This new AC section is final and approved for inclusion in AC 29-2C. This AC section is effective and will be incorporated into the next published change or revision to AC 29-2.

AC 29.571B. § 29.571 (Amendment 29-55) FATIGUE TOLERANCE EVALUATION OF METALLIC STRUCTURE.

a. Purpose. This advisory material provides an acceptable means of compliance with the provisions of § 29.571 Amendment 29-55 of the FAA regulations dealing with the fatigue tolerance evaluation of transport category rotorcraft metallic structure. This guidance applies to conventional metallic materials. (Corresponding guidance for composite structure can be found in AC 29–2C MG 8, supplemented by AC 20-107B. Note: once § 29.573 is effective, AC 29.573 will be the current guidance for composite structures.) The fatigue evaluation procedures outlined in this advisory material are for guidance purposes only and are neither mandatory nor regulatory in nature. Although a uniform approach to fatigue tolerance evaluation is desirable, it is recognized that in such a complex area, new design features and methods of fabrication, new approaches to fatigue tolerance evaluation, and new configurations may require variations and deviations from the procedures described herein. It should be noted that § 29.571 requires that the methodology used by the applicant be approved by the FAA to assure compliance with the regulatory requirements.

b. Special Considerations. The unique performance capabilities of rotorcraft and their typical operational environment make fatigue tolerance evaluations both complex and critically important. Due to the many rotating elements inherent in their design, rotorcraft structures are potentially subject to damaging cyclic stresses in practically every regime of flight. The complexity of the fatigue loading is compounded by the fact that rotorcraft are highly maneuverable and are utilized for many widely varying roles. Corrosion and other environmental damages are not uncommon in rotorcraft operations; neither are inadvertent damages from maintenance that is typically frequent and intensive. For these reasons, special attention should be focused on the fatigue tolerance evaluation of rotorcraft structure.

c. Background.

(1) Fatigue of rotorcraft dynamic components was first addressed in the 1950’s by means of a Safe-Life methodology. The application of this methodology, as described in AC 27-1B MG 11, has proven to be successful in providing an adequate level of reliability for transport category rotorcraft. However, it was recognized in the 1980’s that higher levels of reliability might be realized by taking into account the fatigue strength-reducing effects of damage that experience has shown can occur in manufacture or in operational service. The introduction of composites led the manufacturers and regulatory authorities to develop a robust Safe-Life methodology by taking into account the specific static and fatigue strength-reducing effects of aging, temperature, moisture absorption, impact damage, and recognition of an accepted

industry standard. Furthermore, where clearly visible damages resulted from impact or other sources, inspection programs were developed to maintain safety. In parallel, crack growth methodology has been successfully used for solving short-term airworthiness problems in metallic structures of rotorcraft, and as the certification basis for civil and military transport aircraft applications. These advances in design, analytical methods, and industry practices made it feasible to address certain types of damage, which could result in fatigue failure. Consistent with this, the regulatory requirements of § 29.571 were substantially revised by Amendment 28. While many years have passed since its introduction, Amendment 28 has had little exposure to use for certification of completely new rotorcraft designs. However, the general understanding of rotorcraft fatigue tolerance evaluation has developed considerably in the interim and an additional amendment was determined to be appropriate. The latest Amendment 29-55 of part 29 and the associated revisions to advisory material were introduced to improve the currency and understanding of the rule and clarify the differing approaches and methods available for accomplishing fatigue tolerance evaluation of rotorcraft metallic structure.

(2) This guidance provides material with respect to the fatigue tolerance requirements for metallic structure and is supplemented by AC 27-1B MG 11 for evaluations using the Safe-Life methodology and other general fatigue considerations.

d. Introduction.

(1) Definitions. The following definitions are applicable when used within the context of this guidance material.

(i) As-manufactured structure is a structure that passes the applicable quality control process and has been found to conform to an approved design within the allowable tolerances.

(ii) Barely Detectable Flaw (BDF) is the worst-case flaw that is expected to remain on the structure for its operational life.

(iii) Catastrophic failure is an event that could prevent continued safe flight and landing.

(iv) Clearly Detectable Flaw (CDF) is the worst-case detectable flaw that would not be expected to remain in place for a significant period of time without corrective action.

(v) Damage is a detrimental change to the condition of the structure or assembly. In the context of this guidance material it is used as a generic term to describe all types of flaws including those caused by environmental effects and accidental damage arising in manufacture, maintenance or operation.

(vi) Damage Tolerance is the attribute of the structure that permits it to retain its required residual strength without detrimental structural deformation for a period of un-repaired use after the structure has sustained a given level of fatigue, corrosion, accidental, or discrete source damage.

(vii) Discrete flaw is a flaw that is not inherent in the design and is caused by an external action, such as corrosion, scratches, gouges, nicks, fretting, wear, impact, and potentially cracks initiated by fatigue.

(viii) Fatigue is a degradation process of a structure subject to repeated loads that may involve four phases (e.g., nucleation of many micro-cracks, coalescence of some micro-cracks to one major macro-crack, stable crack growth, unstable crack growth) and immediate failure. The boundaries between these phases are, in practice, not always easily defined. Crack initiation methods (e.g., using the S-N curve and the Miner's Rule) are generally used to address the first two phases. Linear Fracture Mechanics methods (e.g., using da/dn - ΔK and fracture toughness data) are generally used for the latter two phases.

(ix) Fatigue Loads are repeated loads, which induce a repeated variation of stress versus time in a structure.

(x) Fatigue Tolerance is the ability of a structure, either in an as-manufactured or damaged condition, to tolerate specified operational loading for a given period of use without initiating cracks, and assuming they initiate, tolerate their growth, without failure, under specified residual strength loads.

(xi) Flaw is an imperfection, defect, or blemish and may be either discrete or intrinsic.

(xii) Inspection interval is the maximum period of usage allowed for a structure between inspections. At the end of this period, the structure is inspected and if there is no damage detected, the structure may be returned to service for another inspection interval.

(xiii) Intrinsic flaw is a flaw that is inherent in the design and manufacture of the part, situated within it or peculiar to it, such as inclusions, cracks, forging laps, or porosity.

(xiv) Limit Loads are the maximum loads to be expected in service, as defined in § 29.301(a).

(xv) Multiple Load Path is identified with a redundant structure of multiple and distinct elements, in which the applied loads would be safely redistributed to other load carrying members after complete failure of one of the elements. These may be Active, where two or more elements are loaded during operation to a similar load spectrum, or

Passive, where one or more of elements of the structure are relatively unloaded until failure of the other element(s).

(xvi) Principal Structural Elements (PSE) are structural elements that contribute significantly to the carrying of flight or ground loads and the fatigue failure of which could result in catastrophic failure of the rotorcraft.

(xvii) Residual Strength is the level of strength retained by a structure with damage present.

(xviii) Retirement (Replacement) Time of a component is that number of events such as flight hours or landings at which the part must be removed from service regardless of its condition.

(xix) Safe-Life is the number of events, such as flight hours or landings, for a structural component during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue damage initiating cracks.

(2) General. The objective of fatigue tolerance evaluation is to prevent catastrophic failure of the structure by mitigation of the effects of damage in combination with fatigue throughout the life of the rotorcraft.

(i) Fatigue tolerant design as substantiated by fatigue tolerance evaluation methods such as those outlined in this guidance is required for all PSE's, unless it entails such complications that an effective structure that is tolerant to damage cannot be achieved within the limitations of geometry, inspectability, or good design practice. In such cases, the particular type of damage at issue must be identified and alternative measures should be taken to minimize both the risk of acquiring that damage and its consequences.

(ii) To perform an evaluation first requires an understanding of the potential threats (resulting in damage) that may modify the fatigue behavior of the component. The principal concerns of this guidance are consideration of all damage sources and of the fatigue loads and rotorcraft usage. Further mitigation of the sources of damage may be achieved by adoption of a critical parts plan to help ensure that the condition of the part remains as envisaged by the designer throughout its life cycle (see § 29.602).

(iii) The need for the use of complex inspection techniques or equipment or highly trained personnel (resources that may not be available to the small operator or in remote areas of operation) should be considered when establishing the methodology. When inspections cannot be relied upon for detection of small cracks or other damage, then retirement times must be established that account for the probable types and locations of the damage, including consideration of cracks.

(iv) A retirement time should be provided for all components, including those subject to inspection, whose fatigue behavior is not reliably established to a point well beyond the life of the rotorcraft. This is intended to prevent the continued use of components beyond the point that ultimate load capability may no longer be assumed to exist in the rotorcraft due to the onset of fatigue cracking. This is particularly important for single load path components or a structure prone to widespread fatigue damage.

(v) Experience with the application of methods of fatigue tolerance evaluation indicates that a relevant test background should exist in order to achieve the design objective. It is general practice within industry to conduct tests to obtain design information and for certification purposes. Damage location, fatigue characteristics, and crack growth data based on test results and service history of similar parts, if available, should be considered when establishing inspections and retirement times. The FAA should agree upon the extent of supporting evidence necessary for each phase of the evaluation process outlined below.

(3) Essential Considerations. In order to satisfy the requirements of § 29.571, consideration should be given to the following issues in order to demonstrate compliance.

