Federal Aviation Administration
Aviation Rulemaking Advisory Committee

Transport Airplane and Engine Issue Area
General Structures Harmonization Working Group

Task 5 – Damage Tolerance and Fatigue

Task Assignment

[Federal Register: January 20, 1995 (Volume 60, Number 13)]
[Notices]
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DEPARTMENT OF TRANSPORTATION

Aviation Rulemaking Advisory Committee; Transport Airplane and Engine Issues--New Tasks

AGENCY: Federal Aviation Administration (**FAA**), DOT.

ACTION: Notice of new task assignments for the Aviation Rulemaking Advisory Committee.

SUMMARY: Notice is given of new tasks assigned to the Aviation Rulemaking Advisory Committee (ARAC). This notice informs the public of the activities of ARAC.

FOR FURTHER INFORMATION CONTACT:
Stewart R. Miller, Manager, Transport Standards Staff, ANM-110, Transport Airplane Directorate, Federal Aviation Administration, 1601 Lind Avenue SW, Renton, Washington, 98055-4056; telephone (206) 227-2190; (206) 227-1320.

SUPPLEMENTARY INFORMATION: The Federal Aviation Administration (**FAA**) has established an Aviation Rulemaking Advisory Committee (56 FR 2190, January 22, 1991; and 58 FR 9230, February 19, 1993). One area the ARAC deals with is transport airplane and engine issues. These issues involve the airworthiness standards for transport category airplanes and engines in parts 25, 33, and 35 of the Federal Aviation Regulations (FAR) and parallel provisions in parts 121 and 135 of the FAR.

The **FAA** announced at the Joint Aviation Authorities (JAA)-Federal Aviation Administration (**FAA**) Harmonization Conference in Toronto, Canada, June 2-5, 1992, that it would consolidate within the ARAC structure an ongoing objective to ``harmonize'' the Joint Aviation Requirements (JAR) and the Federal Aviation Regulations (FAR).

Tasks

The following three new harmonization tasks are being assigned to ARAC:

Task 1--Material Strength Properties and Design Values

Review Title 14 Code of Federal Regulations, Section 25.613, corresponding Paragraph 25.613 of the European Joint Aviation Requirements (JAR), and supporting policy and guidance material, and recommend to the **FAA** appropriate revisions for harmonization, including advisory material.

Task 2--Proof of Structure

Review Title 14 Code of Federal Regulations, Section 25.307, corresponding Paragraph 25.307 of the JAR, and supporting policy and guidance material, and recommend to the **FAA** appropriate revisions relative to the issue concerning limit load tests, ultimate load tests, and structural testing for harmonization, including advisory material.

Task 3--Damage Tolerance and Fatigue

Review Title 14 Code of Federal Regulations, Section 25.571, [[Page 4223]] corresponding Paragraph 25.571 of the JAR, and supporting policy and guidance material and recommend to the **FAA** appropriate revisions for harmonization, including advisory material.

ARAC recommendations to the **FAA** should be accompanied by appropriate documents. Recommendations for rulemaking should be accompanied by a complete draft of the notice of proposed rulemaking, including the Benefit/Cost Analysis and other required analyses. Recommendations for the issuance of guidance material should be accompanied by a complete draft advisory circular.

ARAC normally forms working groups to analyze and recommend to it solutions to issues contained in assigned tasks. If ARAC accepts the working group's recommendations, it forwards them to the **FAA**. At this point, ARAC has not identified working groups for these tasks.

ARAC working groups are comprised of technical experts on the subject matter. A working group member need not necessarily be a representative of one of the member organizations of ARAC. An individual who has expertise in the subject matter and wishes to become a member of the working group should write the person listed under the caption FOR FURTHER INFORMATION CONTACT expressing that desire, describing his or her interest in the task, and the expertise he or she would bring to the working group. The request will be reviewed by the ARAC assistant chair and working group leader, and the individual will be advised whether or not the request can be accommodated.

Working Group Reports

Each working group formed to consider ARAC tasks is expected to comply with the procedures adopted by ARAC and given to the working group chair. As part of the procedures, the working group is expected to:

A. Recommend time line(s) for completion of the tasks, including rationale, for consideration at the meeting of the ARAC to consider transport airplane and engine issues held following publication of this notice.

B. Give a detailed conceptual presentation on the tasks to the ARAC before proceeding with the work stated under item C below.

C. Give a status report on the tasks at each meeting of ARAC held to consider transport airplane and engine issues.

The Secretary of Transportation has determined that the formation and use of the ARAC are necessary in the public interest in connection with the performance of duties imposed on the **FAA** by law. Meetings of the ARAC will be open to the public except as authorized by section 10(d) of the Federal Advisory Committee Act. Meetings of the working group will not be open to the public, except to the extent that individuals with an interest and expertise are selected to participate.

No public announcement of working group meetings will be made.

Issued in Washington, DC, on January 13, 1995.
Chris A. Christie,
Executive Director, Aviation Rulemaking Advisory Committee.
[FR Doc. 95-1539 Filed 1-19-95; 8:45 am]
BILLING CODE 4910-13-M

Recommendation Letter

Action: ARM

Gerald R. Mack
Director
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cc Barry
Peg
AFS
AIR

BOEING

August 5, 1996
B-T000-ARAC-96-007

Mr. Barry L. Valentine
Acting Associate Administrator for
Regulation and Certification
Department of Transportation
Federal Aviation Administration
800 Independence Avenue, S.W.
Washington, DC 20591

Dear Mr. Valentine:

On behalf of the Aviation Rulemaking Advisory Committee (ARAC), I am pleased to submit the proposed Advisory Circular (AC) 25-571-1X, *Damage-Tolerance and Fatigue Evaluation of Structure*. This document was developed by the General Structures Working Group chaired by Herb Lancaster.

The language on AC 25-571-1X proposed by the working group was accepted unanimously to be forwarded to the FAA with a recommendation for adoption.

The members of ARAC appreciate the opportunity to participate in the FAA rulemaking process.

Sincerely,

GRMack

Gerald R. Mack
Chairman,
Transport Airplane & Engine Issues Group
Aviation Rulemaking Advisory Committee
Tele: (206) 234-9570, FAX: (206) 237-4838

Enclosure

Recommendation

U.S. Department of Transportation
Federal Aviation Administration

DRAFT
5-30-96

Advisory Circular

JUN - 4 1996

Subject: DAMAGE-TOLERANCE AND FATIGUE EVALUATION OF STRUCTURE

Date:
Initiated by: ANM-110

AC No: 25.571-1X
Change:

1. PURPOSE. This advisory circular (AC) sets forth an acceptable means of compliance with the provisions of Part 25 of the Federal Aviation Regulations (FAR) dealing with the damage-tolerance and fatigue evaluation requirements of transport category aircraft structure. It also provides rational guidelines for the evaluation of scatter factors for the determination of life for parts categorized as Safe-Life.

2 CANCELLATION. Advisory Circular 25.571-1A, dated March 5, 1986, is cancelled.

3. DEFINITIONS OF TERMS USED IN THIS AC.

a. Damage tolerance means that the structure has been evaluated to ensure that should serious fatigue, corrosion, or accidental damage occur within the operational life of the airplane, the remaining structure can withstand reasonable loads without failure or excessive structural deformation until the damage is detected.

b. Fail-safe means that the structure has been evaluated to assure that catastrophic failure is not probable after fatigue failure or obvious partial failure of a single, principal structural element.

c. Safe-life means that the structure has been evaluated to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks.

d. Principal structural elements are those which contribute significantly to carrying flight, ground, and pressurization loads, and whose failure could result in catastrophic failure of the airplane.

e. Critical structural elements are those elements whose failure would result in catastrophic failure of the airplane.

f. Primary structure is that structure which carries flight, ground, or pressure loads.

g. Secondary structure is that structure which carries only air or inertial loads generated on or within the secondary structure.

h. Single load path is where the applied loads are eventually distributed through a single member within an assembly, the failure of which would result in the loss of the structural integrity of the component involved.

i. Multiple load path is identified with redundant structures in which (with the failure of individual elements) the applied loads would be safely distributed to other load-carrying members.

j. Reliability refers to detail designs or methodologies which service history has demonstrated to be reliable.

k. Probability refers to a probability of occurrence of an event consistent with past successful experience..

l. Scatter factor. A life reduction factor used in the interpretation of fatigue analysis and test results.

4. BACKGROUND.

a. Since the early 1970's, there have been significant state-of-the-art and industry-practice developments in the area of structural fatigue and fail-safe strength evaluation of transport category airplanes. Recognizing that these developments could warrant some revision of the existing fatigue requirements in §§ 25.571 and 25.573 of Part 25 of the FAR, the FAA, on November 18, 1976, gave notice of its Transport Category Airplane Fatigue Regulatory Review Program and invited interested persons to submit proposals to amend those requirements (41 FR 50956). The proposals and related discussions formed the basis for the revision of the structural fatigue evaluation standards of §§ 25.571 and 25.573 and the development of guidance material. To that end, § 25.571 was revised, § 25.573 was deleted (the scope of § 25.571 was expanded to cover the substance of the deleted section), and guidance material (AC 25.571-1) was provided which contained compliance provisions related to the proposed change.

b. Since issuance of AC 25.571-1 on 9/28/78, additional guidance material, including discrete source damage, was developed and incorporated in revision 1A on 3/5/86. The AC is further revised to add guidance on the elements to be considered in developing scatter factors for certification.

5. INTRODUCTION.

a. The contents of this advisory circular are considered by the FAA in determining compliance with the damage-tolerance and fatigue requirements of § 25.571.

(1) Although a uniform approach to the evaluation required by § 25.571 is desirable, it is recognized that in such a complex field new design features and methods of

fabrication, new approaches to the evaluation, and new configurations could necessitate variations and deviations from the procedures described in this advisory circular. Close adherence to the procedures in this advisory circular is encouraged.

(2) Damage tolerance design is required, unless it entails such complications that an effective damage-tolerant structure cannot be achieved within the limitations of geometry, inspectability, or good design practice. Under these circumstances, a design that complies with the fatigue evaluation (safe-life) requirements is used. A typical example of structure that might not be conducive to damage-tolerance design is the landing gear and its attachments.

(3) Experience with the application of methods of fatigue evaluation indicates that a test background should exist in order to achieve the design objective. Even under the damage tolerance method discussed in paragraph 6 of this AC, it is the general practice within industry to conduct damage tolerance tests for design information and guidance purposes. Damage location, growth, and detection data should also be considered in establishing a recommended inspection program.

b. Typical loading spectrum expected in service. The loading spectrum should be based on measured statistical data of the type derived from government and industry load history studies and, where insufficient data are available, on a conservative estimate of the anticipated use of the airplane. The principal loads that should be considered in establishing a loading spectrum are flight loads (gust and maneuver), ground loads (taxiing, landing impact, turning, engine runup, braking, thrust reversing, and towing), and pressurization loads.. The development of the loading spectrum includes the definition of the expected flight plan which involves climb, cruise, descent, flight times, operational speeds and altitudes, and the approximate time to be spent in each of the operating regimes. Operations for crew training and other pertinent factors, such as the dynamic stress characteristics of any flexible structure excited by turbulence or buffeting, should also be considered. For pressurized cabins, the loading spectrum should include the repeated application of the normal operating differential pressure, and the superimposed effects of flight loads and external aerodynamic pressures.

c. Components to be evaluated. In assessing the possibility of serious fatigue failures, the design should be examined to determine probable points of failure in service. In this examination, consideration should be given, as necessary, to the results of stress analyses, static tests, fatigue tests, strain gage surveys, tests of similar structural configurations, and service experience. Service experience has shown that special attention should be focused on the design details of important discontinuities, main attach fittings, tension joints, splices, and cutouts such as windows, doors, and other openings. Locations prone to accidental damage (such as that due to impact with ground servicing equipment near airplane doors) or to corrosion should also be considered.

d. Analyses and tests. Unless it is determined from the foregoing examination that the normal operating stresses in specific regions of the structure are of such a low order that serious damage growth is extremely improbable, repeated load analyses or tests should be conducted on structures representative of components or subcomponents of the wing, control surfaces, empennage, fuselage, landing gear, and their related primary attachments. Test specimens should include structure representative of attachment fittings, major joints, changes in section, cutouts, and discontinuities. Any method used in the analyses should be supported, as necessary, by test or service experience. Typical (average) values of material properties and other parameters may be used in residual strength, crack growth, and damage detection analyses for damage tolerance evaluations per paragraph 6 and discrete source damage per paragraph 8.

6. DAMAGE-TOLERANCE EVALUATION.

a. General. The damage tolerance evaluation of structure is intended to ensure that should serious fatigue, corrosion, or accidental damage occur within the operational life of the airplane, the remaining structure can withstand reasonable loads without failure or excessive structural deformation until the damage is detected. Included are the considerations historically associated with fail-safe design. The evaluation should encompass establishing the components which are to be designed as damage-tolerant, defining the loading conditions and extent of damage, conducting structural tests or analyses, or both, to substantiate that the design objective has been achieved, and establishing data for inspection programs to ensure detection of damage. Although this evaluation applies to either single or multiple load path structure, the use of multiple load path structure should be given high priority in achieving damage-tolerant design. Design features which should be considered in attaining a damage-tolerant structure include the following:

(1) Multiple load path construction and the use of crack stoppers to control the rate of crack growth, and to provide adequate residual static strength;

(2) Materials and stress levels that, after initiation of cracks, provide a controlled slow rate of crack propagation combined with high residual strength;

(3) Arrangement of design details to ensure a sufficiently high probability that a failure in any critical structural element will be detected before the strength has been reduced below the level necessary to withstand the loading conditions specified in § 25.571(b), so as to allow replacement or repair of the failed elements; and

(4) Provisions to limit the probability of concurrent multiple damage, particularly after long service, which could conceivably contribute to a common facture path. Examples of such multiple damage are:

(i) A number of small cracks which might coalesce to form a single long crack;

(ii) Failures, or partial failures, in adjacent areas due to the redistribution of loading following a failure of a single element; and

(iii) Simultaneous failure, or partial failure, of multiple load path discrete elements, working at similar stress levels.

b. Normally, the damage tolerance assessment consists of a deterministic evaluation of the above design features. This paragraph provides guidelines for this approach. In certain specific instances, however, damage-tolerant design might be more realistically assessed by a probabilistic evaluation employing methods such as risk analysis. They are routinely employed in fail-safe evaluations of airplane systems and have occasionally been used where structure and systems are interrelated. These methods can be of particular value for structure consisting of discrete isolated elements where damage tolerance depends on the ability of the structure to sustain redistributed loads after failures of discrete elements resulting from fatigue, corrosion, or accidental damage. Where considered appropriate on multiple load path structure, probabilistic analysis may be used if it can be shown that loss of the airplane is extremely improbable, and the statistical data employed in the analysis is based on tests or operational experience, or both, of similar structure.

c. Identification of principal structural elements. Principal structural elements are those which contribute significantly to carrying flight, ground, and pressurization loads, and whose failure could result in catastrophic failure of the airplane. Typical examples of such elements are as follows:

(1) Wing and empennage.

