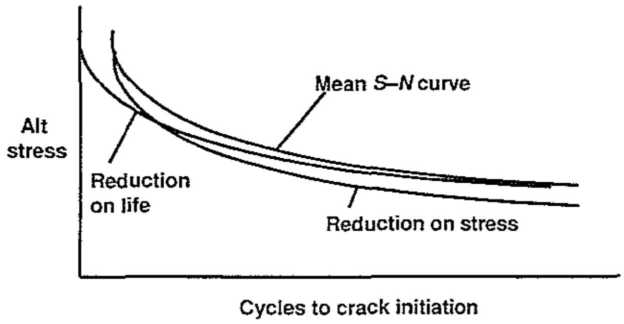


Figure 7-4. Working curve reduction factor.

considered as on-condition or unlimited life parts with no retirement time specified. Retirement times from 20,000 to 100,000 hours have been used and accepted to define unlimited life. The hours selected for unlimited life are presented by each manufacturer and approved by the certifying agency.

8. CONCLUDING REMARKS

This section presents concluding remarks based on the material presented in the paper. This paper is considered to be the consensus opinion of the US and European helicopter companies.

1. Close adherence to the procedures presented in this paper for rotorcraft fatigue and damage tolerance design and certification are recommended. It is recognized that both fatigue and damage tolerance methodologies require judgement and interpretation by the analyst. Thus deviations in certain instances may be appropriate.

2. The material in this paper may influence future regulatory changes.

3. The rotorcraft industry transition to damage/flaw tolerance should proceed cautiously while retaining the safe-life methods that have served it well for more than 50 years. A roadmap depicting this transition together with a historical perspective is presented in Figure 8-1.

4. More pertinent research and development on damage/flaw tolerance is needed. The ongoing NRTC Damage Tolerance Project and similar R&D projects should be used as the vehicle for such research, to include crack growth data, crack growth software, threshold investigation, etc.

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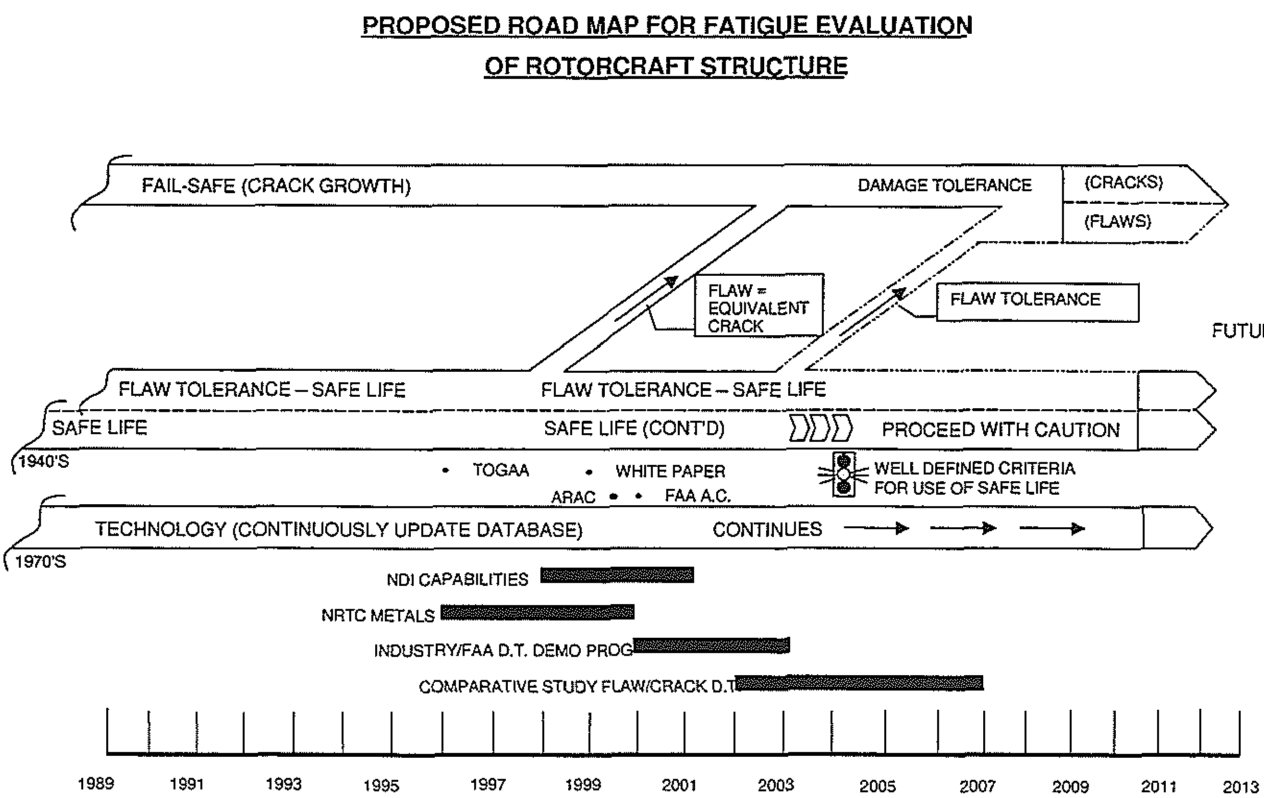

Figure 8-1. Proposed Roadmap for Fatigue Evaluation of Rotorcraft Structure