(i) Selection of PSE. All structure, structural elements, and assemblies, the failure or undetected failure of which could result in catastrophic failure of the rotorcraft, should be identified as PSE [see paragraph f.(2)]. To do this, a failure mode and effects analysis or similar method may be used. Specific areas of interest within the PSE that may require particular attention include the following:

(A) irregularly shaped parts, or those containing numerous or super-imposed fillets, holes, threads, or lugs;

(B) parts of unique design for which no past service experience is available;

(C) new materials or processes for which there is no previous experience;

(D) bolted or pinned connections;

(E) parts subject to fretting;

(F) complex casting; and

(G) welded sections.

(ii) In-flight measurement to determine the loads or stresses (steady and oscillatory) for the PSEs in all critical conditions throughout the range of limitations in § 29.309 (including altitude effects), except that maneuvering load factors need not exceed the maximum values expected in operations. See paragraph f.(3).

(iii) Loading spectra as severe as those expected in operation including external load operations, if applicable, and other high frequency power cycle operations. See paragraphs f.(3) and f.(4).

(iv) A threat assessment of probable damage, including a determination of the probable locations, types, and sizes should be performed. In particular, the assessment should include an evaluation of the details of the specific work processes used on each component, operational environment, and maintenance practices to determine the potential for damage. See paragraph f.(5).

(v) Inspectability of the rotorcraft, inspection methods, and detectable flaw sizes should be compatible with the chosen fatigue tolerance methods and validated by trials conducted under realistic conditions. See paragraph f.(6).

(vi) For each PSE, one or more fatigue tolerance methodologies should be selected to ensure each specific damage resulting from the threat assessment is addressed and to satisfy the requirement for inspections and retirement times as discussed in paragraph e. of this guidance. The fatigue tolerance characteristics (including variability) of the structure and materials therein should be evaluated as necessary to support the evaluation. Generally, this will include understanding the fatigue strength, fatigue crack propagation characteristics of the materials used, and the structure and the residual strength of the damaged structure. See paragraphs e., f.(7) and f.(8).

(vii) Fatigue Tolerance Results of the evaluation should be used to provide data in the Limitations Section of the Instructions for Continued Airworthiness. See paragraph f.(9).

e. Fatigue Tolerance Evaluation. A fatigue tolerance evaluation, by analysis and tests, of the PSE is required to establish inspections and retirement times, or approved equivalent means, to avoid catastrophic failure due to fatigue cracking during the operational life of the rotorcraft. The evaluation should consider the impact of the probable threats identified on the fatigue performance and residual strength of all critical areas of each PSE. A number of different fatigue evaluation methods have evolved over the years. Seven of these methods are recognized and discussed in detail in this guidance. The seven methods are summarized as a table in Figure AC 29.571B-1. Also noted in the table is the safety management strategy the specific method supports, the analysis category in which they belong, and whether the specific method can be used to address the types of damage identified in the threat assessment.

(1) Each approach results in information that can be used to support establishment of retirement times or inspection requirements. Four methods are used to support safety-by-retirement strategies and they result in retirement times. The other three methods are used to support safety-by-inspection strategies and the result is in-service inspection requirements.

(2) In some cases, application of one method may be sufficient to achieve acceptable fatigue tolerance. In other cases more than one method may be needed. For example, use of Safe-Life Retirement in combination with Crack Growth Inspections could be an effective way to manage fatigue due to all possible sources.

(3) All the methods listed, with the exception of Safe-Life Retirement, were developed to explicitly address some level of damage. All the methods can theoretically be implemented analytically or by test. However, some of the methods are more practically implemented analytically and some are best implemented by test.

METHOD	PARAGRAPH	STRATEGY	ANALYSIS CATEGORY	THREAT ASSESSMENT RESULTS
Safe-Life Retirement	e.(6)(i)(A)	Retire	Crack Initiation	Not Included
Safe-Life Retirement with BDF(s)	e.(6)(i)(B)	Retire	Crack Initiation	Not Including Cracks
Safe-Life Retirement with CDF(s)	e.(6)(i)(C)	Retire	Crack Initiation	Not Including Cracks
Safe-Life Inspection for CDF(s)	e.(6)(i)(D)	Inspect	Crack Initiation	Included
Safe-Life Inspection for a failed element	e.(6)(i)(E)	Inspect	Crack Initiation	Included if Considered for all Elements
Crack Growth Retirement	e.(6)(ii)(A)	Retire	Crack Growth	Included if Crack Bounds Damage
Crack Growth Inspection	e.(6)(ii)(B)	Inspect	Crack Growth	Included

Figure AC 29.571B-1. Seven Fatigue Evaluation Methods discussed in this guidance

(4) From an analytical standpoint, these methods fall into one of two categories, crack initiation or crack growth. Each of the seven methods is briefly described below in paragraphs e.(6)(i) and e.(6)(ii), depending on the category.

(5) In-service experience may be used to support establishing fatigue tolerance characteristics when it is shown on a similar structure.

(6) Fatigue Evaluation Methods.

(i) Crack Initiation Methods. The methods described in this section are categorized as crack initiation methods since they involve quantifying the time it takes for a crack to initiate at a critical area in an as-manufactured part or at a critical area that has sustained some level of damage. Analytically these methods depend on fatigue data (e.g., stress versus number of cycles (S-N) curves) and cumulative fatigue damage algorithms (e.g., Miner's Rule) to establish a high margin retirement time. Testing that supports these methods employs specimens that are as-manufactured or ones that have been preconditioned with damage as identified in the threat assessment.

(A) Safe-Life Retirement. Safe-Life Retirement is a crack initiation method that accounts for damage induced by fatigue loading but does not account for flaws and defects due to manufacturing and in-service conditions. Application of this method results in a replacement time based on the time to initiate a crack in an as-manufactured part. Analysis or tests may be used to determine the crack initiation life. The rationale behind this method is based on part replacement before the probability of initiating a crack becomes significant. This method needs to be supplemented by other methods to account for damage. For compliance details, see paragraph f.(7)(i).

(B) Safe-Life Retirement with a Barely Detectable Flaw (BDF). Safe-Life Retirement with a BDF is a crack initiation methodology that explicitly addresses the effect of damage that is considered barely detectable and is therefore likely to go unnoticed for the life of the part. Application of this method results in a replacement time based on the time to initiate a crack from a BDF. Analysis or tests may be used to determine the crack initiation life. The rationale behind this method is based on part replacement before the probability of initiating a crack is significant. Damage in excess of the BDF must be addressed using other methods. For compliance details, see paragraph f.(7)(ii).

(C) Safe-Life Retirement with a Clearly Detectable Flaw (CDF). Safe-Life Retirement with a CDF is a crack initiation methodology that explicitly addresses the effect of damage that is considered clearly detectable but conservatively recognizes that it would remain in place without corrective action prior to the retirement time of the part. Application of this method results in a retirement time based on the time to initiate a crack from a CDF. Analysis or tests may be used to determine the crack initiation life. The rationale behind this method is based on part replacement before the probability of initiating a crack is significant. Use of this method by itself could achieve acceptable fatigue tolerance and may preclude the need for any mandated directed inspections. See paragraph f.(7)(iii) for compliance details.

(D) Safe-Life Inspection for a CDF. Safe-Life Inspection for a CDF is a crack initiation method that explicitly addresses the effect of damage that is considered clearly detectable and would therefore not be expected to remain in place without corrective action for any significant period of time. Application of this method results in a directed inspection task with an interval based on the time to initiate a crack from a clearly detectable flaw. Analysis or tests may be used to determine the crack initiation life. The rationale behind this method is based on visual detection and disposition of the flaw before the probability of initiating a crack is significant. Damage that is not detectable must be addressed by other methods and the cumulative effects of fatigue prior to and following the advent of the damage should be considered. For compliance details, see paragraph f.(8)(i).

(E) Safe-Life Inspection for a failed element. Safe-Life Inspection for a failed element is a crack initiation method. It results in an inspection for a completely failed load path with an interval based on the crack initiation life of the adjacent structure accounting for internal load redistribution due to failure of the load path that is to be inspected. This method can only be applied if the structure is initially designed for limit load capability with the failed element. The rationale behind this method is based on visual detection and disposition of the failed load path before the probability of initiating a crack in the adjacent structure becomes significant. Therefore it may not be appropriate if the damage that has led to the failure of the first load path could similarly affect the remaining path. For compliance details, see paragraph f.(8)(iii).

(ii) Crack Growth Methods. The methods described in this paragraph are categorized as crack growth methods since they involve quantifying the time it takes a crack at a critical area to grow from some initial size to some final size. Analytically these methods depend on crack growth rate properties (e.g., da/dN vs. ΔK vs. R) and fracture properties (e.g., K_{IC}). Using these properties, Fracture Mechanics based tools are used to predict crack growth and final fracture. Testing that supports these methods employs specimens that contain cracks and involves close monitoring to document actual crack growth and final fracture.