(i) Control surfaces, slats, flaps, and their mechanical systems and attachments (hinges, tracks, and fittings);

(ii) Integrally stiffened plates;

(iii) Primary fittings;

(iv) Principal splices;

(v) Skin or reinforcement around cutouts or discontinuities;

(vi) Skin-stringer combinations;

(vii) Spar caps; and

(viii) Spar webs.

(2) Fuselage.

(i) Circumferential frames and adjacent skin;

(ii) Door frames;

(iii) Pilot window posts;

(iv) Pressure bulkheads;

(v) Skin and any single frame or stiffener element around a cutout;

(vi) Skin or skin splices, or both, under circumferential loads;

(vii) Skin or skin splices, or both, under fore and aft loads;

(viii) Skin around a cutout;

(ix) Skin and stiffener combinations under fore and aft loads;

(x) Door skins, frames, and latches; and

(xi) Window frames.

(3) Landing gear and their attachments.

(4) Engine mounts.

d. Extent of damage. Each particular design should be assessed to establish appropriate damage criteria in relation to inspectability and damage extension characteristics. In any damage determination, including those involving multiple cracks, it is possible to establish the extent of damage in terms of detectability with the inspection techniques to be used, the associated initially detectable crack size, the residual strength capabilities of the structure, and the likely damage-extension rate, considering the expected stress redistribution under the repeated loads expected in service and with the expected inspection frequency. Thus, an obvious partial failure could be considered to be the extent of the damage for residual strength assessment, provided a positive determination is made that the fatigue cracks will be detectable by the available inspection techniques at a sufficiently early stage of the crack development. In a pressurized fuselage, an obvious partial failure might be detectable through the inability of the cabin to maintain operating pressure or controlled decompression after occurrence of the damage. The following are typical examples of partial failures which should be considered in the evaluation:

(1) Detectable skin cracks emanating from the edge of structural openings or cutouts;

(2) A detectable circumferential or longitudinal skin crack in the basic fuselage structure;

(3) Complete severance of interior frame elements or stiffeners in addition to a detectable crack in the adjacent skin;

(4) A detectable failure of one element where dual construction is utilized in components such as spar caps, window posts, window or door frames, and skin structure;

(5) The presence of a detectable fatigue failure in at least the tension portion of the spar web or similar element; and

(6) The detectable failure of a primary attachment, including a control surface hinge and fitting.

e. Inaccessible areas. Every reasonable effort should be made to ensure inspectability of all structural parts, and to qualify them under the damage tolerance provisions (reference § 25.611).

f. Testing of principal structural elements. The nature and extent of residual strength tests on complete structures or on portions of the primary structure will depend upon applicable previous design, construction, tests, and service experience, in connection with similar structures. Simulated cracks should be as representative as possible of actual fatigue damage. Where it is not practical to produce actual fatigue cracks, damage can be simulated by cuts made with a fine saw, sharp blade, guillotine, or other suitable means. If sawcuts in primary structure are used to simulate sharp fatigue cracks, sufficient evidence should be available from element tests to indicate equivalent residual strength. In those cases where bolt failure, or its equivalent, is to be simulated as part of a possible damage configuration in joints or fittings, bolts can be removed to provide that part of the simulation.

g. Identification of locations to be evaluated. The locations of damage to structure for damage tolerance evaluation should be identified as follows:

(1) Determination of general damage locations. The location and modes of damage can be determined by analysis or by fatigue tests on complete structures or subcomponents. However, tests might be necessary when the basis for analytical prediction is not reliable, such as for complex components. If less than the complete structure is tested, care should be taken to ensure that the internal loads and boundary conditions are valid.

(i) If a determination is made by analysis, factors such as the following should be taken into account:

(A) Strain data on undamaged structure to establish points of high stress concentration, as well as the magnitude of the concentration;

(B) Locations where permanent deformation occurred in static tests;

(C) Locations of potential fatigue damage identified by fatigue analysis; and

(D) Design details which service experience of similarly designed components indicates are prone to fatigue or other damage.

(ii) In addition, the areas of probable damage from sources such as severe corrosive environment damage should be determined from a review of the design and past service experience.

(2) Selection of critical damage areas. The process of actually locating where damage should be simulated in principal structural elements identified in paragraph 6c of this AC should take into account factors such as the following:

(i) Review analysis to locate areas of maximum stress and low margin of safety;

(ii) Select locations in an element where the stresses in adjacent elements would be the maximum with the damage present;

(iii) Select partial fracture locations in an element where high stress concentrations are present in the residual structure; and

(iv) Select locations where detection would be difficult.

h. Damage tolerance analysis and tests.

(1) It should be determined by analysis, supported by test evidence, that:

(i) The structure, with the extent of damage established for residual strength evaluation, can withstand the specified design limit loads (considered as ultimate loads); and

(ii) The damage growth rate under the repeated loads expected in service (between the time the damage becomes initially detectable and the time the extent of damage reaches the value for residual strength evaluation) provides a practical basis for development of the inspection program and procedures described in paragraph 6i of this AC.

(2) The repeated loads should be as defined in the loading, temperature, and humidity spectra. The loading conditions should take into account the effects of structural flexibility and rate of loading where they are significant.

(3) The damage tolerance characteristics can be shown analytically by reliable or conservative methods such as the following:

(i) By demonstrating quantitative relationships with structure already verified as damage tolerant;

(ii) By demonstrating that the damage would be detected before it reaches the value for residual strength evaluation; or

(iii) By demonstrating that the repeated loads and limit load stresses do not exceed those of previously verified designs of similar configuration, materials, and inspectability.

(4) The maximum extent of immediately obvious damage from discrete sources should be determined and the remaining structure shown to have static strength for the maximum load (considered as ultimate load) expected during the completion of the flight. Normally, this would be an analytical assessment. In the case of uncontained engine failures, the fragments and paths to be considered should be consistent with those used in showing compliance with § 25.903(d)(1) of the FAR, and with typical damage experienced in service.

i. Inspection.

(1) Detection of damage before it becomes critical is the ultimate control in ensuring the damage tolerance characteristics of the structure. Therefore, the applicant should provide sufficient guidance information to assist operators in establishing the frequency, extent, and methods of inspection of the critical structure. This kind of information must, under § 25.571(a)(3) of the FAR, be included in the Instructions for Continued Airworthiness required by § 25.1529 of the FAR.

(2) Due to the inherent, complex interactions of the many parameters affecting damage tolerance, such as operating practices, environmental effects, load sequence on crack growth, and variations in inspection methods, related operational experience should be taken into account in establishing inspection procedures.

(3) A comparative analysis can be used to guide the changes from successful past practice when necessary. Therefore, maintenance and inspection requirements should recognize the dependence on experience and should be specified in a document that provides for revision as a result of operational experience, such as the one containing the operator's FAA-approved structural inspection program developed through the Maintenance Review Board (MRB) procedures for FAR Part 121 operators.

7. FATIGUE EVALUATION.

a. General. The evaluation of structure under the following fatigue (safe-life) strength evaluation methods is intended to ensure that catastrophic fatigue failure, as a result of the repeated loads of variable magnitude expected in service, will be avoided throughout the structure's operational life. Under these methods, the fatigue life of the structure should be determined. The evaluation should include the following:

(1) Estimating or measuring the expected loading spectra for the structure;

(2) Conducting a structural analysis including consideration of the stress concentration effects;

(3) Performing fatigue testing of structure which cannot be related to a test background to establish response to the typical loading spectrum expected in service;

(4) Determining reliable replacement times by interpreting the loading history, variable load analyses, fatigue test data, service experience, and fatigue analyses;

(5) Evaluating the possibility of fatigue initiation from sources such as corrosion, stress corrosion, disbonding, accidental damage and manufacturing defects based on a review of the design, quality control and past service experience; and

(6) Providing necessary maintenance programs and replacement times to the operators. The maintenance program should be included in Instructions for Continued Airworthiness in accordance with § 25.1529.

b. Scatter Factor for Safe-life Determination. In the interpretation of fatigue analyses and test data, the effect of variability should, under § 25.571(c), be accounted for by an appropriate scatter factor. In this process it is appropriate that the applicant justify the scatter factor chosen for any safe-life part. The following guidance is provided (see Figure 1):

(1) The base scatter factors applicable to test results are: $BSF_1 = 3.0$, and BSF_2 = (see paragraph 7b(5) of this AC). If the applicant can meet the requirements of paragraph 7b(3) of this AC, he may use BSF_1 or, at his option, BSF_2.

(2) The base scatter factor, BSF_1, is associated with test results of one representative test specimen.

(3) Justification for use of BSF_1. BSF_1 may only be used if the following criteria are met:

(i) Understanding of load paths and failure modes. Service and test experience of similar in-service components that were designed using similar design criteria and methods should demonstrate that the load paths and potential failure modes of the components are well understood.

(ii) Control of design, material, and manufacturing process quality. The applicant should demonstrate that his quality system (e.g., design, process control, and material standards) ensures the scatter in fatigue properties is controlled, and that the design of the fatigue critical areas of the part account for the material scatter.

(iii) Representativeness of the test specimen.

(A) The test article should be full scale (component or sub-component) and represent that portion of the production aircraft requiring test. All differences between the test article and production article should be accounted for either by analysis supported by test evidence or by testing itself.

(B) Construction details, such as bracket attachments, clips, etc., should be accounted for, even though the items themselves may be non-load bearing.

(C) Points of load application and reaction should accurately reflect those of the aircraft, ensure correct behavior of the test article, and guard against uncharacteristic failures.

(D) Systems used to protect the structure against environmental degradation can have a negative effect on fatigue life and therefore should be included as part of the test article.

(4) Adjustments to base scatter factor BSF_1. Having satisfied the criteria of paragraph 7b(3), justifying the use of BSF_1, the base value of 3.0 should be adjusted to account for the following considerations, as necessary, where not wholly taken into account by design analysis. As a result of the adjustments, the final scatter factor may be less than, equal to, or greater than 3.0.

(i) Material fatigue scatter. Material properties should be investigated up to a 99% probability of survival and a 95% level of confidence.

(ii) Spectrum severity. Test load spectrum should be derived based on a spectrum sensitive analysis accounting for variations in both utilization (i.e. aircraft weight, cg etc.) and occurrences/size of loads. The test loads spectrum applied to the structure should be demonstrated to be conservative when compared to the usage expected in service.

(iii) Number of representative test specimens. Well established statistical methods should be used that associate the number of items tested with the distribution chosen, to obtain an adjustment to the base scatter factor.

(5) If the applicant cannot satisfy the intent of all of paragraph 7b(3) of this AC, BSF_2 should be used.

(i) The applicant should propose scatter factor BSF_2 based on careful consideration of the following issues: the required level of safety, the number of representative test specimens, how representative the test is, expected fatigue scatter, type of repeated load test, the accuracy of the test loads spectrum, spectrum severity, and the expected service environmental conditions.

(ii) In no case should the value of BSF_2 be less than 3.0.

(6) Resolution of test loadings to actual loadings. The applicant may use a number of different approaches to reduce both the number of load cycles and number of test set-ups required. These include, but are not limited to, spectrum blocking (e.g., a change in the spectrum load sequence to reduce the total number of test setups); high load clipping (e.g., the reduction of the highest spectrum loads to a level such that the beneficial effects of compression yield are reduced or eliminated); and low load truncation (e.g., the removal of non-damaging load cycles to simplify the spectrum). Due to the modifications to the flight-by-flight loading sequence caused by these changes, the applicant should propose either analytical or empirical approaches to quantify an adjustment to the number of test cycles which represents the difference between the test spectrum and assumed flight-by-flight spectrum. In addition, an adjustment to the number of test cycles may be justified by raising or lowering the test load levels, as long as appropriate data supports the applicant's position. Other effects to be considered are different failure locations, different response to fretting conditions, temperature effects, etc. The analytical approach should use well established methods or be supported by test evidence.

c. Replacement times. Replacement times should be established for parts with established safe-lives and should, under § 25.571(a)(3), be included in the information prepared under § 25.1529. These replacement times can be extended if additional data indicates an extension is warranted. Important factors that should be considered for such extensions include, but are not limited to, the following:

(1) Comparison of original evaluation with service experience.

(2) Recorded load and stress data. Recorded load and stress data entails instrumenting airplanes in service to obtain a representative sampling of actual loads and stresses experienced. The data to be measured include airspeed, altitude, and load factor versus time data; or airspeed, altitude, and strain ranges versus time data; or similar data. The data, obtained by instrumenting airplanes in service, provide a basis for correlating the estimated loading spectrum with the actual service experience.

(3) Additional analyses and tests. If test data and analyses based on repeated load tests of additional specimens are obtained, a re-evaluation of the established safe-life can be made.

(4) Tests of parts removed from service. Repeated load tests of replaced parts can be utilized to reevaluate the established safe-life. The tests should closely simulate service loading conditions. Repeated load testing of parts removed from service is especially useful where recorded load data obtained in service are available, since the actual loading experienced by the part prior to replacement is known.

(5) Repair or rework of the structure. In some cases, repair or rework of the structure can gain further life.

d. Type design developments and changes. For design developments or design changes involving structural configurations similar to those of a design already shown to comply with the applicable provisions of § 25.571(c), it might be possible to evaluate the variations in critical portions of the structure on a comparative basis. Typical examples would be redesign of the wing structure for increased loads, and the introduction in pressurized cabins of cutouts having different locations or different shapes, or both. This evaluation should involve analysis of the predicted stresses of the redesigned primary structure and correlation of the analysis with the analytical and test results used in showing compliance of the original design with § 25.571(c).

e. Environmental effects such as temperature and humidity should be considered in the damage tolerance and fatigue analysis and should be demonstrated through suitable testing.

8. DISCRETE SOURCE DAMAGE.

a. General. The purpose of this section is to establish FAA guidelines for consistent selection of load conditions for residual strength substantiation in showing compliance with § 25.571(e), Damage tolerance (discrete source) evaluation. The intent of these guidelines is to define load conditions that will not be exceeded with a satisfactory level of confidence on the flight during which the specified incident of § 25.57l(e) occurs. In defining these load conditions, consideration has been given to the expected damage to the airplane, the anticipated response of the pilot at the time of the incident, and the actions of the pilot to avoid severe load environments for the remainder of the flight consistent with his knowledge that the airplane may be in a damaged state. With these considerations in mind, the following ultimate loading conditions should be used to establish residual strength of the damaged structure.

b. The maximum extent of immediately obvious damage from discrete sources (§ 25.571(e)) should be determined and the remaining structure shown, with an acceptable level of confidence, to have static strength for the maximum load (considered as ultimate load) expected during completion of the flight.

c. The ultimate loading conditions should not be less than those developed from the following conditions:

(1) At the time of the incident:

(i) The maximum normal operating differential pressure, multiplied by a 1.1 factor, plus the expected external aerodynamic pressures during 1g level flight, combined with 1g flight loads.