(A) Crack Growth Retirement is a crack growth method that explicitly addresses the largest damage that could occur during manufacture or operation of the rotorcraft. This damage is modeled as a bounding equivalent crack (BEC) established based on the results of the threat assessment. Application of this method results in a retirement time based on the time for the initial crack to grow large enough to reduce the residual strength to design limit level. Since typical BECs are relatively small and thus difficult to induce in test specimens, this method is typically implemented analytically. The rationale behind this method is based on part retirement before the largest probable damage, modeled as a crack, would reduce the residual strength below design limit. Use of this method by itself could achieve acceptable fatigue tolerance and preclude the need for any mandated inspections provided all threats are accounted for by the BECs. For compliance details, see paragraph f.(7)(iv)

(B) Crack Growth Inspection is a crack growth method that explicitly addresses damage that could occur during manufacture or operation of the rotorcraft. An in-service inspection method is selected that defines a detectable crack size, which could be as large as a completely failed load path. An inspection interval is established based on the time for the detectable crack to grow to critical size or for the residual strength of the adjacent structure to drop to design limit due to continuing crack growth in it. This method is applicable to single or multiple load path structure and inspection for a completely failed load path or less. This method may be addressed by analysis supported by test depending on the difficulty of introducing into the specimen the inspectable crack or failed load path. The rationale behind this approach is based on detection and disposition of a crack or failed load path before residual strength is reduced below the design limit load. For compliance details, see paragraph f.(8)(ii).

f. Means of Compliance.

(1) GENERAL. The results of the fatigue tolerance evaluation required by § 29.571 are used to establish operational procedures that are meant to minimize the risk of catastrophic failures during the operational life of the rotorcraft. It is required that the evaluation performed considers the effect of damage that could result from potential threats present during manufacture and operation. An assessment of probable threats is required to identify the damage that must be considered in the fatigue tolerance evaluation.

(i) The fatigue tolerance evaluation should establish both retirement times and inspection intervals, or approved equivalent means, to prevent any catastrophic failures. Retirement times should be set to ensure that baseline ultimate strength capability is not compromised for as-manufactured structures and structures where the damage is likely to be undetected during the operational life. Intervals for inspections for detectable damage must be established so that strength capability will never fall below maximum design limit level. The intent is that if damage does occur, the structure will retain the capability to withstand reasonable loads without catastrophic failure or excessive structural deformation until the damage is detected and the structure is replaced or repaired. If inspections cannot be established within the limitations of geometry, inspectability, or good design practice, then supplemental procedures, when available, should be established that would minimize the risk of damage being present or leading to a catastrophic failure.

(ii) The following considerations will assist the successful design of a fatigue tolerant structure.

(A) Use multiple-element and multiple load path construction with provisions for crack stoppers that can limit (arrest) the growth of cracks while maintaining adequate residual strength.

(B) Select materials and stress levels that preclude crack growth or crack initiation from flaws or that provide a controlled slow rate of crack propagation combined with high residual strength after initiation of cracks. Test data should substantiate material properties.

(C) Design for detection of damage (i.e., cracks and flaws) and retirement or repair.

(D) Provide provisions that limit the occurrence of damage and the probability of concurrent multiple damage, particularly after long service.

(iii) Section 29.571 requires that the applicant's proposed compliance methodology must be submitted and be approved by to the Administrator. Therefore, the applicant should coordinate the involvement of the FAA from an early stage. The proposed means of compliance should include the following items.

(A) A list of PSEs to be evaluated.

(B) The results of threat analyses for each PSE including type, location, and size of the damage that will be considered in order to establish retirement times, inspections, or other procedures.

(C) Inspection criteria that includes an estimate of detectability or inspectability, along with any supplemental procedure to minimize the risk of damage.

(D) The analysis methods and supporting test data that will establish retirement times, inspections, or other procedures.

(2) IDENTIFICATION OF PSE. The fatigue tolerance evaluation should first consider all airframe structure and structural elements, and assemblies in order to identify the PSE. The structural elements and assemblies identified as PSE should be formally submitted to the FAA with justification for the PSE based on good design practice, service history with similar structure, drawing reviews, static analysis issues, or other appropriate means.

(i) A Failure Mode and Effects Analysis or similar method may be used to identify structures whose failure due to fatigue can lead to catastrophic failure of the rotorcraft. The need to design a PSE for fatigue tolerance when they are supplied by third parties (e.g., actuators) should be clearly identified in the rotorcraft manufacturer’s specification for the part. The list of PSE will likely include structural elements and assemblies that will be subjected to significant fatigue loading expected during the operational life of the rotorcraft. This may include the following rotorcraft parts:

(A) Rotors: blades, hubs, hinges, attachment fittings, vibration dampening devices;

(B) Rotor drive systems (parts connecting rotors to engines): gears, shafts, gear housings, couplings;

(C) Rotor control systems: actuators, pitch control system, swashplate, servo flaps;

(D) Fuselage (airframe): rotor system support structure, landing gear attachment;

(E) Fixed and movable control surfaces: stabilizer;

(F) Engine, transmission or equipment mountings: APU, auxiliary gearbox;

(G) Landing gear;

(H) Folding systems: main blade, tail beam.

(ii) Analyses and fatigue tests on complete structures or representative sub-element structures can determine the locations within PSE that need to be identified for fatigue tolerance evaluation. The following should be considered:

(A) Strain gauge data on undamaged structure that can identify high stress points.

(B) Analysis that shows high stress or small margin of safety values.

(C) Locations where permanent deformation occurred in static tests.

(D) Locations where failure has occurred in as-manufactured structure fatigue tests.

(E) Locations where the potential for fatigue damage has been identified by analysis.

(F) Locations where the maximum allowed stress occurs when an adjacent element fails.

(G) Locations in structure needed to maintain adequate residual strength that has high stress concentration values.

(H) Locations where detection would be difficult.

(I) Locations where service experience with similar components indicates potential for fatigue or other damage (e.g., fretting, corrosion, wear).

(3) FLIGHT LOADS MEASUREMENT PROGRAM. The simulation of expected spectrum loads for each PSE should be based on flight recorded strain gauge data collected as part of a structured flight test program. The PSE spectrum loads include the steady state, transient, and vibratory loads that are expected in operation. AC 27-1B MG 11, provides further detail for development and use of flight measured loads as the basis for spectrum loads used in the fatigue tolerant evaluations.

(4) ROTORCRAFT USAGE SPECTRUM.

(i) The usage and loading spectrum should be developed so that it is unlikely that the actual usage and loads will cause fatigue damage or crack growth rates beyond those associated with the defined spectrum used in the fatigue tolerance evaluation. The usage spectrum allocating percentage of time or frequencies of occurrence to flight conditions or maneuvers should be based on the expected usage of the rotorcraft. Considerations should include flight history, recorded flight data, design limitations established in static strength requirements, and recommended operating conditions and limitations specified in the rotorcraft flight manual.

(ii) The fatigue load spectrum developed for fatigue testing and analysis purposes should be representative of the anticipated service usage. Low amplitude load levels that can be shown not to contribute fatigue damage may be omitted (truncated). Simplification of the spectrum loads may also include summing (binding) of percent times or cycles with common steady and vibratory load values.

(iii) The steady state, transient, and vibratory flight load assigned to each regime in the spectrum and utilized in the fatigue tolerance evaluations for each condition should take into account combinations of altitude, center of gravity (CG), gross weight (GW), airspeed, etc., considered to be representative of expected GW/CG mission configurations.

(iv) The usage spectrum should be presented to the FAA for their concurrence. It should include normal operation over the range of rotorcraft configurations including a percent time under ‘external load’ conditions. This spectrum should represent a “composite worst-case” compilation that includes all of the critical conditions that the rotorcraft is expected to experience during performance of the design missions.

(v) AC 27-1B MG 11, provides further detail for the development of the usage spectrums used in the fatigue tolerance evaluations.

(5) THREAT ASSESSMENT.

(i) A determination should be made of all potential threats that could occur during the manufacturing and service life that may cause damage to each PSE. A threat assessment should be performed for each PSE. To acquire sufficient knowledge of the component and of its global environment, the following items must be identified:

(A) manufacturing process

(B) quality control process

(C) prescribed storage, transport, handling, assembly and maintenance aspects of the component, and of the surrounding components

(D) operational environment

(E) potential for corrosion including that from contamination by corrosive fluids

(F) potential for impact damages from debris, dropped tools, hail, tramping underfoot during maintenance, etc.

(G) potential for wear

(ii) To determine types, locations, and sizes of the probable damages, considering the time and circumstances of their occurrence, the following should be considered:

(A) Intrinsic flaws and other damage that could exist in an as-manufactured structure based on the evaluation of the details and potential sensitivities involved in the specific manufacturing work processes used.

(B) Damage that could be expected to occur during prescribed activities associated with storage, transport, handling, assembly, maintenance, overhaul, repair and operation of the component and of the surrounding components including impacts, scratches, fretting, corrosion, contamination, wear, and loss of bolt torque.

(C) Previous experience and data collected on similar events and on similar components; materials, and processes should be considered in identifying risks and causes of damages and their effects in inducing flaws or cracks.

(D) Metallurgical evaluations, manufacturing records and overhaul and repair reports, field service reports, incident and accident investigations, and engineering judgment may be used as supporting data.

(E) When data are not available, the threat should be experimentally simulated and the effect established through tests and analysis. With agreement of the FAA, an upper cut-off value may be established for each class of damage.

(F) Credit may be given to manufacturing, transport, handling, installation, and maintenance instructions finalized to minimize or avoid damages. Examples of these processes or instructions could be: "frozen manufacturing processes," Flight Critical Parts programs, material selection to mitigate intrinsic flaws like inclusions and defects, procedures to reduce deviations from nominal structures, etc.

(G) Credit may be given to protection of structures, such as the use of protective coatings, shielding and plating against corrosion, fretting, and impacts.

(H) Critical areas will be assumed as a typical location of the damage, unless proper justification is provided to limit the applicability to specific areas or sections of the part.

(iii) Classification of Damage.

(A) The results of the threat assessment are used to classify the damage used in the fatigue tolerance evaluation. The process employed to classify the damage will depend on the fatigue tolerance evaluation method to be used. Depending on the method, a BDF, a CDF, a BEC, or an initial inspectable crack must be established.

(B) For each damage type identified, the sizes to be considered should be representative of the maximum sizes that might not be detected by the inspection techniques established for the component. Sizes exceeding those that are likely to occur do not need to be considered. Standard sizes of damage or standard level of aggression may be derived from previous experience. Each applicant will be required to present justification for damage and crack sizes to be used in the fatigue tolerance evaluations. Within the operational life, defect sizes that have been found in service should be correlated with the sizes used in the design certification.

(C) Barely Detectable Flaw (BDF). For retirement time analysis, flaw sizes that are “barely detectable” may be used to conservatively represent the worst case of undetectable flaws. Alternatively, when the detectable size is larger than the one identified by the threat assessment, a smaller size, but one not less than the flaw size likely to occur, can be used. Sometimes an “allowable” detectable size is established as acceptable for a specific manufacturing process, such as castings, to remain in place for the life of the structure. When it is impossible to simulate that maximum allowable size in the test specimen, the sizes available in the specimen may be used, provided the subsequent analysis of the test result conservatively accounts for the shortfall in the damage size.

(D) Clearly Detectable Flaw (CDF). For inspection intervals, flaw sizes that are “clearly detectable” may be used. The largest discrete size of a CDF to be considered may be limited to the maximum size of the CDF that is likely to remain in place for a significant period of time and not be detected during routine inspections for general conditions and normal observations by knowledgeable personnel. The damage size used may be limited to the maximum probable size identified in the threat assessment. For multiple load path structure, the number of failed load paths to be considered should be established.

(E) Bounding Equivalent Crack (BEC). A Bounding Equivalent Crack must be defined to determine a retirement time using the Crack Growth Retirement method. The size of the BEC should bound the life reducing effect of damage that could occur as a result of manufacturing, maintenance, or the service environment. The size may be established by analytical back calculations from coupon or service fatigue life data accounting for material variability effects in the data. In any case, there should be no probable damage from any source that would lead to failure of the part in less time that it would take the BEC to reach critical size. Each applicant must justify the BEC sizes used in the analysis; however, there has been some limited experience that indicates that the following BEC sizes could be appropriate.

(1) 0.015 inch or 0.380 mm radius semicircular surface crack for precision-machined mechanical parts

(2) 0.050 inch or 1.270 mm radius quarter-circular corner crack in fastener holes for typical aluminum airframe structure

(F) Initial Inspectable Crack. The size and shape of the initial inspectable crack (a_{DET}) must be established when the Crack Growth Inspection approach is used. The inspection interval is based on the time for the initial inspectable crack to grow to a size (a_{CRIT}) that would result in catastrophic failure of the rotorcraft if limit loads were applied. The initial inspectable crack is a function of the inspection method that is used to detect it. Regardless of the inspection method, the probability of detecting this size crack should be high and it should be substantiated.

(6) INSPECTABILITY AND INSPECTION METHODS. This section provides guidance on selecting and substantiating damage detection methodology for use with the methods of paragraphs f.(8) (Inspection Intervals) and f.(10) (Approved Equivalent Means). The methods of paragraph f.(8) can result in a mandated inspection program that must be included in the Airworthiness Limitations Section (ALS) of the Instructions for Continued Airworthiness in accordance with § 29.1529 of the regulatory requirements. Qualified personnel must conduct these inspections at the specified interval using the approved method or methods. Additionally, § 29.571 allows that substantiation may be accomplished by “Approved Equivalent Means,” which is discussed in paragraph f.(10). These Approved Equivalent Means may include actions that detect damage or flaws indirectly, and are substantiated using the methods of paragraph f.(8). These actions should be shown to be reliable and systematically conducted by knowledgeable personnel. The following are considerations for establishing inspections, inspection methods, or indirect damage detection.

(i) Inspectability. The ease of conducting an inspection should be a design goal for principal structural elements. Design features such as open construction, access panels or ports, or other easy access to fatigue critical areas for needed inspections should be considered. A design that requires disassembly in order to conduct a required inspection, other than during a scheduled maintenance disassembly, should be avoided.

(ii) The specific inspection methods that are used to accomplish fatigue substantiation should be:

(A) Compatible with the threats identified in the threat assessment, paragraph f.(5), and provide a high probability of detection in the threat assessment and their development, under the operational loads and environment.

(B) Consistent with the capabilities, facilities, and resources of the potential operators of the helicopter. The need to conduct complex or difficult field-level inspections should be avoided, especially when the projected usage of the helicopter may include extended periods of operation in remote areas.

(C) Developed and substantiated for each specific application by means of a full-scale test program, or by experience with similar methods in similar applications.

(D) Included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness in accordance with § 29.1529 as required by § 29.571(g).

(iii) Detectable Damage Size Assessment.

(A) In the case where the substantiation is predicated on the detection of a specific flaw or crack size, an assessment should be conducted to assure that the selected inspection method would be highly reliable in detecting that size of damage in service. This assessment may be based on the known capability of currently available inspection methods and equipment, provided that this capability is verified by a full-scale test program or by experience with the method in service for similar structure and damage.

(B) If the current capability of a specific inspection method is in question, or if the capability of a specific method needs to be extended to a smaller damage size, then a systematic assessment and substantiation of the method for the intended purpose is appropriate. This assessment could include the determination of the Probability of Detection (POD) as a function of damage size and should consider the capabilities of the potential operators of the helicopter and the environment in which the inspections will be conducted.

(iv) Indirect Detection of Damage. Several damage detection procedures are available that could be used as “Approved Equivalent Means” to support substantiation of a structure [reference paragraph f.(10)]. These procedures, if systematically required and conducted by knowledgeable personnel, can be used in conjunction with the methods presented in paragraph f.(8) to achieve the substantiation. Examples of this type of substantiation are:

(A) In-flight damage detectable by vibration, noise, or observing a blade-out-of-track tip path plane. Consideration should be given to the background levels of noise and vibration, as well as whether the indication is of a different character (more detectable) rather than just a change in level (less detectable).

(B) Damage that is obvious in a preflight check or routine visual examination. This could include obvious flaws or cracking, but also could include structure that is found to be loose, broken, or soft when deflected by hand. Other obvious damage detection could include fluid leaks, missing fasteners, structure bent or out of alignment, or jamming of mechanical parts.

(C) Damage that is indicated following flight completion. Spectrographic oil analysis would be an example.

(D) Damage detection by automated means. This includes crack detection by foil, fiber, or wire break, load monitoring (to detect a change in internal load distribution), acoustic emission monitoring, or other on-board sensors that meet the goals of damage detectability and reliability.

(7) RETIREMENT TIMES. Each of the four methods below provides a means to establish a retirement time for each PSE. The determination of the fatigue tolerance characteristics should include an assessment using the conventional Safe-Life methodology. In addition, this serves as a baseline for comparison to retirement times determined with flaws and defects included, and should be used as the structure's retirement time if it is the lowest calculated time.

(i) The conventional Safe-Life methodology accounts for damage induced by fatigue loading but does not account for flaws and defects due to manufacturing and in-service conditions. If the retirement time is established using this method, then the damage identified in paragraph f.(5) (as required by § 29.571(d)(iii)) must be addressed by inspections or other equivalent means. Information to guide a fatigue evaluation based on a conventional Safe-Life approach is provided in detail in AC 27-1B MG 11. The method consists of:

(A) Establishing mean fatigue curves (e.g., stress-life or strain-life) based on crack initiation in constant-amplitude or spectrum testing of as-manufactured structure;

(B) Establishing working fatigue curves with strength and life margins; and

(C) Conducting a cumulative damage working life calculation using known flight loads and estimated usage.

(ii) A Safe-Life retirement time substantiation with BDF provides a safe period of operation of a structure with probable flaws that may remain in place without detection for that period. Barely detectable flaws are intended to conservatively represent a worst-case of undetectable flaws. The substantiation is accomplished by testing and analysis employing conventional Safe-Life methodology except that an intrinsic and discrete critical flaw in critical locations on the structure is considered. It should be noted that this method, since it is a Safe-Life (crack initiation) method, is not appropriate for use when the flaw being considered is already a crack.

(A) The types, sizes, and locations of flaws to be considered are determined by the threat assessment (paragraph f.(5)). These flaws may be represented by "equivalent flaws" if it is demonstrated that they have the same or a more severe strength-reducing effect than the corresponding representative flaws.

(B) The mean fatigue strength of the structure with flaws may be determined by one of the following three methods:

(1) Testing a full-scale structure with flaws:

(i) Representative flaws as determined by the threat assessment, or equivalent flaws if substantiated, are imposed at the critical locations on the structure where flaws are likely to occur.