(ii) The airplane, assumed to be in 1g level flight, should be shown to be able to survive any maneuver or any other flight path deviation caused by the specified incident of § 25.571(e), taking into account any likely damage to the flight controls and pilot normal corrective action.

(2) Following the incident:

(i) Seventy percent (70%) limit flight maneuver loads and, separately, 40 percent of the limit gust velocity (vertical and lateral) at the specified speeds, each combined with the maximum appropriate cabin differential pressure (including the expected external aerodynamic pressure).

(ii) The airplane must be shown by analysis to be free from flutter up to V_D/M_D with any change in structural stiffness resulting from the incident.

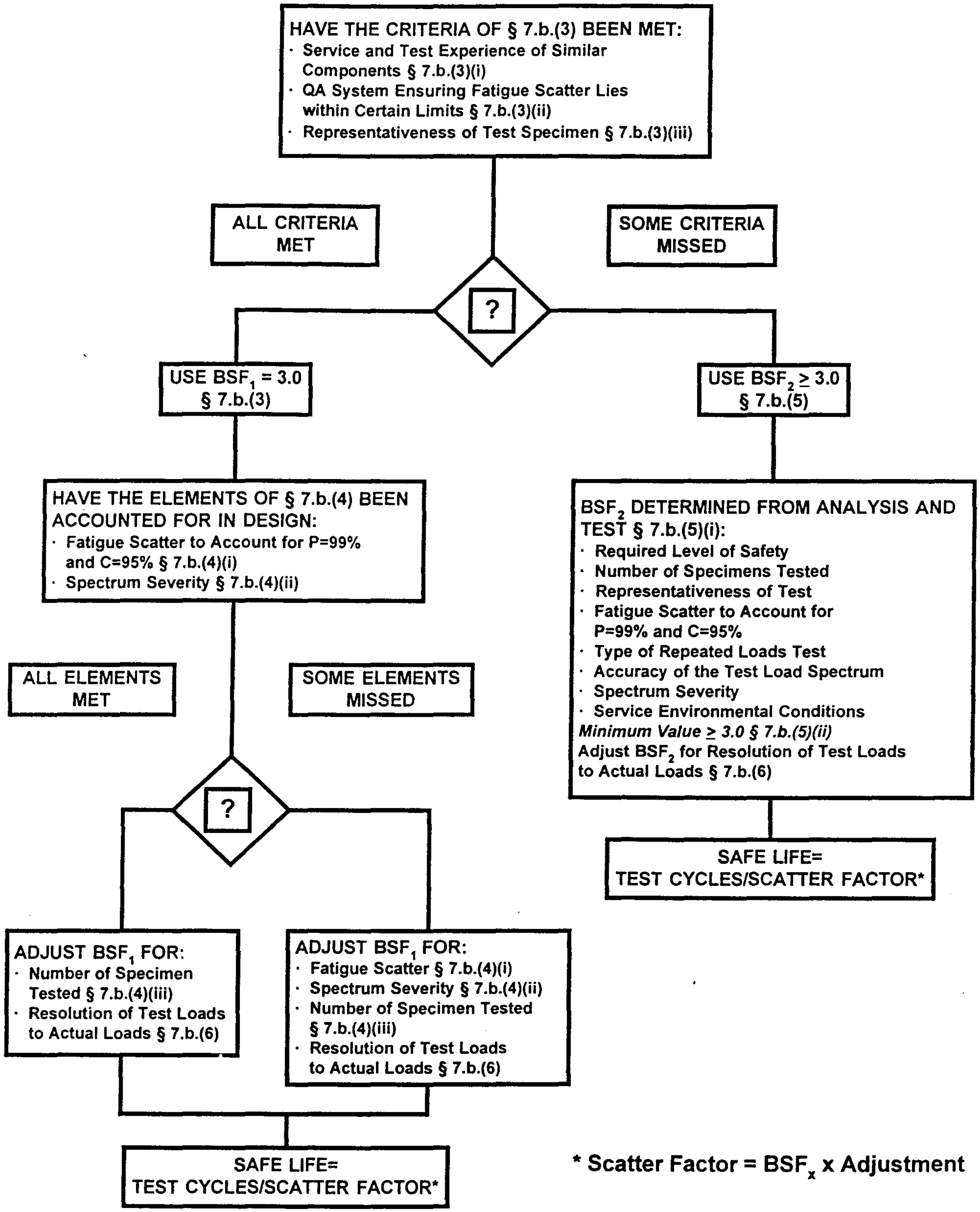

Figure 1. Safe-Life Determination

Recommendation Letter

Pratt & Whitney
400 Main Street
East Hartford, CT 06108

Pratt & Whitney
A United Technologies Company

October 22, 2003

Federal Aviation Administration
800 Independence Avenue, SW
Washington, D.C. 20591

Attention: Mr. Nicholas Sabatini, Associate Administrator for Regulation and Certification

Subject: ARAC Recommendations, General Structures – 25.571 Damage Tolerance

Reference: ARAC Tasking, Federal Register, dated January 20, 1995

Dear Nick,

The Transport Airplane and Engine Issues Group is pleased to submit the following as a recommendation to the FAA in accordance with the reference tasking. This information has been prepared by the General Structures Harmonization Working Group.

- GSHWG Report – FAR/JAR 25.571, Damage Tolerance and Fatigue Evaluation of Structure

The Working Group did achieve consensus on the report which was approved by TAEIG with one abstention (AIA).

Sincerely yours,

Craig R. Bolt

C. R. Bolt
Assistant Chair, TAEIG

Copy: Dionne Krebs – FAA-NWR
Mike Kaszycki – FAA-NWR
Effie Upshaw – FAA-Washington, D.C.
Andrew Kasowski - Cessna

Acknowledgement Letter

MAR 8 2004

Mr. Craig Bolt
Assistant Chair, Transport Airplanes and
Engines Issues Area
400 Main Street, MS 162-14
East Hartford, CT 01608

Dear Mr. Bolt,

This letter responds to several letters from the Aviation Rulemaking Advisory Committee (ARAC) on Transport Airplanes and Engines (TAE) during calendar year 2003.

Date of Letter: May 14

Purpose: A request for economic support for a proposed part 25 rulemaking addressing ice protection systems.

FAA Action/Status: Kathy Ishimaru, the Federal Aviation Administration (FAA) representative on the Ice Protection Harmonization Working Group, and George Thurston of the FAA Policy Office indicated that Mr. Thurston has already provided the economic data to the working group. No further action is warranted.

Date of Letter: July 22

Purpose: Transmittal package with opposing views related to the ease of search task from the members of the Design for Security Harmonization Working Group.

FAA Action/Status: At the June TAE ARAC meeting, after learning the working group could not reach consensus, Mr. Kaszycki asked the working group to document its views and forward the package to the FAA through ARAC. The package has since been forwarded to the Transport Airplane Directorate for review and decision.

We may request the working group to help us dispose of substantive comments once the comment period for the notice of proposed rulemaking closes. Hence, we consider the working group to be in existence, but in-active until further notice.

This letter also acknowledges receipt of several recommendation packages:

Date of Letter	Task No.	Description of Recommendation	Working Group
Sep 18	7	Working group report with a long term plan addressing the effects of multiple complex structural supplemental type certification modifications on the structural integrity and continued safe operations of transport category	Airworthiness Assurance

		airplanes	
Sep 19	11 9	Working group report that provides language for a requirement to substantiate the operation of the airplane control systems is not adversely affected (jamming, friction, disconnection, damage) by the presence of deflections of the airplane structure due to the separation of pitch, roll, and yaw limit maneuver loads (25.683) Working group report that provides harmonized rule language and advisory material for fuel tank access cover impact resistance (§ 25.963(e))	General Structures Harmonization
Oct 21	3, Part 1	Working group report addressing ventilation (heating and humidity), § 25.831(g)	Mechanical Systems Harmonization
Oct 21	3, Part 2	Working group report addressing cabin pressurization, § 25.841(a)	Mechanical Systems Harmonization
Oct 22 ANM-95-195	5	Working group report that provides harmonized § 25.571 language and accompanying advisory material for damage tolerance and fatigue evaluation of structure	General Structures Harmonization
Oct 22	6	Working group reports on widespread fatigue damage that address training syllabus, multiple element damage, and mandatory modifications	Airworthiness Assurance

I wish to thank ARAC and the working groups for the resources that industry gave to develop these recommendations. Since we consider submittal of the recommendation as completion of the tasks, we have closed the tasks, and placed the recommendations on the ARAC website at http://www1.faa.gov/avr/arm/arac/aracTransportAirplane.cfm?nav=6. The recommendation packages have been forwarded to the Transport Airplane Directorate for review and decision. We will continue to keep you apprised of our efforts on the ARAC recommendation at the regular ARAC meeting.

Sincerely,

Original Signed By
Nicholas A. Sabatini

Nicholas A. Sabatini
Associate Administrator for Regulation
and Certification

ARM-209:Eupshaw;fs:1/9/04; PC Docs #20579
cc: ARM-1/20/200/209; AIR-100; ANM-110
File #ANM-01-024-A; ANM-00-083-A; ANM-98-466-A; ANM-01-111-A; ANM-95-195-A.; ANM-99-969-A
Control Nos. 20032768-0, 20033095-0, 20033096-0, 20033097-0, 20033098-0, 20033099-0

Recommendation

July 2, 2003

IN REPLY, REFER TO
L350-03-115

Mr. Craig R. Bolt
Assistant Chair, TAEIG
Pratt & Whitney
400 Main Street
East Hartford, Ct 06108

Subject: **Submittal of Results of Harmonization Effort on FAR/JAR §25.571, Damage Tolerance and Fatigue Evaluation of Structure**

Dear Craig:

The General Structures Harmonization Working Group herewith submits the Working Group Report on the subject regulatory material to the TAEIG for acceptance and recommendation to the FAA.

Summary

In July of 1995, ARAC tasked the General Structures Harmonization Working Group to develop harmonized requirements and advisory material for Damage Tolerance and Fatigue Evaluation of Structure, §25.571. Technical agreement of the full Harmonization Working Group (HWG) was achieved in March of 1998 and a draft NPRM and revision to existing advisory material was developed. This material was formally submitted to the TAEIG in July of 1999 (reference Boeing Letter BYK10HLL-M99-066, dated June 29, 1999). Concurrent with the attainment of technical agreement on FAR/JAR harmonized rule and advisory material within the HWG, Amendment 96 to the FAR was released (63 FR 15708 March 31, 1998) which incorporated significant changes to FAR §25.571 and Advisory Circular 25.571-1 and thereby changed the basis upon which harmonization by the HWG was attained. In August of 1999 the GSHWG agreed to withdraw the previously submitted harmonized draft NPRM and advisory material and accept a re-tasking to reach harmonization between the JAR and FAR requirements with respect to Amendment 96 while re-introducing fail safe requirements back into the rule and advisory material and embodying the work of the AAWG with regard to continued airworthiness. In June of 2002, technical agreement was again reached within the full GSHWG on harmonized rule and advisory material for FAR/JAR §25.571, Damage Tolerance and Fatigue Evaluation of Structure.

The proposed harmonized rulemaking and accompanying advisory material contained in the Working Group Report has three main features: 1) it creates a harmonized text that is compatible with the rulemaking accomplished by the FAA at amendment 96, but harmonized with the JAR, 2) it requires a Limit of Validity (the time period in airplane flight hours or cycles over which the maintenance program is considered to be adequate) to be established for the Instructions for Continued Airworthiness provided to the operator at the time of initial certification, and 3) it establishes evaluation criterion for the amount of structure that must be considered as damaged with the remaining structure still able to carry residual strength loads (i.e. a damage-capability level that must be demonstrated to ensure that the airplane maintenance program will not be defeated by unforeseen damage sources).

The GSHWG submits this Working Group Report containing proposed rule and advisory material for §25.571, Damage Tolerance and Fatigue Evaluation of Structure, as the culmination of eight years of continuing and often controversial effort by the group to reach consensus on a very significant requirement in regard to overall and continuing aircraft safety. Special recognition goes to Amos Hoggard for his relentless encouragement of the group to attain this goal.

Sincerely,

Andrew H. Kasowski
General Structures HWG Chairperson
316-517-6008
316-517-1820 FAX
akasowski@cessna.textron.com

Attachment A

General Structures Harmonization Working Group Report

Damage Tolerance and Fatigue Evaluation of Structures
FAR/JAR §25.571

ARAC WG Report Format
Harmonization (Category 3) and New Projects

1 - What is underlying safety issue to be addressed by the FAR/JAR?

FAR 25.571 provides for the evaluation of the strength of structure in the presence of damage. FAR 25.571 provides for the establishment of requirements for maintenance programs to protect the airframe structure against the effects of fatigue.
FAR 25.1529 provides for the establishment of Instructions for Continued Airworthiness.
FAR 25 Appendix H provides the establishment of requirements for the preparation of Instructions for Continued Airworthiness required by FAR 25.1529.

2 - What are the current FAR and JAR standards relative to this subject?

Current FAR text:

§ 25.571 Damage-tolerance and fatigue evaluation of structure.

(a) *General.* An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, manufacturing defects, or accidental damage, will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of paragraphs (b) and (e) of this section, except as specified in paragraph (c) of this section, for each part of the structure that could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachments). For turbojet powered airplanes, those parts that could contribute to a catastrophic failure must also be evaluated under paragraph (d) of this section. In addition, the following apply:

(1) Each evaluation required by this section must include--

(i) The typical loading spectra, temperatures, and humidities expected in service;

(ii) The identification of principal structural elements and detail design points, the failure of which could cause catastrophic failure of the airplane; and

(iii) An analysis, supported by test evidence, of the principal structural elements and detail design points identified in paragraph (a)(1)(ii) of this section.

(2) The service history of airplanes of similar structural design, taking due account of differences in operating conditions and procedures, may be used in the evaluations required by this section.

(3) Based on the evaluations required by this section, inspections or other procedures must be established, as necessary, to prevent catastrophic

failure, and must be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness required by Sec. 25.1529.
Inspection thresholds for the following types of structure must be established based on crack growth analyses and/or tests, assuming the structure contains an initial flaw of the maximum probable size that could exist as a result of manufacturing or service-induced damage:

(i) Single load path structure, and

(ii) Multiple load path "fail-safe" structure and crack arrest "fail-safe" structure, where it cannot be demonstrated that load path failure, partial failure, or crack arrest will be detected and repaired during normal maintenance, inspection, or operation of an airplane prior to failure of the remaining structure.