(ii) S-N or spectrum safe-life fatigue testing is conducted; see paragraph e of AC 27-1B MG 11.

(iii) A mean S-N curve with flaws is derived directly from this data.

(2) As-manufactured structure strength modified by the effect of flaws.

(i) A mean strength for as-manufactured structure (without flaws) can be determined using full-scale S-N or spectrum safe-life fatigue testing.

(ii) The effect of flaws may be determined by analysis, by similarity to components where the effect of the flaws has previously been determined, or by a specimen test program incorporating the pertinent features of the full-scale component. Consideration should be given to the material form, geometric features, surface finish, and steady and vibratory load levels, in combination with flaws representative of those identified in the threat assessment.

(iii) The effect of the flaws is combined with the fatigue result determined on the as-manufactured structure without flaws.

(3) Analytical mean strength modified by the effect of flaws:

(i) A mean strength for as-manufactured structure (without flaws) can be determined analytically, provided that correlation with a similar design can be accomplished, or if additional conservatism is included in the working curve reductions employed in paragraph f.(7)(ii)(C).

(ii) The effect of flaws may be determined by analysis, by similarity to components where the effect of the flaws has previously been determined, or by a specimen test program incorporating the pertinent features of the full-scale component. Consideration should be given to the material form, geometric features, surface finish, and steady and vibratory load levels in combination with flaws representative of those identified in the threat assessment.

(iii) The effect of the flaws is combined with the fatigue result analytically determined for the as-manufactured structure without flaws.

(C) Working Curve Determination. Reduction factors should be applied to the mean curve determined above to derive a working fatigue curve. As outlined in AC 27-1B MG 11, working curve reduction factors should include consideration of the number of specimens tested, variability (scatter), previous test data on the same materials or similar structures, as well as service experience. Different reduction factors from those used for conventional Safe-Life methodology may be employed if appropriately justified.

(D) Retirement Time Determination. The working fatigue curve, flight loads (paragraph f.(3)), and usage spectrum (paragraph f.(4)) are used with a cumulative damage analysis such as shown in AC 27-1B MG 11, to calculate a safe retirement time.

(iii) Safe-Life Retirements with Clearly Detectable Flaws.

(A) A retirement time may also be based on flaws larger than the BDF case, up to the clearly detectable size described in paragraph f.(5), if the applicant chooses. This could be the case, for example, if it was desired to allow a specific manufacturing-related flaw of detectable size to remain in place for the life of the structure without further inspection.

(B) The substantiation for this case can be the same as described in paragraph f.(7)(ii), except that the larger flaws selected for the replacement time substantiation are used instead of the BDFs.

(iv) Crack Growth Retirement.

(A) General.

(1) This approach depends on retirement rather than inspection to ensure the continued airworthiness of a PSE. The retirement time is established based on consideration of crack growth characteristics. Fatigue with damage is addressed by

timely retirement and there are no explicit inspection requirements that are derived from this approach.

(2) This approach requires demonstration either by analysis, testing, or both, that the BEC (a_{BEC}), the most severe crack consistent with manufacturing, maintenance, and service environment, will not grow or will not grow to critical size (a_{CRIT}) under the service loading and environment before the structure is retired. The critical crack size (a_{CRIT}) is established by limit load. 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. It is recommended that full scale fatigue testing be undertaken to provide an understanding of the fatigue behavior of the component in support of the chosen methodology. In particular it ensures hot spots are identified, which experience has shown analysis often fails to identify.

(3) A threat assessment (see paragraph f.(5)) should be performed to support establishing the BEC size to be used. It is intended that the BEC conservatively bounds the most severe defect resulting from manufacturing, maintenance, or the service environment. That is, there should be no probable defect, from any source, that would lead to failure of the part in less time than it would take the BEC to reach critical size. It should be noted that the resulting crack is a mathematical expedient that may not represent a true physical crack. If the BEC is defined by analytical back calculations from coupon or service fatigue life data, it will be highly dependent on the predictive tool used (i.e., growth algorithm, material data, etc.). Therefore, the same predictive tool must be used to perform the fatigue tolerance evaluation. When the BEC is based upon test or service data, it must account for material variability in initiation and growth.

(4) To determine the retirement, the BEC should be assumed at the critical location and the crack growth characteristics should be determined for the expected load and environment spectrum. There are three different scenarios that could result from a crack growth assessment and be used for establishing a retirement time. These scenarios are illustrated in Figure AC 29.571B-2, Figure AC 29.571B-3, and Figure AC 29.571B-4.

(B) No Growth. The no crack growth scenario is illustrated in Figure AC 29.571B-2. Here the BEC does not grow when using top-of-scatter crack growth rate data. In this case the retirement time should not exceed the design service life (L_{DES}).

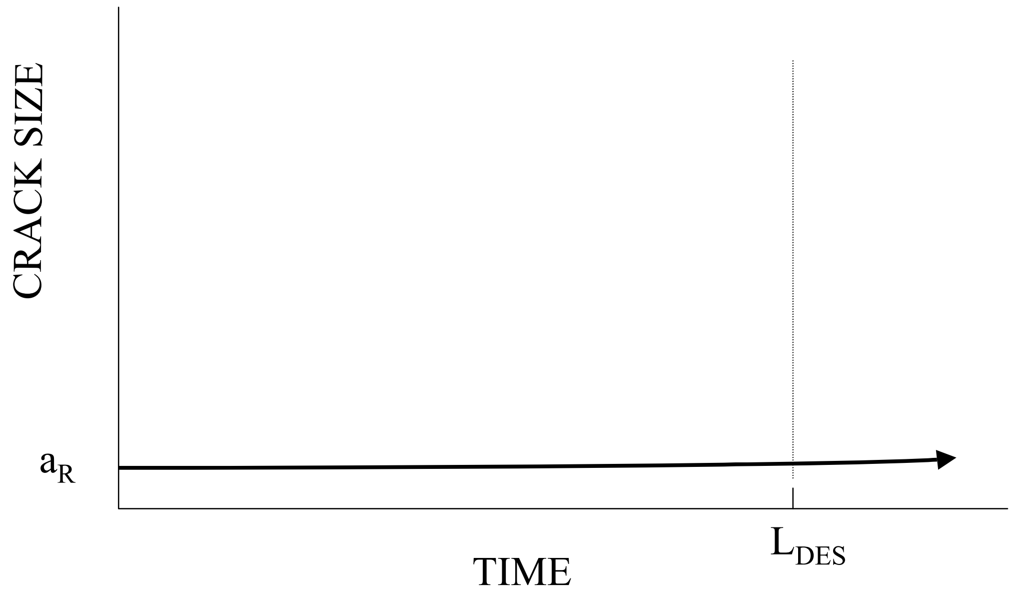

Figure AC 29.571B-2. No Growth

(C) Slow Growth of Undetectable Crack. Figure AC 29.571B-3 illustrates the scenario where the BEC grows relatively slowly but becomes critical prior to becoming detectable (a_{DET}). In this case, the retirement time should be set equal to the total crack growth life (L_T) divided by a factor N.

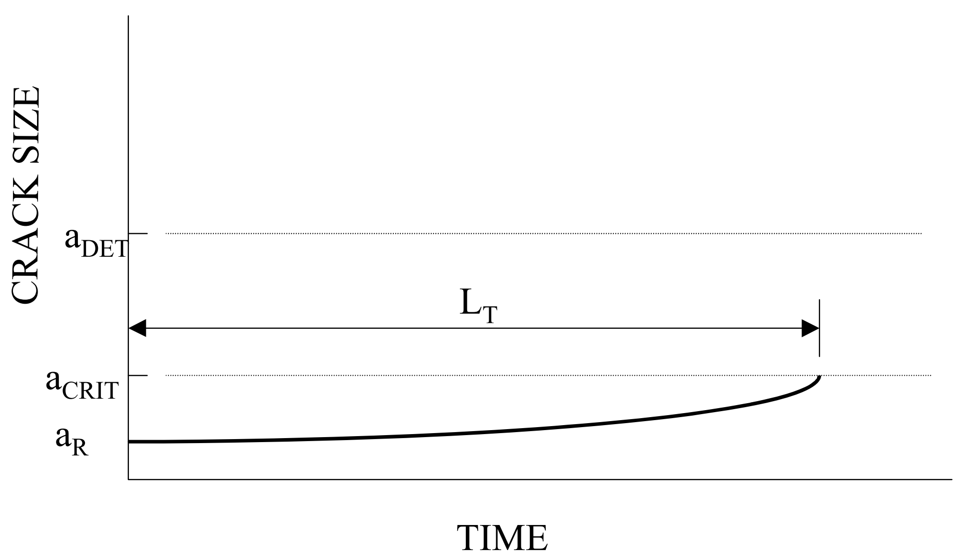

Figure AC 29.571B-3. Slow Growth of Undetectable Crack

(D) Slow Growth of Detectable Crack. Figure AC 29.571B-4 illustrates the scenario where the BEC grows to a detectable size (at L_1) before becoming critical (at L_1+L_2). In this case, the retirement time should be set equal to the total crack growth life (L_1+L_2) divided by a factor N.

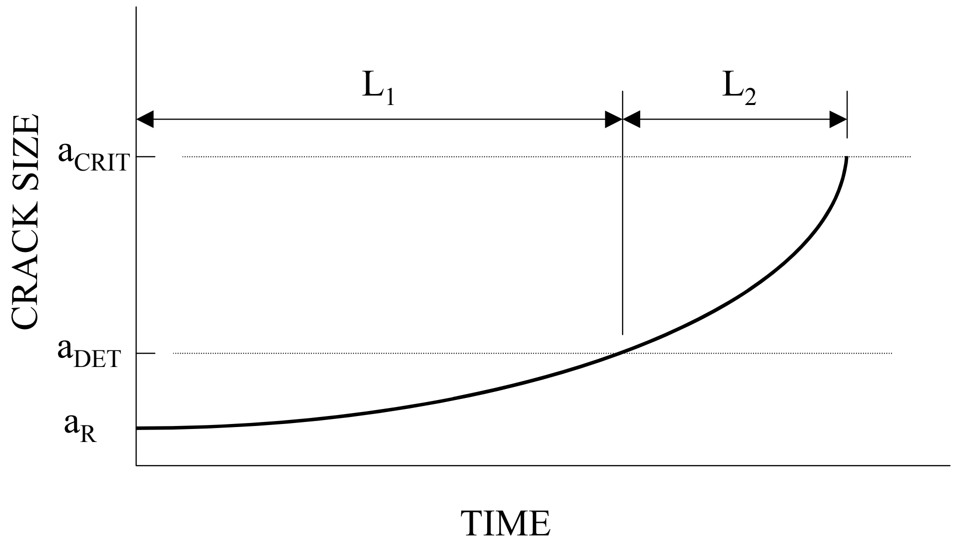

Figure AC 29.571B-4. Slow Growth of detectable Crack

(E) Life Factors for Crack Growth Retirement.

(1) In determining the factor of N to be used for determining the retirement time, consideration should be given to the crack growth data used (e.g., top of scatter data versus average data, number of specimens used to generate data, etc.).

(2) The minimum suggested N value should be N=2 in the case where the conservative top-of-scatter crack growth data are used in the crack growth analysis, or N=4 when the average crack growth data are used in the crack growth analysis, or N=4 when the crack growth life is obtained from the crack growth test of one specimen (for two or more full scale specimens, N=3 of the shortest crack growth life can be used).

(3) It should also be noted that with this approach, the validity of the crack growth threshold, ΔK_{th}, is especially important since there is no element of inspection to ensure continued airworthiness. Consistent with this, additional attention may be required for validating the crack growth threshold value(s) used in the analyses. Consideration should be given to the influence of the test procedure used to develop values, microstructure, heat treatment, crack size, loading conditions, environment, grain size and orientation, etc. In general, a coupon-testing program may be necessary to develop a consistent ΔK_{th} database and the use of published data may require additional conservatism.

(8) INSPECTION INTERVALS. Each of the following three methods provides a means to establish inspection intervals for detectable damage or detectable damage growth. The time of the first inspection should coincide with the repetitive interval established unless the applicant can substantiate an alternate time.

(i) Safe-Life Inspection for a CDF provides a safe interval of operation between repetitive inspections for the presence of probable detectable flaws. The substantiation is accomplished by testing and analysis employing conventional Safe-Life methodology except that intrinsic and discrete critical flaws are considered. The size of flaws considered should be “clearly detectable”, which is intended to be a conservative representation of detectable flaws that could remain in place for the entire interval in spite of routine inspections for general condition. It should be noted that this method, since it is a Safe-Life (crack initiation) method, is not appropriate for use when the flaw being considered is already a crack.

(A) The method described in paragraph f.(7)(iii), Safe-Life Retirements with Clearly Detectable Flaws, may be employed for this case, except that the calculated retirement time is used as a repetitive inspection interval.

(B) The repetitive inspection consists of examination of the structure for the presence of the flaw using the substantiated inspection method. If no flaw is found, the structure may be returned to service for another inspection interval period, up to the established retirement time. If the flaw is found, the structure is retired; or, if a repair procedure for the specific flaw type has been substantiated, the structure is repaired and returned to service for another inspection interval period, up to the established retirement time for the structure.

(C) Substantiation of repairs should include careful consideration as to whether undetectable cracks may now exist and whether the original certification approach is still applicable.

(ii) Crack Growth Inspection. This approach depends on detection of cracks before they become critical to ensure the continued airworthiness of a PSE. While any inspections that are capable of detecting cracks with high reliability may be used with this approach, the criteria stated in paragraph f.(6), Inspectability and Inspection Methods, should be considered in making the selection. The inspection method chosen will define the initial inspectable crack that will be used to perform the fatigue tolerance evaluation. Once the initial inspectable crack is defined, crack growth, and residual strength assessments must be performed to determine the time for the initial inspectable crack (a_{DET}) to grow to a size (a_{CRIT}) that would result in a catastrophic failure of the rotorcraft if limit loads were applied. This assessment could be theoretically done analytically or by test; however, in most cases it is performed analytically using fracture mechanics methods. The resulting life for a_{DET} to grow to a_{CRIT} is used to set the inspection interval. This general process applies to both single and multiple load path structure regardless of the level of inspection (e.g., for complete load path failure or less than load path failure in a multiple load path structure). The details of defining the interval once the crack growth life has been determined are discussed later.

(A) Single Load Path Structure. The time for a detectable crack (a_{DET}) to grow to critical size (a_{CRIT}) in a structure is denoted as L_2 in Figure AC 29.571B-4. If this were a single load path structure, the inspection interval would be established as L_2 divided by N. (See paragraph f.(8)(ii)(C) for guidance on values of N.) This interval is valid until the part is retired.

(B) Multiple Element Structure.

(1) Depending on inspectability considerations and residual life characteristics of the structure following a load path failure, it may be beneficial to take advantage of the redundancy of a multiple load path structure. On the other hand, the safety of a multiple load path structure can be managed without taking advantage of its redundancy. In this case, each load path would be considered independently and inspection intervals established for each load path consistent with paragraph f.(8)(ii)(A). This may be necessary for similarly stressed load paths when damage according to the threat assessment could occur in each element at the same time.

(2) When considering redundancy in a multiple load path structure, two scenarios might be possible; one where the required inspection is for a completely failed load path and one where the inspection is for less than a load path failure. In either case, the remaining life of the secondary load path after primary load path failure is used to determine the inspection interval. Consistent with this, the resulting intervals are only valid until the cumulative fatigue damage or crack growth in the intact structure is taken into account. This issue is illustrated in a crack growth context in Figure AC 29.571B-5. Crack growth in the secondary load path from an initial crack as detailed in paragraph f.(8)(ii)(B)(3)(i) will proceed along curve A-B as long as the primary load path remains intact and load redistribution is negligible. However, at the time of primary load path failure, loading on the secondary load path will increase due to load redistribution and crack growth will be accelerated (e.g., subsequent growth from point 1, 2, or 3 depending on if the failure occurs at time t_1, t_2 or t_3). Note that the residual life, L_r, in the secondary load path is inversely proportional to the time at which primary load path failure occurs. This should be considered whenever L_r is used in establishing repeat inspection intervals.

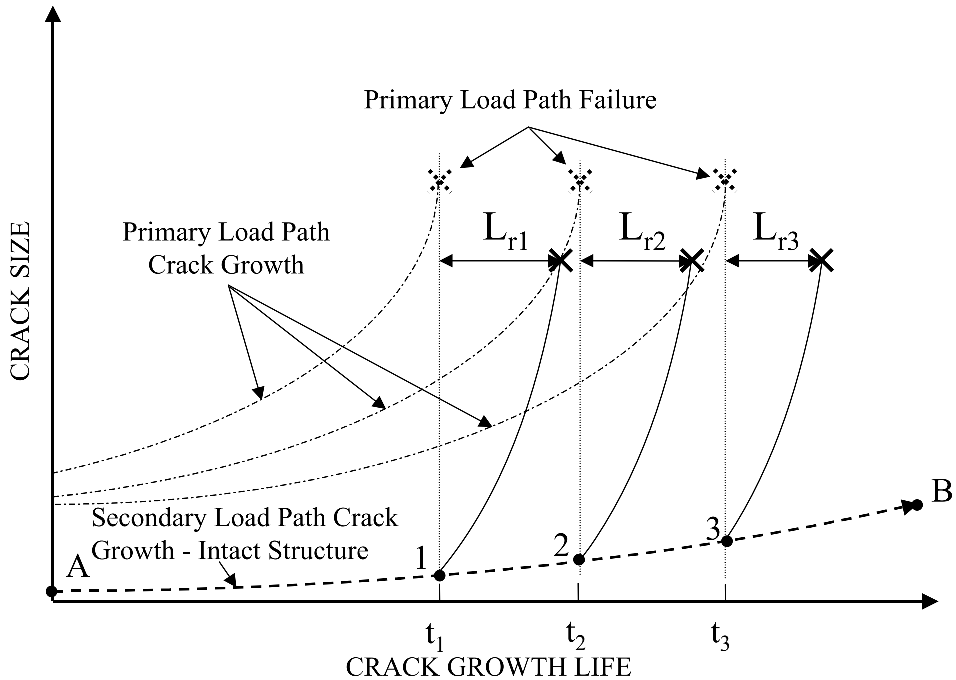

Figure AC 29.571B-5. Decreasing Residual Life in Secondary Load Path for Multiple Element Crack Growth with Inspections.