(b) *Damage-tolerance evaluation.* The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage.
Repeated load and static analyses supported by test evidence and (if available) service experience must also be incorporated in the evaluation. Special consideration for widespread fatigue damage must be included where the design is such that this type of damage could occur. It must be demonstrated with sufficient full-scale fatigue test evidence that widespread fatigue damage will not occur within the design service goal of the airplane. The type certificate may be issued prior to completion of full-scale fatigue testing, provided the Administrator has approved a plan for completing the required tests, and the airworthiness limitations section of the instructions for continued airworthiness required by Sec. 25.1529 of this part specifies that no airplane may be operated beyond a number of cycles equal to ½ the number of cycles accumulated on the fatigue test article, until such testing is completed. The extent of damage for residual strength evaluation at any time within the operational life of the airplane must be consistent with the initial detectability and subsequent growth under repeated loads. The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

(1) The limit symmetrical maneuvering conditions specified in Sec. 25.337 at all speeds up to V_C and in Sec. 25.345.**]**

(2) The limit gust conditions specified in Sec. 25.341 at the specified speeds up to V_C and in Sec. 25.345.

(3) The limit rolling conditions specified in Sec. 25.349 and the limit unsymmetrical conditions specified in Secs. 25.367 and 25.427(a) through (c), at speeds up to V_C.

(4) The limit yaw maneuvering conditions specified in Sec. 25.351(a) at the specified speeds up to V_C.

(5) For pressurized cabins, the following conditions:

(i) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in paragraphs (b)(1) through (4) of this section, if they have a significant effect.

(ii) The maximum value of normal operating differential pressure (including the expected external aerodynamic pressures during 1*g* level flight) multiplied by a factor of 1.15, omitting other loads.

(6) For landing gear and directly-affected airframe structure, the limit ground loading conditions specified in Secs. 25.473, 25.491, and 25.493.

If significant changes in structural stiffness of geometry, or both, follow from a structural failure, or partial failure, the effect on damage tolerance must be further investigated.

(c) *Fatigue (safe-life) evaluation.* Compliance with the damage-tolerance requirements of paragraph (b) of this section is not required if the applicant establishes that their application for particular structure is impractical. This structure must be shown by analysis, supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe-life scatter factors must be applied.

(d) *Sonic fatigue strength.* It must be shown by analysis, supported by test evidence, or by the service history of airplanes of similar structural design and sonic excitation environment, that--

(1) Sonic fatigue cracks are not probable in any part of the flight structure subject to sonic excitation; or

(2) Catastrophic failure caused by sonic cracks is not probable assuming that the loads prescribed in paragraph (b) of this section are applied to all areas affected by those cracks.

(e) *Damage-tolerance (discrete source) evaluation.* The airplane must be capable of successfully completing a flight during which likely structural damage occurs as a result of--

(1) Impact with a 4-pound bird when the velocity of the airplane relative to the bird along the airplane's flight path is equal to V_C at sea level or 0.85 V_C at 8,000 feet, whichever is more critical;**]**

(2) Uncontained fan blade impact;

(3) Uncontained engine failure; or

(4) Uncontained high energy rotating machinery failure.

The damaged structure must be able to withstand the static loads (considered as ultimate loads) which are reasonably expected to occur on the flight. Dynamic effects on these static loads need not be considered. Corrective action to be taken by the pilot following the incident, such as limiting maneuvers, avoiding turbulence, and reducing speed, must be considered. If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on damage tolerance must be further investigated.

Amdt. 25-96, Eff. 3/31/98

§ 25.1529 Instructions for Continued Airworthiness.

The applicant must prepare Instructions for Continued Airworthiness in accordance with appendix H to this part that are acceptable to the Administrator. The instructions may be incomplete at type certification if a program exists to ensure their completion prior to delivery of the first airplane or issuance of a standard certificate of airworthiness, whichever occurs later.

§ H25.4 *Airworthiness Limitations section*.

(a) The Instructions for Continued Airworthiness must contain a section titled Airworthiness Limitations that is segregated and clearly distinguishable from the rest of the document. This section must set forth-

(1) Each mandatory replacement time, structural inspection interval, and related structural inspection procedure approved under § 25.571; and

(2) Each mandatory replacement time, inspection interval, related inspection procedure, and all critical design configuration control limitations approved under § 25.981 for the fuel tank system.

(b) If the Instructions for Continued Airworthiness consists of multiple documents, this section required by this paragraph must be included in the principle manual. This section must contain a legible statement in a prominent location that reads: "The Airworthiness Limitations section is FAA approved and specifies maintenance required under §§ 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA approved."

<u>Current JAR text</u>:

JAR 25.571 Damage-Tolerance And Fatigue Evaluation Of Structure

(a) *General*. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage, will be avoided throughout the operational life of the aeroplane. This evaluation must be conducted in accordance with the provisions of sub-paragraphs (b) and (e) of this paragraph, except as specified in sub-paragraph (c) of this paragraph, for each part of the structure which could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachments). (See ACJ 25.571(a).) For turbine engine powered aeroplanes, those parts which could contribute to a catastrophic failure must also be evaluated under sub-paragraph (d) of this paragraph. In addition, the following apply:

(1) Each evaluation required by this paragraph must include--

(i) The typical loading spectra, temperatures, and humidities expected in service;

(ii) The identification of principal structural elements and detail design points, the failure of which could cause catastrophic failure of the aeroplane; and

(iii) An analysis, supported by test evidence, of the principal structural elements and detail design points identified in sub-paragraph (a)(1)(ii) of this paragraph.

(2) The service history of aeroplanes of similar structural design, taking due account of differences in operating conditions and procedures, may be used in the evaluations required by this paragraph.

(3) Based on the evaluations required by this paragraph, inspections or other procedures must be established as necessary to prevent catastrophic failure, and must be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness required by JAR 25.1529.

(b) *Damage-tolerance* (*fail-safe*) *evaluation*. The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. The determination must be by analysis supported by test evidence and (if available) service experience. Damage at multiple sites due to prior fatigue exposure must be included where the design is such that this type of damage can be expected to occur. The evaluation must incorporate repeated load and static analyses supported by test evidence. The extent of damage for residual strength evaluation at any time within the operational life must be consistent with the initial detectability and subsequent growth under repeated loads. The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

(1) The limit symmetrical manoeuvring conditions specified in JAR 25.337 up to V_C and in JAR 25.345.

(2) The limit gust conditions specified in JAR 25.341 at the specified speeds up to V_C and in JAR 25.345.

(3) The limit rolling conditions specified in JAR 25.349 and the limit unsymmetrical conditions specified in JAR [25.367 and JAR 25.427(a) through (c), at] speeds up to V_C.

(4) The limit yaw manoeuvring conditions specified in JAR 25.351 at the specified speeds up to V_C.

(5) For pressurised cabins, the following conditions:

(i) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in sub-paragraphs (b)(1) to (b)(4) of this paragraph if they have a significant effect.

(ii) The maximum value of normal operating differential pressure (including the expected external aerodynamic pressures during 1 g level flight) multiplied by a factor of 1·15 omitting other loads.

(6) For landing gear and directly-affected airframe structure, the limit ground loading conditions specified in JAR 25.473, JAR 25.491 and JAR 25.493.

If significant changes in structural stiffness or geometry, or both, follow from a structural failure, or partial failure, the effect on damage tolerance must be further investigated. (See ACJ 25.571(b).) The residual strength requirements of this sub-paragraph (b) apply, where the critical damage is not readily detectable. On the other hand, in the case of damage which is readily detectable within a short period, smaller loads than those of sub-paragraphs (b)(1) to (b)(6) inclusive may be used by agreement with the Authority. A probability approach may be used in these latter assessments, substantiating that catastrophic failure is extremely improbable. (See ACJ 25.571(a), paragraph 2.1.2.)

(c) *Fatigue (safe-life) evaluation*. Compliance with the damage-tolerance requirements of sub-paragraph (b) of this paragraph is not required if the applicant establishes that their application for particular structure is impractical. This structure must be shown by analysis, supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe-life scatter factors must be applied.

(d) *Sonic fatigue strength*. It must be shown by analysis, supported by test evidence, or by the service history of aeroplanes of similar structural design and sonic excitation environment, that--

(1) Sonic fatigue cracks are not probable in any part of the flight structure subject to sonic excitation; or

(2) Catastrophic failure caused by sonic cracks is not probable assuming that the loads prescribed in sub-paragraph (b) of this paragraph are applied to all areas affected by those cracks.

(e) *Damage-tolerance (discrete source) evaluation*. The aeroplane must be capable of successfully completing a flight during which likely structural damage occurs as a result of--

(1) Bird impact as specified in JAR 25.631;

(2) Reserved

(3) Reserved

(4) Sudden decompression of compartments as specified in JAR 25.365(e) and (f).

The damaged structure must be able to withstand the static loads (considered as ultimate loads) which are reasonably expected to occur at the time of the occurrence and during the completion of the flight. Dynamic effects on these static loads need not be considered. Corrective action to be taken by the pilot following the incident, such as limiting manoeuvres, avoiding turbulence, and reducing speed, may be considered. If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on damage tolerance must be further investigated. (See ACJ 25.571(a), paragraph 2.7.2 and ACJ 25.571(b).)

JAR 25.1529 Instructions for Continued Airworthiness.

The applicant must prepare Instructions for Continued Airworthiness in accordance with appendix H to this part that are acceptable to the Authority. The instructions may be incomplete at type certification if a programme exists to ensure their completion prior to delivery of the first aeroplane or issuance of a certificate of airworthiness, whichever occurs later.

JAR H25.4 ***Airworthiness Limitations section*****.**

The Instructions for Continued Airworthiness must contain a section titled Airworthiness Limitations that is segregated and clearly distinguishable from the rest of the document. This section must set forth each mandatory replacement time, structural inspection interval, and related structural inspection procedure approved under JAR 25.571. If the Instructions for Continued Airworthiness consists of multiple documents, the section required by this paragraph must be included in the principal manual. This section must contain a legible statement in a prominent location that reads: “The Airworthiness Limitations section is approved and variations must also be approved.”

2a – If no FAR or JAR standard exists, what means have been used to ensure this safety issue is addressed?

N/A

3 - What are the differences in the FAA and JAA standards or policy and what do these differences result in?

Amendment 96 of FAR 25.571 contains full-scale fatigue test evidence requirements that the JAR does not. In addition to issuing a recommendation to the FAA to revise the Safe-life section of FAR 25.571, the GSHWG was requested to accomplish the following harmonization work on FAR/JAR 25.571:

a. Harmonize FAR 25.571 with JAR 25.571 to address the changes in the full-scale fatigue test requirements introduced in FAR 25.571 at amendment 25-96;

b. Revise FAR/JAR 25.571 rule for compatibility with the Widespread Fatigue Damage rulemaking of FAA;

c. Incorporate fail-safe concepts of pre-amendment 45 versions of FAR 25.571 into both the FAR and JAR.

The JAR does not presently have the same full-scale fatigue test requirements as FAR 25.571. Neither the JAR nor the FAR have requirements for a Limit of Validity (LOV) for maintenance programs. The FAR operational rules (14 CFR Parts 121 and 129) are being changed to include such a limitation, and the JAR will have a compatible way of managing freedom from widespread fatigue damage, based on a maintenance program with a Limit of Validity. Neither the FAR nor the JAR presently contain a requirement for any fail-safe design features. However, as a result of investigations after the Aloha Airlines accident of 1988, it was jointly concluded by the European and American aviation authorities that a change of this kind was necessary to ensure that the airplane structure has some level of robustness in the presence of small cracks.

4 - What, if any, are the differences in the current means of compliance?

N/A

5 – What is the proposed action?

The proposed rulemaking has three main features:

a. It creates a harmonized text that is compatible with the rulemaking accomplished by the FAA at amendment 96, but harmonized with the JAR. This is basically the same requirement adopted by the FAA at amendment 96, with changes introduced to create an improved harmonized standard. The prescriptive requirement of existing FAR paragraph 25.571(a)(3) for setting damage-tolerance inspection thresholds based on crack growth, assuming the structure has an initial flaw has been abandoned in favor of a performance based requirement (proposed 25.571(a)(5)), which states that inspection thresholds must be established to ensure that cracking will be detected before it result in a catastrophic failure, and must account for variations in manufacturing quality

b. The regulation requires a Limit of Validity (LOV) to be established for the Instructions for Continued Airworthiness provided to the operator at the time of initial FAA certification. Although Instructions for Continued Airworthiness had previously been required by 25.571and 25.1529 (inspections, replacements, etc.), these instructions were established without respect to the age of the airplane. It has been recognized since the Aloha Airlines accident of 1988, that although the Instructions for Continued Airworthiness established at the time of certification are valid for a new airplane, they cannot be relied on to be as effective throughout the life of the airplane, as deleterious age related effects accumulate in the structure. Therefore, it is now recognized that the maintenance program established at the time of certification program should be limited to the time scale to which it was analyzed and tested for at the time of original certification. After that time period has passed (aircraft hours or flight cycles) it will be incumbent on the airplane operator to obtain new or revised Instructions for Continued Airworthiness that are compatible with the second phase of the airplane's life. The proposed regulation imposes a requirement for establishing a Limit of Validity (LOV) for the Instructions for Continued Airworthiness in FAR and JAR 25.

c. A new section of the rule was created, establishing a requirement for damage capability. This new requirement grew out of a concern that the rule adopted at amendment 45 did not contain a design requirement in regard to the smallest size of structural damage that can be tolerated between inspection intervals. This has led to situations in which fatigue safety is managed by inspecting to find damage that can only be detected through artificial means. It is expected that the inspections will be able to safely detect the damage for which it was intended, if it develops and progresses according to computations. However, service experience has shown that damage of significant size frequently develops on airframes in ways that were never anticipated by designers. This damage is usually found before it represents a hazard to the airplane by means other than the fatigue damage inspection program established by the fatigue analyses and evaluations required by FAR 25.571. It is postulated that this kind of damage has been detected because manufacturers have historically designed to a fail-safe philosophy, even though not specifically required to do so by post amendment 45 versions of 25.571. This design philosophy results in the manufacturer providing a generalized structural capability in the presence of damage, so that even if the structure "fails" partially, there will still be enough structure remaining to be "safe." To a large extent this philosophy ignores the details of the way damage can develop, but simply assumes a certain part of the structure will fail, and requires that the rest of the structure can sustain the appropriate residual strength loads.

   In implementing this reintroduction of a fail-safe concept back into the regulation it became necessary to adopt a somewhat different point of view than the earlier "fail-safe" one. The fail-safe concept does not apply well in several cases: If the airplane has a unitized piece of structure, which is not subdivided into individual components, it is not objectively clear what the individual elements or load paths are; and for structure composed of very small individual pieces, what constitutes an individual element. The damage-capability requirement establishes evaluation criterion for the amount of structure that has to be considered as damaged with the remaining structure still able to carry residual strength load.