(3) Inspect for Load Path Failure. If a failed load path is easily detectable and the residual life and strength of the remaining structure is sufficient, this approach may be optimum. Analysis or tests as described in the following paragraphs can determine the inspection interval.

(i) Evaluation by analysis. Figure AC 29.571B-6 illustrates an example of multiple load path structure for which a completely failed load path is easily detectable. The inspection interval is based on the life of the secondary load path (L_r) after primary load path failure at time N_F. Consistent with this, damage accumulated in the secondary load path prior to primary load path failure must be accounted for in the analysis. In order to do this within the context of a crack growth analysis, it is necessary to assume some initial crack, of size a_i, exists in the secondary load path at time zero. This initial crack size should be representative of a normal manufacturing quality unless the threat assessment indicates that larger damage could exist. Crack growth accumulated prior to a load path failure is accounted for by calculating the amount of growth, (Δa_i), between time zero and N_F. Load redistribution that may occur prior to N_F should be considered. The residual life, (L_r), then becomes the time for a crack of size $a_i + \Delta a_i$ to grow to critical size, assuming a complete load path failure has occurred (i.e., "failed" condition loads used). It should be noted that the assumed time of load path failure would also represent an upper limit of validity for any repeat inspection period based on L_r. It is therefore recommended that N_F be assumed equal to the retirement time for the structure being inspected or the rotorcraft design life if the structure has no declared retirement time. Based on the above,

(A) Inspection Interval = L_r/N [For N refer to paragraph f.(8)(ii)(C)].

(B) Limit of validity = N_F (i.e., repetitive inspection time would not be valid for operation beyond N_F).

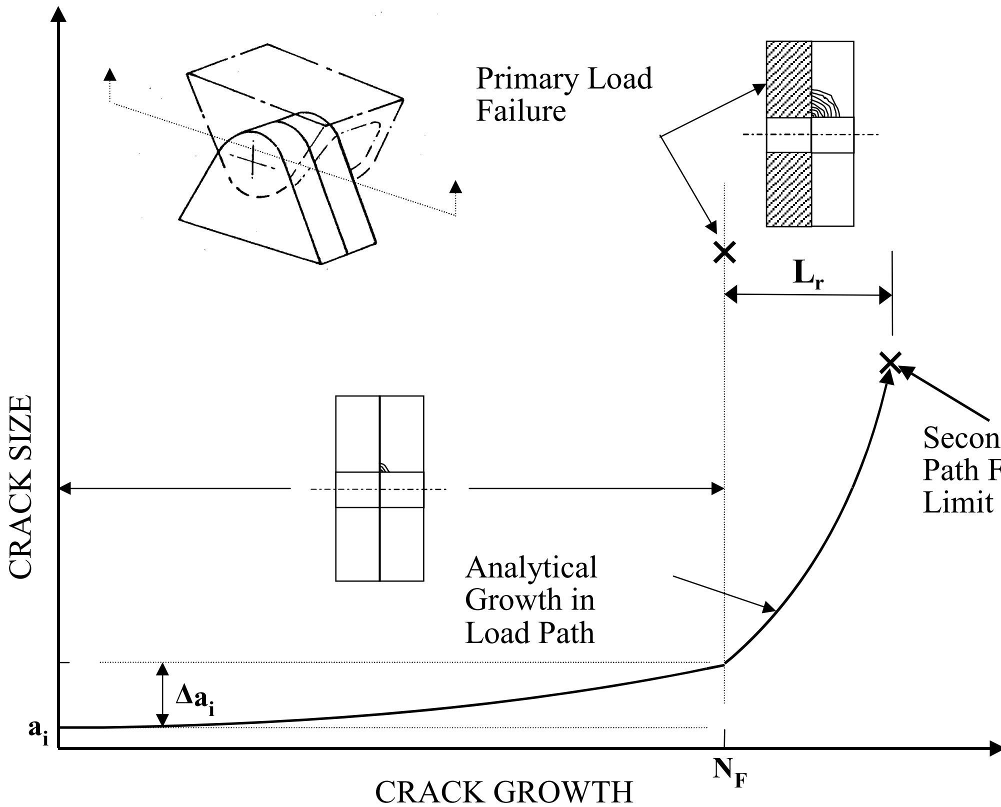

Figure AC 29.571B-6. Multiple Load Path Structure Analytical Evaluation to Support Inspection for a Failed Load Path.

(ii) Evaluation by Test. Figure AC 29.571B-7 illustrates some key points if an inspection for a complete load path failure is to be developed based on testing. The inspection interval is based on the test demonstrated residual life (L_r) subsequent to load path failure. Because the residual life decreases with the time accumulated prior to a load path failure, there will be a limit of validity to the L_r and it will be dependent on the time at which a load path failure is simulated, (N_D).

(A) The test article should consist of as-manufactured production parts. Representative "well" condition loading should be applied for some predetermined period of time, (N_D). It is recommended that the "well" condition loading be of sufficient duration so that N_D/L_{SF} is not less than the retirement time minus one inspection interval for the structure being inspected or the rotorcraft design life if the structure has no declared retirement time. At the end of this period, the load path that is to be inspected for complete failure should be disabled (e.g., saw cutting, attachment(s) removal, member removal) to simulate its failure. The test should then be restarted with a representative “failed” condition loading. (Note that the external loads may be the same as for the “well” condition if the member failure simulation results in the correct “failed” condition internal load redistribution.) The test should continue until the desired residual life has been achieved or to the time at which the secondary load path can no longer support limit loads without failure, whichever is less, (N_0).

(B) In developing the test spectrum, consideration should be given to proper use of representative loads, truncation of non-damaging loads, inclusion of ground-air-ground cycles, clipping of high magnitude loads, and load sequence.

(C) Based on the above,

(a) Demonstrated residual life = $L_r = N_0 - N_D$.

(b) Repetitive inspection time = L_r/N [For N refer to paragraph f.(8)(ii)(C)].

(c) Limit of validity = N_D/L_{SF}.

(d) $L_{SF} = 2$, Life safety factor.

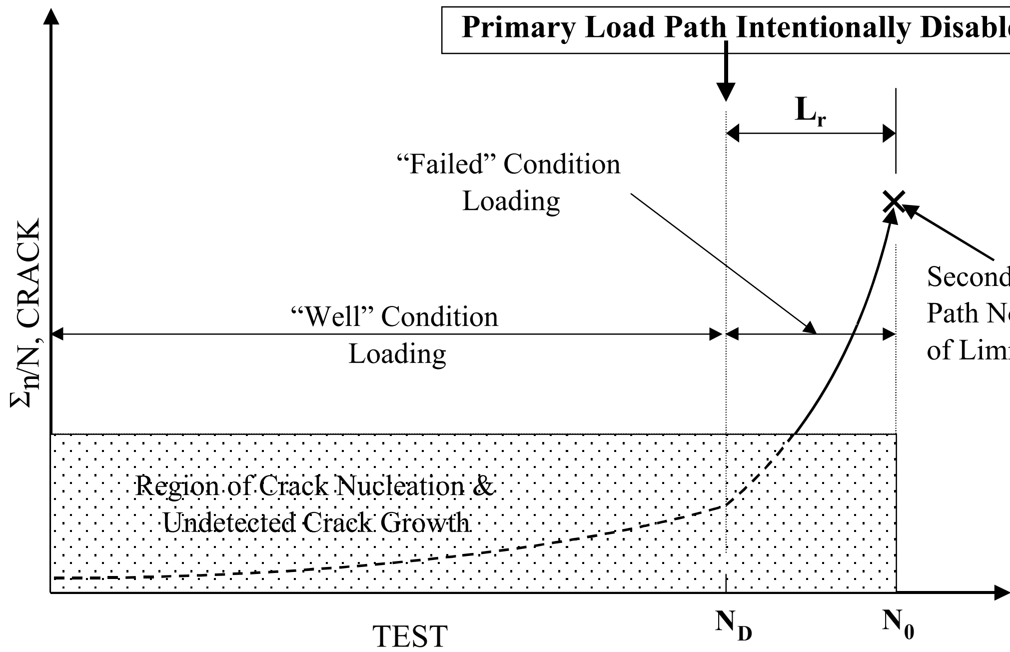

Figure AC 29.571B-7. Multiple Load Path Structure Evaluation by Test to Support Inspection for a Failed Load Path.

(4) Inspect for Less Than a Load Path Failure. Inspection for less than a load path failure may require special non-destructive Inspection (NDI) procedures but will result in longer inspection intervals. Figure AC 29.571B-8 illustrates how inspection intervals could be established on the basis of crack growth and residual strength evaluation.

(i) In this case, the inspection interval is based on the life of the secondary load path (L_r) subsequent to primary load path failure at N_F plus the time (L_P) for a detectable crack (a_{DET}) in the primary load path to grow to critical size under in-service loads. The determination of L_r is the same as discussed in paragraph f.(8)(ii)(B)(3)(i).

(ii) Based on the above,

(A) Repetitive Inspection = $(L_P + L_r)/N$ [For N refer to paragraph f.(8)(ii)(C)].

(B) Limit of validity = N_F.