For each proposed change from the existing standard, answer the following questions:

6 - What should the harmonized standard be?

§ 25.571 Damage-tolerance and fatigue evaluation of structure.

(a) *General*. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage, will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of paragraphs (b), (e), (f), and (g) of this section, except as specified in paragraph (c) of this section, for each part of the structure that could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachments). For turbine engine powered airplanes, those parts that could contribute to a catastrophic failure must also be evaluated under paragraph (d) of this section. In addition, the following apply:

(1) Each evaluation required by this section must include –

(i) The determination of typical loading spectra, temperatures, and humidities expected in service;

(ii) The identification of principal structural elements (PSE) and detail design points, the failure of which could contribute to a catastrophic failure of the airplane; and

(iii) An analysis, supported by test evidence, of the principal structural elements and detail design points identified in paragraph (a)(1)(ii) of this section.

(2) The service history of airplanes of similar structural design, taking due account of differences in operating conditions and procedures, may be used in the evaluations required by this section.

(3) Based on the evaluations required by this section, inspections or other procedures must be established, as necessary, to prevent catastrophic failure, and must be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness required by §25.1529. The limit of validity (LOV) of this maintenance program must also be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness required by §25.1529.

(4) Damage tolerant design is primarily associated with the use of multiple load path structure or structure that contains damage containment features that significantly retard or arrest a crack.

(5) When special inspections are required to prevent catastrophic fatigue failure, inspection thresholds must be established to ensure that cracking in a PSE will be detected before it results in a catastrophic failure. The inspection thresholds must account for the variations of manufacturing quality.

(6) Inspection programs for corrosion and service induced accidental damage must be proposed to protect the structure against catastrophic failure.

(b) *Damage-tolerance evaluation*. The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. The evaluation must also incorporate repeated load and static analyses supported by test evidence and (if available) service experience. It must be demonstrated with sufficient full-scale fatigue test evidence that widespread fatigue damage will not occur within the limit of validity of the maintenance program for the airplane. If full-scale fatigue testing is conducted as part of the type certification program, then the type certificate may be issued prior to completion of that testing, provided that the Administrator has approved a plan for completing the required tests and analysis, and that at least one calendar year of safe operation has been substantiated at the time of type certification.

The extent of damage for residual strength evaluation at any time within the operational life of the airplane must be consistent with the initial detectability and subsequent growth under repeated loads.

The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

(1) The limit symmetrical maneuvering conditions specified in §25.337 at all speeds up to V_C and in §25.345.

(2) The limit gust conditions specified in §25.341 at the specified speeds up to V_C and in §25.345.

(3) The limit rolling conditions specified in §25.349 and the limit unsymmetrical conditions specified in §§25.367 and 25.427(a) through (c), at speeds up to V_C.

(4) The limit yaw maneuvering conditions specified in §25.351 at the specified speeds up to V_C.

(5) For pressurized cabins, the following conditions:

(i) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in paragraphs (b)(1) to (b)(4) of this section, if they have a significant effect.

(ii) The maximum normal operating differential pressure multiplied by a factor of 1.15, combined with the expected external aerodynamic pressures during 1g level flight, omitting other loads.

(6) For landing gear and other affected airframe structure, the limit ground loading conditions specified in §§25.473, 25.491, and 25.493.

If significant changes in structural stiffness or geometry, or both, follow from a structural failure, or partial failure, the effect on damage tolerance must be further evaluated.

(c) *Fatigue (safe-life) evaluation*. Compliance with the damage-tolerance requirements of paragraph (b) of this section is not required if the applicant establishes that their application for the particular structure is impractical. This structure must be shown by analysis, supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe-life scatter factors must be applied.

(d) *Sonic fatigue strength*. It must be shown by analysis, supported by test evidence, or by the service history of airplanes of similar structural design and sonic excitation environment, that –

(1) Sonic fatigue cracks are not probable in any part of the flight structure subject to sonic excitation; or

(2) Catastrophic failure caused by sonic fatigue cracks is not probable assuming that the loads prescribed in paragraph (b) of this section are applied to all areas affected by those cracks.

(e) *Discrete source damage evaluation*. The airplane must be capable of successfully completing a flight during which likely structural damage occurs as a result of –

(1) Impact with a 4-pound bird when the velocity of the airplane relative to the bird along the airplane's flight path is equal to V_C at sea level or $0.85V_C$ at 8,000 feet, whichever is more critical;

(2) Uncontained engine rotor failure;

(3) Uncontained APU rotor failure; or
(4) Revoked.

The damaged structure must be able to withstand the static loads (considered as ultimate loads) which are reasonably expected to occur at the time of the occurrence and during the completion of the flight. Dynamic effects on these static loads need not be considered. Corrective action to be taken by the pilot following the incident, such as limiting maneuvers, avoiding turbulence, and reducing speed, may be considered. If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on damage tolerance must be further investigated.

(f) *Structural damage capability*.

1) Except as noted in subparagraph f(2), for structure evaluated according to the damage-tolerance requirements of paragraph (b) of this section, it must be shown by analysis, supported by test evidence, that the structure is able to withstand the loads specified in paragraphs (b)(1) to (b)(6) of this section in the presence of damage equivalent to:

i) the complete failure of any single element, or
ii) partial failure between damage containment features that significantly retard or arrest a crack

2) For single load path structure, the intent of the SDC requirement shall be achieved through the demonstration of slow crack growth, an upper bound inspection threshold of 50% DSG and consideration of the quality control procedures used in manufacture. The requirement for an upper bound inspection threshold of 50% DSG may be extended based upon a rational analysis that is approved by the Administrator.

(g) *Inspectability.* The inspectability of the extent of damage established in accordance with paragraph (f) must be addressed and reflected in the threshold determination required by paragraph (a)(5).

§ 25.1529 Instructions for Continued Airworthiness.

The applicant must prepare Instructions for Continued Airworthiness in accordance with appendix H to this part that are acceptable to the Administrator. The instructions may be incomplete at type certification if a program exists to ensure their completion prior to delivery of the first airplane or issuance of a standard certificate of airworthiness, whichever occurs later.

Editorial Note: The word "standard" in the last sentence of the proposed harmonized text of § 25.1529 above is recognized as an editorial difference between the FAR and JAR and will not appear in the harmonized version of the JAR per NPA25C-312.

§ H25.1 *General*.

(d) The applicant must consider the effect of aging of structures in the Instructions for Continued Airworthiness. (See AC 91-56 or GAI ACJ 20X11).

§ H25.4 *Airworthiness Limitations section*.

The Instructions for Continued Airworthiness must contain a section titled Airworthiness Limitations that is segregated and clearly distinguishable from the rest of the document. This section must set forth:

1) Each mandatory replacement time, structural inspection threshold and interval, and related structural inspection procedure approved under § 25.571;

2) Elements of the corrosion prevention and control program and the accidental damage detection program that are required under § 25.571(b) to prevent catastrophic failure of the aircraft due to fatigue cracking;

3) A limit of validity of the maintenance program for the prevention of WFD; and

4) Each mandatory replacement time, inspection interval, related inspection procedure, and all critical design configuration control limitations approved under § 25.981 for the fuel tank system.

If the Instructions for Continued Airworthiness consist of multiple documents, this section required by this paragraph must be included in the principle manual. This section must contain a legible statement in a prominent location that reads: "The Airworthiness Limitations section is FAA approved and specifies maintenance required under §§ 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA approved."

Editorial Note: This last sentence in the ACJ will read as follows and is considered harmonized per NPA 25C-312: This section must contain a legible statement in a prominent location that reads: "The Airworthiness Limitations section is approved and variations must also be approved". In addition, the extra requirement in the FAR version of §H25.4 for fuel tank systems will not be included in the JAR since this is not within the scope of the GSHWG tasking.

7 - How does this proposed standard address the underlying safety issue (identified under #1)?

The changes in the proposed harmonized standard address the underlying safety issue in the following way:

a. It retains full-scale fatigue test evidence as a requirement for demonstrating that the airplane will not be subject to WFD during its operational usage.
b. It establishes a Limit of Validity (LOV) for the maintenance program which sets the time period (in airplane flight hours or cycles) over which the maintenance program is considered to be adequate.
c. It establishes a damage-capability level that must be demonstrated to ensure that the airplane maintenance program will not be defeated by unforeseen damage sources.

8 - Relative to the current FAR, does the proposed standard increase, decrease, or maintain the same level of safety?

a. The full-scale fatigue test requirement of amendment 96 is retained.
b. The introduction of a Limit of Validity (LOV) maintains the current level of safety, but provides for compatibility with the proposed FAA operational rules, and relieves the FAA from having to take individual rulemaking action against individual models as their maintenance programs expire.
c. The damage-capability requirement represents an increased level of safety over the existing rule. Presently the regulations allow for inspecting for small cracks to maintain safety; this would be eliminated in most cases under the proposed rule.

9 - Relative to current industry practice does the proposed standard increase, decrease, or maintain the same level of safety?

a. The proposed standard maintains the level of safety relative to industry practice in regard to compliance with Amendment 96 requirements of the FAR. The harmonized changes consolidate the European and US standards and generally capture what has been industry practice.
b. The proposed standard relative to industry practice in regard to the establishment of a Limit of Validity would increase the level of safety since it establishes the point where new maintenance actions must be considered for the prevention of widespread fatigue damage.
c. The proposed standard supports broad industry historical practice to include fail-safe concepts in damage tolerant designs, although not required specifically to do so by current regulations. In some cases the proposed standard maintains the level of safety relative to industry practice, in others it increases the level of safety because the damage capability evaluation is now based on damage tolerance concepts. In other instances, some manufacturers have incorporated a greater amount of damage capability on some structure than the amount required by the minimum standard delineated in the proposed standard. Again, this practice is exercised by some

manufacturers despite the lack of a specific requirement to do so. Under the proposed standard it is anticipated that manufacturers will continue to produce designs that in many cases exceed the minimum requirements.

10 - What other options have been considered and why were they not selected?

Options were not considered for harmonization of the Amendment 96 text, since this requirement was retained in the FAR, except for improved text.

As an option to setting a limitation to the maintenance program, a limitation on the life of the airplane was considered. This was rejected, since appropriately designed modifications and revised inspections could safely extend the useful life of the airplane. It was judged to be more appropriated to time limit the airplane's maintenance program developed at the time of airplane certification.

A fail-safe design option was discussed in lieu of the damage-capability evaluation requirement, however, it was felt that the older fail-safe design requirement could not be applied consistently. Although the older fail-safe design requirement works most of the time on typical airframe structure, in which the sizes of the structural elements are fairly large compared to a human being, it results in inconsistencies when applied to unitized structures or structures that have many small parts. The fail-safe concept does not work well for monolithic structures, where there are no individual elements to consider failed, and where discrete load paths have to be inferred or defined. It also does not work well for structures in which the individual elements are of very small size. Therefore, it was considered necessary to develop a new concept, damage-capability, that would be more compatible with the current generation of airplane structural designs.

11 - Who would be affected by the proposed change?

Manufacturers of new type designs would be affected by the damage tolerance evaluation requirements and the damage-capability requirements. Airplane operators would be affected because the new requirements would have a limitation on the Instructions for Continued Airworthiness. After exceedance of the Limit of Validity (LOV) for the document as defined at the time of initial certification, the document would have to be updated and re-approved in order for the airplane to remain in service.

12 - To ensure harmonization, what current advisory material (e.g., ACJ, AMJ, AC, policy letters) needs to be included in the rule text or preamble?

The new text for the rule and AC are provided herein, and this does not include the inclusion of current advisory material in the Rule.

13 - Is existing FAA advisory material adequate? If not, what advisory material should be adopted?

New advisory material is provided below:

Damage Tolerance and Fatigue - 25.571
Draft AC 25.571-1x Rev. 6 (2/20/03)

1. PURPOSE. This advisory circular (AC) provides guidance for compliance with the provisions of 14 CFR Part 25 of the Federal Aviation Regulations (FAR) pertaining to the damage-tolerance and fatigue evaluation requirements for airplane metallic and non-metallic structure. It also provides rational guidelines for the evaluation of scatter factors for the determination of life for parts categorized as safe-life. Additional Guidance material for non-metallic structures is contained in AC 20-107A. Like all advisory circular material, this advisory circular is not, in itself, mandatory and does not constitute a regulation. It is issued to provide an acceptable means, but not the only means, of compliance with the rules. Terms used in this AC, such as "shall" and "must" are used only in the sense of ensuring applicability of this particular method of compliance when the acceptable method of compliance described herein is used. While these guidelines are not mandatory, they are derived from extensive FAA and industry experience in determining compliance with the pertinent FAR. This advisory circular does not change, create any additional, authorize changes in, or permit deviations from, regulatory requirements.

2. CANCELLATION. Advisory Circular 25.571-1C, dated April 29, 1998, is canceled.

3. *References*
AC20-107A
AC91-56B
25.1529

4. DEFINITIONS OF TERMS USED IN THIS AC.

a. 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 use after the structure has sustained a given level of fatigue, corrosion, accidental, or discrete source damage.

b. Structural Damage Capability, SDC, is the attribute of the structure which permits it to retain its required residual strength in the presence of large damage.

c. Safe-life of a structure is that number of events such as flights, landings, or flight hours, during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue cracking.

d. Design Service Goal, DSG, is the period of time in flight hours/cycles/years, established at design and/or certification that represents the initially anticipated operational life of the airplane.

e. Principal structure element, PSE, is an element that contributes significantly to the carrying of flight, ground, or pressurization loads, and whose integrity is essential in maintaining the overall structural integrity of the airplane.

f. Detail Design Point, (DDP), is an area within a PSE where fatigue cracking is likely to occur and where the damage tolerance assessment is made.

g. Single Load Path, SLP, is where the applied loads are carried through a single member, the failure of which would result in the loss of the structural capability to carry the applied loads.

h. Damage containment features are specific design characteristics of a load-carrying member within the structure which are introduced in order to significantly retard or arrest a crack and enhance the capability to carry the applied loads in the event of partial failure of that member.

i. Multiple Load Path, MLP, is identified with redundant structures in which (with the failure of individual elements) the applied loads would be safely distributed to other load-carrying members.

j. Widespread Fatigue Damage, WFD, in a structure is characterized by the simultaneous presence of cracks at multiple structural details that are of sufficient size and density whereby the structure will no longer meet its damage tolerance requirement (i.e., to maintain its required residual strength after partial structural failure).

(1) Multiple Site Damage, MSD, is a source of widespread fatigue damage characterized by the simultaneous presence of fatigue cracks in the same structural element (i.e. fatigue cracks that may coalesce with or without other damage leading to a loss of required residual strength).

(2) Multiple Element Damage, MED, is a source of widespread fatigue damage characterized by the simultaneous presence of fatigue cracks in similar adjacent structural elements.

k. Scatter factor is a life reduction factor used in the interpretation of fatigue analysis and fatigue test results.

l. Limit of Validity ,LOV, is the period of time, expressed in appropriate units (e.g. flight cycles), for which it has been shown that the established inspections and replacement times will be sufficient to preclude development of wide spread fatigue damage.

m. Special Inspections are inspections other than those included in the normal maintenance or AD/ED inspection program for the airplane necessary to assure the continued airworthiness of the airplane

n. Normal maintenance – is understood to be those scheduled maintenance checks during minor or base maintenance inputs, normally associated with a zonal programme, requiring general visual inspections. The zonal programme is a collective term comprising selected general visual inspections and visual checks that are applied to each zone, defined by access and area, to check system and powerplant installations and structure for security and general condition. A general visual inspection is a visual examination of an interior or exterior area, installation or assembly to detect obvious damage, failure or irregularity. This level of inspection is made from within touching distance unless otherwise specified. A mirror may be necessary to enhance visual access to all exposed surfaces in the inspection area. This level of inspection is made under normally available lighting conditions such as daylight, hangar lighting, flashlight or drop-light and may require removal or opening of access panels or doors. Stands, ladders or platforms may be required to gain access.

5. BACKGROUND.

a. Since the early 1970's, there have been significant state-of-the-art and industry-practice developments in the area of structural fatigue and fail-safe strength evaluation of transport category airplanes. Recognizing that these developments could warrant some revision of the existing fatigue requirements of §§ 25.571 and 25.573 of 14 CFR Part 25, the Federal Aviation Administration (FAA), on November 18,1976, (41 FR 50956) gave notice of the Transport Category Airplane Fatigue Regulatory Review Program and invited interested persons to submit proposals to amend those requirements. The proposals and related discussions formed the basis for the revision of the structural fatigue evaluation standards of §§ 25.571 and 25.573 and the development of guidance material. To that end, § 25.571 was revised, § 25.573 was deleted (the scope of § 25.571 was expanded to cover the substance of the deleted section), and guidance material (AC 25.571-1) was provided which contained compliance provisions related to the proposed changes.

b. Since issuance of AC 25.571-1 on 9/28/78, additional guidance material, including information regarding discrete source damage, was developed and incorporated in revision -1A on 3/5/86. The AC was further revised on 2/18/97, revision 1B, to add guidance on the elements to be considered in developing safe life scatter factors for certification. The AC was further revised on 4/29/98, revision 1C, to add guidance material whose objective was to preclude widespread fatigue damage (resulting from MSD or MED) from occurring within the Design Service Goal (DSG) of the airplane, and to aid in the determination of thresholds for fatigue inspection and/or other special fleet actions. This current revision now harmonizes the revision 1C with the Joint Aviation Regulations ACJ material and industry practice. Further the AC has been revised to provide guidance for establishing a limit of validity (LOV) for the maintenance program and for meeting the new requirement for Structural Damage Capability.

6. INTRODUCTION.

a. The contents of this Advisory Circular are considered by the JAA/FAA in determining compliance with the requirements of JAR/FAR 25.571. The objective is to prevent catastrophic structural failures caused by fatigue, environmental (e.g. corrosion) or accidental damage (FD, ED, or AD). The requirements can be grouped into two different categories. One involves the establishment of mandatory maintenance actions and the other involves design. Taken together, these result in a structure where the combination of design characteristics and mandatory maintenance actions will serve to preclude any failure due to FD, ED, or AD.

§25.571(a)(6) requires development of inspections for ED and AD. §25.571(b) and (c) address FD and require establishment of inspections and replacement times respectively based on the damage tolerance (i.e. crack growth and residual strength) and fatigue characteristics of the structure. §25.571(d) requires the structure to be designed such that sonic fatigue cracking will not result in a failure. §25.571(e) requires the structure to be designed to withstand damage caused by specified threats such that the flight during which the damage is sustained can be completed. §25.571(f) requires the structure to have a minimum structural damage capability.

(1) *<u>§25.571(a)(6) – Environmental and accidental damage inspections</u>.* Inspections for ED and AD must be defined. Special consideration should be given to those areas where past service experience indicates a particular susceptibility to attack by the environment or vulnerability to impact and/or abuse. It is intended that these inspections will be effective in discovering ED or AD soon after it appears or occurs and that the ED or AD will therefore be removed/repaired before it presents a significant risk. Typically these inspections are defined based on past service experience using a qualitative Maintenance Steering Group (MSG-3) process. Those AD and ED inspections which are required to prevent catastrophic failure of the airplane due to fatigue cracking must be included in the Airworthiness Limitations Section (ALS) of the Instructions for Continued Airworthiness (ICA) required by §25.1529.

(2) *<u>§25.571(b) and (c) – Fatigue damage inspections or replacement times</u>.* Inspections for fatigue damage or replacement times must be established as necessary. These actions must be based on quantitative evaluations of the fatigue characteristics of the structure. In general analysis and testing will be required to generate the information needed.

(i) <u>Inspection or Replacement</u>. Compliance with §25.571(b) is required unless it can be demonstrated to the satisfaction of the authority that compliance cannot be shown due to practical constraints. Under these circumstances compliance with §25.571(c) is required. A typical example of structure where compliance with the requirements of §25.571(c), in lieu of §25.571(b), might be accepted would be the landing gear and its attachments.

ii) <u>ALS of the ICA</u>. All inspections and replacement times determined to be necessary, based on the damage tolerance and fatigue evaluations, must be included in the Airworthiness Limitations Section (ALS) of the Instructions for Continued Airworthiness (ICA) required by §25.1529.

iii) <u>Limit of Validity</u>. A Limit of Validity (LOV) for the maintenance program must also be determined and included in the ALS of the ICA. The LOV is the period of time, expressed in appropriate units (e.g. flight cycles), for which it has been shown that the established inspections and replacement times will be sufficient to preclude development of widespread fatigue damage. See section 10.0 of this AC for additional guidance on the LOV.

b. *Typical Loading Spectrum Expected in Service.* The loading spectrum should be based on measured statistical data of the type derived from government and industry load history studies and, where insufficient data are available, on a conservative estimate of the anticipated use of the airplane. The development of the loading spectrum includes the definition of the expected flight plan, which involves ground maneuvers, climb, cruise, descent, flight times, operating speeds, weights and altitudes and the approximate time to be spent in each of the operating regimes. The principal loads that should be considered in establishing a loading spectrum are flight loads (gust and maneuver), ground loads (taxiing, landing impact, turning, engine run-up, braking, thrust reversing and towing), and pressurization loads. Operations for crew training and other pertinent factors, such as the dynamic stress characteristics of any flexible structure excited by turbulence or buffeting should also be considered. For pressurized cabins, the loading spectrum should include the repeated application of the normal operating differential pressure and the superimposed effects of flight loads and aerodynamic pressures.

c. *Areas to be Evaluated.* In assessing the possibility of serious fatigue failures, the design should be examined to determine probable points of failure in service. In this examination, consideration should be given, as necessary, to the results of stress analyses, static tests, fatigue tests, strain gage surveys, tests of similar structural configurations, and service experience. Service experience has shown that special attention should be focused on the design details of important discontinuities, main attach fittings, tension joints, splices, and cutouts such as windows, doors, and other openings. Locations prone to accidental damage (such as that due to impact with ground servicing equipment near airplane doors) or to corrosion should be identified for analysis.

d. *Analyses and Tests.* Fatigue and damage tolerance analyses should be conducted unless it is determined that the normal operating stresses are of such a low order that crack initiation and/or significant crack growth is extremely improbable. Any method used in the analyses should be supported by test or service experience. Typical (average) values of fracture mechanics material properties may be used in residual strength and crack growth analyses. The effects of environment on these properties should be accounted for if significant.

Generally testing will also be necessary to support compliance with §25.571(b) or (c). The nature and extent of testing of complete structures or portions will depend on applicable previous design and structural tests and service experience with similar structures. Structural areas such as attachment fittings, major joints, changes in section, cutouts and discontinuities almost always require some level of testing in addition to analysis. When less than the complete structure is tested care should be taken to ensure that the internal loads and boundary conditions are valid. Any tests conducted to support the identification of areas for evaluation should be conducted to at least two times the design service goal to obtain information on crack initiation times and locations.

e. *Discrete Source Damage*. It must be shown that the airplane is capable of successfully completing a flight during which specified incidents occur and result in immediately obvious damage. The maximum extent of the damage must be quantified and the structure shown to be capable of sustaining the maximum load (considered as ultimate) expected during the completion of the flight. . There are no maintenance actions that result from this evaluation.

f. *Structural Damage Capability.* It must be shown that all structure that is evaluated in accordance with the requirements of §25.571(b) also has a minimum specified level of structural damage capability. The intent of this requirement is to mandate a minimum level of robustness and therefore tolerance to damage irrespective of type or source. There are no maintenance actions that result from this evaluation.

(g) *Inspectability.* The degree to which the extent of damage shown to comply with the SDC requirements of §25.571(f) can be detected by visual inspections during normal maintenance or during AD/ED inspections. Inspectability should be accounted for in the determination of the threshold for directed inspections required to prevent catastrophic fatigue failure.

7. DAMAGE-TOLERANCE EVALUATION.

a. General. The damage tolerance requirements of §25.571(b) are intended to ensure that should fatigue cracking due to fatigue, corrosion, or accidental damage occur within the LOV, the structure will be capable of withstanding the loading conditions specified in §25.571 (b)(1) through (b)(6) without failure or detrimental structural deformation until the cracking is detected. The evaluation should include identifying the PSE's, defining the loading conditions and conducting sufficiently representative structural tests or analyses, or both, to provide sufficient data for the establishment of the inspection program. Although this process applies to either single or multiple load path structure, the use of multiple load path structures should be given high priority in achieving a damage tolerant design. The principle analytical tool used for metallic materials to perform a Damage Tolerance Evaluation is based on Linear Elastic Fracture Mechanics. A discussion of this approach is presented in Appendix 1 of this Advisory Material.

b. *Damage Tolerant Characteristics.* A damage-tolerant structure has two notable attributes:

(1) The structure can tolerate a significant amount of damage, due to fatigue, environmental, or accidental deterioration, without compromising the continued airworthiness of the airplane (residual strength and rigidity).
(2) The structure can sustain that damage long enough to be found and repaired during scheduled or unscheduled maintenance (inspectability).

c. *Design Considerations*. To achieve a damage-tolerant structure, criteria should be established to guide the design process so that this design objective is achieved. The design process should include a damage-tolerance evaluation (test and analysis) to demonstrate that the damage-tolerant design objectives are achieved and to identify inspections or other procedures necessary to prevent catastrophic failure. Reliance on special inspections should be minimized by designing structure with easily detectable (e.g. visual) cracking modes. Since the occurrence of widespread fatigue damage can complicate a damage-tolerant evaluation to the point that reliable inspections programs cannot be developed even with extremely intensive inspection methods, it must be demonstrated, with sufficient full scale fatigue test evidence, that adequate maintenance procedures are contained in the ALS of the ICA, such that WFD will not occur within the LOV. A discussion of several issues an applicant might face in demonstrating freedom from WFD is contained in Appendix 2 of this Advisory Material.

d. *Design Features.* Design features which should be considered in attaining a damage-tolerant structure include the following:

(1) Multiple load path construction and/or the use of damage containment features to arrest fast fracture or reduce the crack growth rate, and to provide adequate residual strength;
(2) Materials and stress levels that provide a slow rate of crack propagation combined with high residual strength;
(3) Arrangement of design details to ensure a sufficiently high probability that a failure in any critical structural element will be detected before the strength has been reduced below the level necessary to withstand the loading conditions specified in § 25.571(b).

e. *Probabilistic Evaluations.* Normally, damage-tolerance assessments consist of a deterministic evaluation of design features described in paragraphs 6d(1), (2), and (3). Paragraphs f. through k. below provide guidelines for this approach. In certain specific instances, however, damage-tolerant design might be more realistically assessed by a probabilistic evaluation if the statistical data employed in the analysis are based on tests or operational experience, or both, of similar structure.

f. *PSE's and Detail Design Points*. In accordance with §25.571(a), a damage tolerance and fatigue evaluation should be conducted for each part of the structure which could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mountings, landing gears, and their related primary attachments).

In identifying PSE's, consideration should be given to the effect caused by partial or complete loss or failure of structure with respect to continued safe flight and landing, considering all flight phases including stability and control and aeroelasticity.

In accordance with §25.571(a)(1)(ii) this evaluation must include the identification of PSE's and the detail design points within a PSE, the failure of which could contribute to catastrophic failure of the airplane. A detail design point is one that may warrant specific actions such as special inspections or other procedures to ensure continued airworthiness. These are areas that are at higher risk for fatigue cracking than other areas.

(1) *PSE and Detail Design Point Selection criteria*

(i) *PSE's*. The selection criteria for PSE's should include the following considerations:

(a) Elements in tension or shear;
(b) Low static margin;
(c) High stress concentrations;
(d) High load transfer;
(e) High stresses in secondary members after primary member failure;
(f) Materials with high crack growth rates;
(g) Areas prone to accidental damage;
(h) Component test results;
(i) Results of full-scale fatigue test.

(ii) *Detail Design Points.* Detail design points, within a PSE, can be determined by analysis or by fatigue tests on complete structures or sub-components. However, tests may be necessary when the basis for analytical prediction is not reliable, such as for complex components. If less than the complete structure is tested, care should be taken to ensure that the internal loads and boundary conditions are valid.

The selection criteria for detail design points should include the following considerations:

(a) Any evidence of cracking encountered in service on comparable structure;

(b) Any evidence of cracking found during fatigue testing on comparable structure;
(c) Available strain gage data;
(d) Locations where permanent deformation occurred on static test articles;
(e) Areas analytically shown to have relatively low crack initiation life;
(f) Susceptibility to corrosion or other environmental deterioration (e.g. disbonding);
(g) Potential for manufacturing anomalies (e.g. new or novel manufacturing processes where the potential for damage may not be well understood);
(h) Vulnerability to in-service induced accidental damage.
(i) Areas, whose failure, would create high stresses in the remaining structure.

(2) *Examples of Principal Structural Elements.* Typical examples of structure which may be Principal Structural Elements are:

(i) *Wing and empennage*
(a) Control surfaces, slats, flaps, and their mechanical systems and attachments (hinges, tracks, and fittings);
(b) Primary fittings;
(c) Principal splices;
(d) Skin or reinforcement around cutouts or discontinuities;
(e) Skin-stringer combinations;
(f) Spar caps;
(g) Spar webs; and
(h) Ribs and bulkheads.

(ii) *Fuselage*
(a) Circumferential frames and adjacent skin;
(b) Pilot window posts;
(c) Pressure bulkheads;
(d) Skin and any single frame or stiffener element around a cut-out;
(e) Skin or skin splices, or both, under circumferential loads;
(f) Skin or skin splices, or both, under fore and aft loads;
(g) Skin and stiffener combinations under fore and aft loads;
(h) Door skins, frames, and latches; and
(i) Window frames.