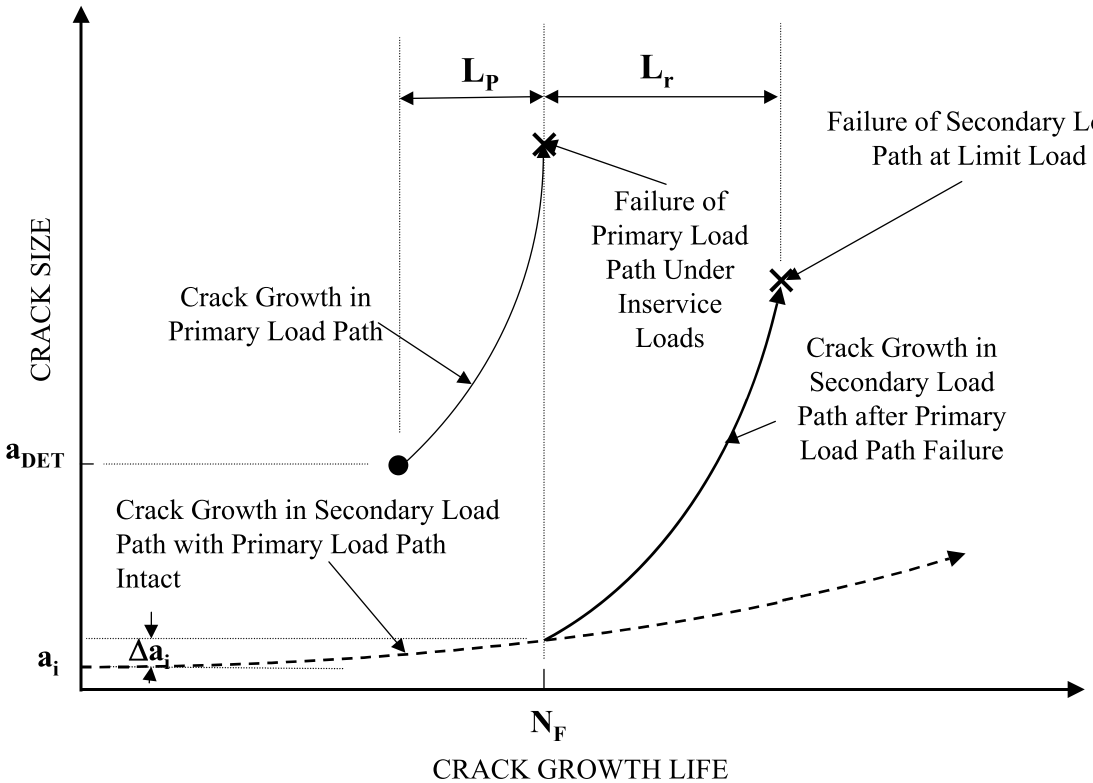

Figure AC 29.571B-8. Multiple Load Path Structure Analytical Evaluation to Support Inspection for Less than a Failed Load Path.

(C) Safety Factors.

(1) In determining the factor of N to be used for determining the inspection time, consideration should be given to the crack growth data used (e.g., top of scatter data versus average data, number of specimens used to generate data, etc.) and the capability of the inspection procedure.

(2) The minimum suggested N value should be N=2 in the case where the conservative top-of-scatter crack growth data are used in the crack growth analysis, or N=4 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 full scale specimens, N=3 of the shortest crack growth life can be used).

(iii) Safe-Life Inspection for a Failed Element.

(A) A Safe-Life Inspection substantiation for a Failed Load Path provides a safe interval of operation between repetitive inspections for the failed load path. The substantiation is accomplished by testing and analysis employing conventional Safe-Life methodology except that the configuration of the structure substantiated is with the critical load path inoperative and appropriate flaws imposed on the remainder of the structure, as determined by the threat assessment.

(B) The method described in paragraph f.(8)(i) can be employed for this case with the following differences:

(1) The principal "flaw" considered is failure or loss of the most critical load path. The load path failure can be the result of fatigue cracking, static failure, or a fractured or missing fastener, as determined by the threat assessment, paragraph f.(5).

(2) The remainder of the structure may be representative of normal manufacturing quality unless the threat assessment indicates that larger damage should exist.

(3) The mean strength for the substantiation should be based on the number of cycles from the first load path failure to the first initiation of cracking at any other point in the remaining structure. Any applied load changes or load distribution changes that occur as a consequence of the load path failure should also be included (bending due to increased deflection, for example).

(4) When the remaining structure may have some pre-existing fatigue damage at the time the first load path fails (due to both load paths being highly loaded, for example), this should be factored into the analysis.

(5) The remaining structure after first load path failure must be shown to have limit load capability, considered as the ultimate loading, except in some cases where no retirement life is provided and fatigue damage is expected (see paragraph f.(10).

(6) The inspection conducted is for the failed or missing load path.

(9) RETIREMENT TIME AND INSPECTION INTERVAL SCHEDULES.

(i) Based on the evaluations required by § 29.571, inspections, retirement times, combinations thereof, or other procedures have been established as necessary to avoid catastrophic failure. These inspections, retirement times, or approved equivalent means must be included in the Airworthiness Limitations Section (ALS) of the Instructions for Continued Airworthiness (ICA) as required by § 29.1529 and Appendix A29.4 of the regulatory requirements. These inspections, retirement times, or a combination of both are normally stated in hours time-in-service, but may be stated in other terms, such as engine starts, landings, external lifts, etc.

(ii) The design service life should be specified in the fatigue evaluation methodology that must be approved by the FAA. In any case, routine inspections for wear, fretting, corrosion, cracking, and service damage are appropriate. These routine inspections should be noted in the ICAs (maintenance manual) but are not required to be contained within the ALS of the ICAs unless they are structural inspection intervals or related structural inspection procedures approved under § 29.571.

(10) APPROVED EQUIVALENT MEANS. The requirement includes the possibility that in place of setting retirement times or inspections for damage, some other means may be used. All proposals for ‘equivalent means’ must be submitted to the FAA for approval. Potentially equivalent means to inspection include, but are not limited to:

(i) Indirect detection of damage used to establish a period of safe operation for a structure with the damage present. In this case, the detection is based on the effect of the damage, which may be recognized through:

(A) A warning in flight or during maintenance from a specific feature, sensor, or health monitor, including: oil analysis, chip detector, crack detection wire or foil, health monitoring, fluid leaks or pressure change in a sealed chamber; or by

(B) Pilot sensitivity to a change in the rotorcraft’s behavior (such as poor blade tracking, noise generation, vibration generation) provided it is well defined and does not require exceptional piloting skills to recognize these behaviors.

(ii) In all cases, an adequate level of residual strength is demonstrated for the period of operation concerned. Generally, limit load will be considered the minimum residual strength requirement. However, load levels less than the critical limit load conditions may be acceptable for consideration of obvious damage sustained in flight and for the completion of that flight only, provided it allows for continued safe flight and landing.

(iii) Two instances are considered here where it may not be necessary to provide a retirement time in the ALS of the ICAs. However, this does not preclude the investigation of fatigue behavior throughout the life of the rotorcraft or of the part if longer.

(A) When fatigue cracking occurs, or is expected to occur, for a specific PSE while in service, then the first approach allows the PSE to operate until the damage is found. Therefore, the inspection must find the damage prior to loss of ultimate load capability. This approach may not be appropriate for a single load path structure. For such a process to be safe, the behavior of the part and associated parts that influence its fatigue behavior must be substantiated for as long as they remain in service. All potential failure modes throughout the life of the rotorcraft must be identified and shown to be consistent, repeatable and addressed by the inspection program. In order to meet the intent of the new fatigue tolerance requirements, a high probability of ultimate load capability is required throughout the lifetime of the component. Therefore, for cracks or other damage that are allowed or highly likely to exist, ultimate load capability should be substantiated for that damage and any growth that may occur during the subsequent inspection period.

(B) It may be acceptable that a PSE does not have a specific retirement time when the fatigue tolerance of the part, including any damage not controlled by an acceptable inspection program, has been demonstrated to be in excess of the rotorcraft design life to such an extent that no safety benefit arises from imposing that requirement.

(11) SUPPLEMENTAL PROCEDURES.

(i) The requirement states that if inspections, for any of the damage types identified during the threat assessment, cannot be established within the limitations of geometry, inspectability or good design practice, then supplemental procedures must be established that will minimize the risk of each of these types of damage being present or leading to catastrophic failure. When assessing good design practice, measures such as improved protection against impact, scratches, and corrosion should already have been considered. If the part cannot be redesigned to reduce the acquisition and influence of damage, then supplemental procedures should be introduced.

(ii) Supplemental procedures that should be considered include, but are not limited to:

(A) Specifying shorter than usual calendar inspection intervals to reduce the probability of occurrence and the extent of the damage.

(B) Improving control of maintenance processes associated with the component and damage type, such as by providing specifically designed tooling and requiring additional quality checks after each operation is performed.

(C) Introducing an overhaul program.

(D) Restricting the allowable repair limits for the part.

(E) Modifying the PSE design based on service experience if this shows the original design assumptions to be overly conservative with respect to demonstrating impracticality at certification.

(F) Specifying a conservative inspection interval, if the calculated interval cannot be established and there are no other alternatives.