(iii) *Landing gear and their attachments*

(iv) *Engine mounts and struts*

(v) *Thrust reversers components, whose failure could result in inadvertent deployment*

g. Inaccessible areas. Every reasonable effort should be made to ensure inspectability of all structural parts, and to qualify them under the damage tolerance provisions (reference 14 CFR § 25.611).

h. Residual strength testing of principal structural elements. Analytical prediction of the residual strength of structures can be very complex due to nonlinear behavior, load redistribution and the potential for a multiplicity of failure modes. The nature and extent of residual strength tests will depend on previous experience with similar structures. Simulated cracks should be as representative as possible of actual fatigue damage. Where it is not practical to produce actual fatigue cracks, damage can be simulated by cuts made with a fine saw, sharp blade, guillotine, or other suitable means. Whatever artificial means are used to simulate sharp fatigue cracks, sufficient evidence should be available from element tests to indicate equivalent residual strength. If equivalency cannot be shown every attempt should be made to apply enough cyclic loading to generate fatigue cracks from the artificial damage prior to applying residual strength loads. Special consideration should be given to the procedure for pre-cracking so that subsequent test results are representative. This can be an issue when slow stable tearing in ductile sheet or plate material is part of the failure mechanism. Inappropriate pre-cracking loads can lead to unconservative results. In those cases where bolt failure, or its equivalent, is to be simulated as part of a possible damage configuration in joints or fittings, bolts can be removed to provide that part of the simulation.

i Damage tolerance analysis and tests.

(1) It should be determined by analysis, supported by test evidence, that:

(i) The structure, with the extent of damage established for residual strength evaluation, can withstand the specified residual strength loads (considered as ultimate loads); and

(ii) The crack growth life under the repeated loads expected in service (between the time the damage becomes initially detectable and the time the extent of damage reaches the value for residual strength evaluation) provides a practical basis for development of the inspection program and procedures described in paragraph 8 of this AC.

(2) The repeated loads should be as defined in the loading, temperature, and humidity spectra. The loading conditions should take into account the effects of structural flexibility and rate of loading where they are significant.

(3) The damage tolerance characteristics can be shown analytically by reliable or conservative methods such as the following:

(i) By demonstrating quantitative relationships with structure already verified as damage tolerant; or

(ii) By demonstrating that the repeated loads and residual strength load stresses do not exceed those of previously verified designs of similar configuration, materials, and inspectability.

8.0 INSPECTION REQUIREMENTS.

a. Damage Detection. Detection of damage before it becomes critical is the most important issue in ensuring the damage tolerance characteristics of the structure. For this reason, Amendment 25-54 revised § 25.571 to require that the applicant establish inspections or other procedures, as necessary, to prevent catastrophic failure from accidental, environmental, or fatigue damage, and include those inspections and procedures in the Airworthiness Limitations Section of the "Instructions for Continued Airworthiness" required by 14 CFR § 25.1529 (see also App. H to part 25).

Due to the inherent, complexities that affect the damage tolerance evaluation, such as operating practices, environmental effects, load sequence effects on crack growth, and variations in inspection methods, operational experience should be taken into account in establishing inspection thresholds, repeat intervals and inspection procedures. Additionally, careful consideration should be given to how practical the inspection procedures are.

b. Environmental and Accidental Damage Inspection Programs. While the inspections developed under § 25.571(b) are for the detection of cracks due to fatigue, accidental damage, and corrosion, a separate program needs to be implemented for the detection of environmental and accidental damage. This is generally accomplished through the Maintenance Review Board activity for a new model large transport airplane. Although such inspections, or other procedures, as necessary, to prevent catastrophic failure of an airplane, are required to be included in the Airworthiness Limitations Section of the "Instructions for Continued Airworthiness," subsequent individual operator experience may indicate that different inspections or other procedures are justified for that individual operator. Sections 43.16 and 91.403(c) provide a means for FAA approval of alternatives to the airworthiness limitations. In reviewing such proposed alternatives, the FAA will evaluate them using the methods described in this AC, or other acceptable methods proposed by the TC holder and operators, to ensure that the objectives of § 25.571 will continue to be met.

c. Inspection Threshold for Fatigue Cracking. The inspection threshold is the point in time at which the first planned structural inspection is performed following entry into service. The threshold may be as low as the repeat interval, or may allow for a significant period of operation before inspections are started.

The concept of delaying an inspection threshold beyond the repeat interval is based on the premise that it will take a certain amount of time before fatigue cracks would develop to a size that would be detectable during a structural inspection. Consequently, it may be acceptable to wait some period of time before starting to inspect for fatigue cracks. Nevertheless, the inspections should begin early enough to ensure that there is a high confidence of detecting those cracks before they could lead to a catastrophic structural failure, even in cases where the structure is of a lower bound manufacturing quality.

The process for establishing a threshold for inspection is illustrated in Figure 1. A damage tolerance and a SDC evaluation of the structure is required.

(1) Structure for Which No Special Inspection is Required. The process of normal maintenance is carried out from the entry of the aircraft into service. There is no associated threshold for this type of inspection. Establishing any special inspections for this kind of structure would be optional and for economic reasons only (*e.g.* to detect cracking at an earlier stage, or to avoid cracking across a fleet, and therefore reduce the subsequent repair task). The method used to establish a threshold for these inspections would also be optional. In order to establish a condition where no special inspections are required, the results of the damage tolerance evaluations and SDC are needed to demonstrate that normal maintenance inspections (including those for ED and AD) are adequate to prevent a catastrophic failure. An example of where no special inspection is required would be when:

(i) the critical crack length is greater than the crack length, which is readily detectable during a General Visual Inspection (GVI), without internal access or removal of fairings, and

(ii) the time from detectable to critical is greater than the GVI interval.

(2) Structure for Which Special Inspection is Required. A threshold must be established where special inspections are required. The threshold may be as low as the repeat interval, or as noted above, may allow for a significant period of operation before inspections are started, based on fatigue life and assuming lower bound manufacturing quality. How long the first inspection can be delayed depends on whether or not the SDC of the structure can be detected during normal maintenance.

(i) Single Load Path Structure. For single load path structure the threshold should not be greater than 50% of the DSG unless a rational analysis including the following is provided:

a. Confidence in loads (loads survey, load enhancement factor, etc.)
b. Susceptibility to damage (from ED/AD and/or maintenance error)
c. Confidence in material data (use of proven material)
d. Initial flaw size (lower bound quality flaw, minimum of 99/95)
e. Critical crack length verified by test evidence (component or full scale)
f. Crack growth verified by test evidence (component or full scale)
g. Slow crack propagation unfactored (a_{det} to a_{crit} = minimum of 2* DSG and $a_{initial}$ to a_{crit} = minimum of 3* DSG)
h. Sampling inspection program starting no greater than 50% DSG.

(ii) Other than Single Load Path Structure. For fatigue-critical areas of the structure, the inspection threshold is established through an assessment of the overall fatigue initiation and crack growth life (*i.e.* until the damage reaches a critical condition such that the remaining undamaged structure fails under the application of the residual strength loads given in §25.571(b)(1) to (6)). In determining the threshold, it should be assumed that the structure contains damage equivalent to a 'lower bound' manufacturing quality (due to possible errors in processing, fabrication, assembly or handling) at the critical detail design point. The maximum probable size of this damage depends on the component being assessed, the manufacturing process, and the capability of any quality control inspections to detect the defect at the time of manufacture. A discussion of the various approaches that have been used to determine fatigue life based thresholds is contained in Appendix 3.

(a) SDC Detectable During Normal Maintenance. When a structure has SDC that is detectable during normal maintenance, the threshold should not be greater than the lesser of:
(A) the DSG, or
(B) the calculated threshold, accounting for lower bound manufacturing quality, for a 95% probability of survival with 95% confidence.

(b) SDC Not Detectable During Normal Maintenance. In some cases, the SDC of a structure in a critical area may not be detectable during normal maintenance. An example would be a crack in the inner skin at a longitudinal fuselage lap splice running continuously along a rivet line. Although the SDC might be as large as a frame bay or more it would not be detectable without NDI from the outside or special access to the interior. This would not be considered normal maintenance. Detection of the fatigue crack would be totally dependent on the special inspection. Here the threshold should not be greater than the lesser of:

(A) 75%DSG, or

(B) the calculated threshold, accounting for lower bound manufacturing quality for a 99% probability of survival with a 95% confidence.

NOTE: If the area being considered is susceptible to accidental damage (*e.g.* cargo door corner), and the fatigue inspection is also the primary inspection for accidental damage, the threshold should be set equal to the repeat interval since this type of damage can occur at any time.

** Threshold for SLP may be > 50% DSG if rational analysis of §8.c(2)(i) is provided.

Figure 1 Establishment of Inspection Threshold

d. Inspection Intervals. The basis for setting inspection intervals is the period of time during which damage is detectable and the residual strength remains above required levels. The reliability of the repeat inspection program (i.e. frequency of inspections and probability of detection) should assure damage detection before the residual strength of the aircraft is compromised. Inspection intervals may be established by applying appropriate reduction factors to this period to ensure that the crack or failed load path will be found before the residual strength of the structure drops below the required level.

Detectable crack sizes and shapes assumed to determine inspection intervals should be consistent with inspection method capabilities and the cracking characteristics of the structure being evaluated. If concurrent cracking in adjacent areas or surrounding structure is expected within the operational life of the airplane, then this should be accounted for in the cracking scenario assumed.

A discussion of how repeat inspection intervals are determined for the more common situations in aircraft structures is contained in Appendix 4.

9.0 FATIGUE EVALUATION.

a. General. The evaluation of structure under the following fatigue (safe-life) strength evaluation methods is intended to ensure that catastrophic fatigue failure, as a result of the repeated loads of variable magnitude expected in service, will be avoided throughout the structure's operational life. Under these methods, the fatigue life of the structure should be determined. The evaluation should include the following:

(1) Estimating or measuring the expected loading spectra for the structure;
(2) Conducting a structural analysis, including consideration of the stress concentration effects;
(3) Performing fatigue testing of structure that cannot be related to a test background to establish response to the typical loading spectrum expected in service;
(4) Determining reliable replacement times by interpreting the loading history, variable load analyses, fatigue test data, service experience, and fatigue analyses;
(5) Evaluating the possibility of fatigue initiation from sources such as corrosion, stress corrosion, disbonding, accidental damage and manufacturing defects based on a review of the design, quality control and past service experience; and
(6) Providing necessary maintenance programs and replacements times to the operators. The maintenance program should be included in the Instructions for Continued Airworthiness in accordance with Section 25.1529.

b. Scatter Factor for Safe-life Determination. In the interpretation of fatigue analyses and test data, the effect of variability should, under § 25.571(c), be accounted for by an appropriate scatter factor. In this process it is appropriate that the applicant justify the scatter factor chosen for any safe-life part. The following guidance is provided (see Figure 2):

(1) The base scatter factors applicable to test results are: BSF_1= 3.0, and BSF_2 = (see paragraph 9b(5) of this AC). If the applicant can meet the requirements of paragraph 9b(3) of this AC, he may use BSF_1 or, at his option, BSF_2.

(2) The base scatter factor, BSF_1, is associated with test results of one representative test specimen.

(3) Justification for use of BSF_1. BSF_1 may only be used if the following criteria are met:

(i) Understanding of load paths and failure modes. Service and test experience of similar in-service components that were designed using similar design criteria and methods should demonstrate that the load paths and potential failure modes of the components are well understood.

(ii) Control of design, material, and manufacturing process quality. The applicant should demonstrate that his quality system (e.g., design, process control, and material standards) ensures the scatter in fatigue properties is controlled, and that the design of the fatigue critical areas of the part account for the material scatter.

(iii) Representativeness of the test specimen.

(A) The test article should be full scale (component or sub-component) and represent that portion of the production aircraft requiring test. All differences between the test article and production article should be accounted for either by analysis supported by test evidence or by testing itself.

(B) Construction details, such as bracket attachments, clips, etc., should be accounted for, even though the items themselves may be non-load bearing.

(C) Points of load application and reaction should accurately reflect those of the aircraft, ensure correct behavior of the test article, and guard against uncharacteristic failures.

(D) Systems used to protect the structure against environmental degradation can have a negative effect on fatigue life and therefore should be included as part of the test article.

(4) Adjustments to base scatter factor BSF_1. Having satisfied the criteria of paragraph 9b(3), justifying the use of BSF_1, the base value of 3.0 should be adjusted to account for the following considerations, as necessary, where not wholly taken into account by design analysis. As a result of the adjustments, the final scatter factor may be less than, equal to, or greater than 3.0.

(i) Material fatigue scatter. Material properties should be investigated up to a 99% probability of survival and a 95% level of confidence.

(ii) Spectrum severity. Test load spectrum should be derived based on a spectrum sensitive analysis accounting for variations in both utilization (i.e. aircraft weight, cg etc.) and occurrences/size of loads. The test loads spectrum applied to the structure should be demonstrated to be conservative when compared to the usage expected in service.

(iii) Number of representative test specimens. Well established statistical methods should be used that associate the number of items tested with the distribution chosen, to obtain an adjustment to the base scatter factor.

(5) If the applicant cannot satisfy the intent of all of paragraph 9b(3) of this AC, BSF_2 should be used.

(i) The applicant should propose scatter factor BSF_2 based on careful consideration of the following issues: the required level of safety, the number of representative test specimens, how representative the test is, expected fatigue scatter, type of repeated load test, the accuracy of the test loads spectrum, spectrum severity, and the expected service environmental conditions.

(ii) In no case should the value of BSF_2 be less than 3.0.

(6) Resolution of test loadings to actual loadings. The applicant may use a number of different approaches to reduce both the number of load cycles and number of test set-ups required. These include, but are not limited to, spectrum blocking (e.g., a change in the spectrum load sequence to reduce the total number of test setups); high load clipping (e.g., the reduction of the highest spectrum loads to a level such that the beneficial effects of compression yield are reduced or eliminated); and low load truncation (e.g., the removal of non-damaging load cycles to simplify the spectrum). Due to the modifications to the flight-by-flight loading sequence caused by these changes, the applicant should propose either analytical or empirical approaches to quantify an adjustment to the number of test cycles which represents the difference between the test spectrum and assumed flight-by-flight spectrum. In addition, an adjustment to the number of test cycles may be justified by raising or lowering the test load levels, as long as appropriate data supports the applicant's position. Other effects to be considered are different failure locations, different response to fretting conditions, temperature effects, etc. The analytical approach should use well established methods or be supported by test evidence.

c. Replacement times. Replacement times should be established for parts with established safe-lives and should, under § 25.571(a)(3), be included in the "Instructions for Continued Airworthiness" required under § 25.1529. These replacement times can be extended if additional data indicate an extension is warranted. Important factors that should be considered for such extensions include, but are not limited to, the following:

(1) Comparison of original evaluation with service experience.

(2) Recorded load and stress data. Recorded load and stress data entails instrumenting airplanes in service to obtain a representative sampling of actual loads and stresses experienced. The data to be measured include airspeed, altitude, and load factor versus time data; or airspeed, altitude, and strain ranges versus time data; or similar data. The data, obtained by instrumenting airplanes in service, provide a basis for correlating the estimated loading spectrum with the actual service experience.

(3) Additional analyses and tests. If test data and analyses based on repeated load tests of additional specimens are obtained, a re-evaluation of the established safe-life can be made.

(4) Tests of parts removed from service. Repeated load tests of replaced parts can be utilized to reevaluate the established safe-life. The tests should closely simulate service loading conditions. Repeated load testing of parts removed from service is especially useful where recorded load data obtained in service are available, since the actual loading experienced by the part prior to replacement is known.

(5) Repair or rework of the structure. In some cases, repair or rework of the structure can gain further life.

d. Type design developments and changes. For design developments or design changes involving structural configurations similar to those of a design already shown to comply with the applicable provisions of § 25.571(c), it might be possible to evaluate the variations in critical portions of the structure on a comparative basis. Typical examples would be redesign of the wing structure for increased loads, and the introduction in pressurized cabins of cutouts having different locations or different shapes, or both. This evaluation should involve analysis of the predicted stresses of the redesigned primary structure and correlation of the analysis with the analytical and test results used in showing compliance of the original design with § 25.571(c).

e. Environmental effects such as temperature and humidity should be considered in the fatigue analysis and should be demonstrated through suitable testing.

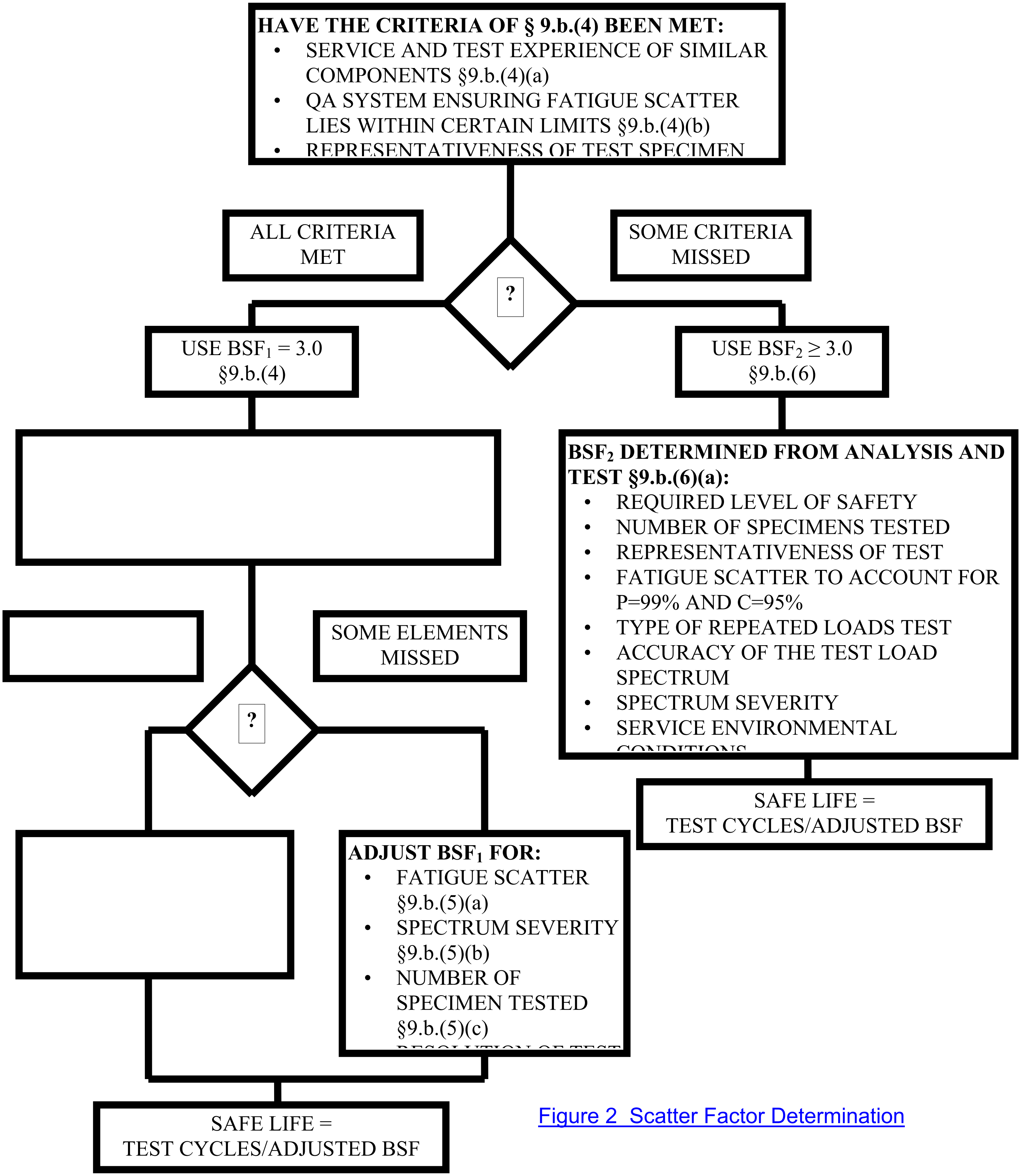

Figure 2 Scatter Factor Determination

10.0 DISCRETE SOURCE DAMAGE.

a. General. The purpose of this section is to establish FAA guidelines for the consistent selection of load conditions for residual strength substantiation in showing compliance with § 25.571 (e), Damage-tolerance (discrete source) evaluation. The intent of these guidelines is to define, with a satisfactory level of confidence, load conditions that will not be exceeded on the flight during which the specified incident of § 25.571(e) occurs. In defining these load conditions, consideration has been given to the expected damage to the airplane, the anticipated response of the pilot at the time of the incident, and the actions of the pilot to avoid severe load environments for the remainder of the flight consistent with his knowledge that the airplane may be in a damaged state.

b. The maximum extent of immediately obvious damage from discrete sources (§ 25.571 (e)) should be determined and the remaining structure shown, with an acceptable level of confidence, to have static strength for the maximum load (considered as ultimate load) expected during completion of the flight. In lieu of a rational analysis, for uncontained rotor failure, likely structural damage may be assumed to be equivalent to that obtained by using the rotor burst model and associated trajectories defined in AC 20-128A/ACJ 25.903(d)(1) paragraph 9.0 "Engine and APU Failure Model". This assessment should also include an evaluation of the controllability of the aircraft in the event of damage to the flight control system.

c. The loads considered as ultimate should not be less than those developed from the following:

(1) At the time of the occurrence:

(i) The maximum normal operating differential pressure, multiplied by a 1.1 factor, combined with 1.0 g flight loads including the external aerodynamic pressures.

(ii) Starting from 1.0g level flight at speeds up to Vc, any maneuver or any other flight path deviation caused by the specified incident of § 25.571 (e), taking into account any likely damage to the flight controls and pilot normal corrective action.

(2) For the continuation of the flight: The maximum appropriate cabin differential pressure (including the external aerodynamic pressure), combined with:

(i) Seventy percent (70%) of the limit flight maneuver loads as specified in 25.571(b) and, separately;

(ii) At the maximum operational speed, taking into account any appropriate reconfiguration and flight limitations, the 1.0g loads plus incremental loads arising from application of forty percent (40%) of the limit gust velocity and turbulence intensities as specified in 25.341 at Vc.

d. At any time, the airplane must be shown by analysis to be free from flutter up to VD/MD with any change in structural stiffness resulting from the incident.

11.0 LIMIT OF VALIDITY

The Limit of Validity is established as an upper limit to airplane operation with the inspections and other procedures provided under §25.1529, and Appendix H. The limit is established because of increased uncertainties in the probable development of widespread fatigue damage associated with airplane operation past the limit. In order to operate the airplane past the initial LOV, the maintenance program will need to be revalidated, through a widespread fatigue damage audit (AC91-56B Appendix 2), to establish additional maintenance actions to prevent the occurrence of WFD in the fleet.

The establishment of the LOV is closely associated with fatigue test and teardown results. (Reference Appendix 2) It is acceptable for the fatigue test and teardown information to not be available at time of certification as long as a plan to produce it is approved prior to certification. The initial LOV, at time of certification, is equivalent to one-half of the total number of flight cycles or hours, as appropriate, on the fatigue test. For certification, the LOV must be specified in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness. As the fatigue test progresses, the LOV may be periodically raised to no more than one half of the total number of flights on the fatigue test article but should not exceed one half of the DSG until such time as the fatigue test program and the validation of the final LOV is complete. Based on the analysis of the test and teardown information (see AC91-56B) the applicant will propose the final LOV. The applicant is expected to provide rationale for the proposed LOV.

12.0 STRUCTURAL DAMAGE CAPABILITY

a. Structural Damage Capability (SDC) is the attribute of the structure which permits it to retain its required residual strength in the presence of large damage. SDC is a characteristic of the design of a structure, and is therefore not associated with the inspectability of that structure. Furthermore, SDC should exist regardless of the type and source of the damage. Consequently, the damage scenario considered in an SDC assessment may not be representative of that anticipated in service. Nevertheless, the residual strength analysis required to demonstrate SDC complements that performed to establish a structural maintenance program, according to paragraph (b) of §25.571, and the analysis methods should be based on similar existing test evidence.
An SDC assessment should be performed for each Principal Structural Element (PSE) considered under paragraph (b) of §25.571.

b. The evaluation of a given PSE for SDC is intended to ensure that, in the event of a large damage that results in partial failure of the structure, the remaining intact structure is capable of carrying the required limit loads. The extent of SDC to be demonstrated should be consistent with the type of PSE under evaluation. To this end, the structure is further classified under the following categories:

(1) Category A. Exclusively single load path structures, such as fittings, single lugs, etc.

(2) Category B. Structure with significant crack arrest or crack retardation features, *i.e.* monolithic structures which incorporate 'damage containment features', such as the integral crack stoppers in machined wing spars and cast doors, that are intended to restrict possible damage to a size whereby the remaining uncracked structure can sustain limit load. Some other examples of this classification are: Skin Cutouts & Discontinuities, Window & Door Frames, Window Posts, Control Surfaces

(3) Category C. Multiple load path structures, including lugs, fittings, door stops, Stiffened Panels, Skin Joints, Wingbox Ribs, Door Latches, Control Surface Attachments, Engine Mounts, Thrust Reversers

NOTE: A stiffened panel structure, as used in conventional fuselage, wing and empennage construction, is considered to be a multiple load path component. In addition to the longitudinal stiffening members and transverse frames attached to the skin, each individual skin bay between stiffeners is assumed to be a separate load path.

c. In general, a structure meeting the SDC criteria must be able to withstand the required residual strength load in the presence of damage equivalent to either the complete failure of any individual load path, or partial failure of a load path between damage containment features. For the different structural categories introduced previously, the required SDC is as follows:

(1) Category 'A' Structure. For single load path structure:

i. Slow crack growth must be demonstrated, i.e. the time (unfactored) for the growth of a crack from detectable to critical is at least one DSG.

ii. A quality control/quality assurance plan that ensures the parts are controlled during design and manufacture so that the risk of failure in service is minimized must be provided to the administrator. See Appendix 5.

(2) Category ‘B’ Structure. The required SDC is defined by the maximum extent of damage that may develop between the damage containment features. In the case of a machined wing spar, this would be equivalent to a crack extending from a failed spar flange, through the spar web, to the integral crack stopper. For areas of major skin cutouts (passenger and emergency exit doors, cargo doors, undercarriage bays, wing box access panels, *etc.*), the SDC should be equivalent to a skin crack that extends from the edge of the cutout to the adjacent stiffening member (stringer, frame, spar, *etc.*) or crack stopper.

It is implicit in this requirement that the damage containment features actually control the rate of crack growth, and provide adequate residual strength. It should therefore be shown that crack growth is arrested, or significantly retarded, by the crack stopper, as compared to the case where the crack stopper is omitted.

(3) Category ‘C’ Structure. It should be ensured that the multiple load path design includes sufficient structural redundancy to allow for the failure of one complete load path *e.g.* multiple hinges and multiple doorstops. As a minimum SDC requirement for stiffened panels, it should be demonstrated that the structure can sustain residual strength load following the failure of any individual load path, but in the absence of other damage, *i.e.*

i. the complete failure of any one stiffening member (stringer, frame, *etc.*) without any additional damage in the skin or adjacent stiffeners, and

ii. the failure of a single skin bay (*i.e.* a crack between two adjacent stiffeners) without any additional damage in the stiffeners or adjacent skin bays.

d. The methodologies used to show that the structure meets the SDC requirement should be based on analysis supported by test experience. The nature and extent of tests required on complete structures, or on sub-assemblies, will depend upon applicable previous design, construction and test experience.

For category ‘B’ structure, in which the SDC consists of the partial failure of the structure between damage containment features, the simulated damage should be represented as a fatigue crack with active crack tips. In this case, the analysis methods should be those used in a conventional fracture mechanics calculation, as undertaken during existing damage tolerance assessments.

For category ‘C’ structure the following applies:

(i) If the failed load path for SDC is a discrete element, the analysis will be limited to the static assessment of the ability of the remaining intact load paths to carry residual strength load. No fracture mechanics calculations are required.

(ii) If the failed load path for SDC is part of a continuous element, such as a skin element which extends over several stiffening elements or features of an integrally stiffened panel the simulated damage should be represented as a fatigue crack with active tips.

Appendix 1 – Crack Growth Analysis and Tests

Crack Growth Analyses and Tests. Crack growth characteristics should be determined for each detail design point identified in accordance with 7(f) above. This information, when combined with the results from the residual strength analyses and tests, will be the basis for establishing the inspection requirements as discussed in section 8. Crack growth characteristics can be determined by analysis or test. However, due to the large number of detail design points that typically are evaluated and practical limitations involved with testing, analyses are generally relied on to determine crack growth at the detail design point.

(1) *Analyses*. In order to perform a crack growth analysis a number of key elements are needed. These include (1) a load/stress spectrum applicable to the detail design point, (2) an initial crack size and shape to be assumed, (3) a cracking scenario to be followed, (4) an applicable stress intensity solution(s), (5) a crack growth algorithm and (6) material crack growth rate properties.

A loading spectrum must be developed for each detail design point. It is derived from the overall aircraft usage spectrum that is discussed in section 6.b. The spectra at each detail design point may be modified for various reasons. The most common modification for metallic structure involves the deletion of high infrequent loads that may have an unrepresentative beneficial effect on crack growth if retardation is considered. Also, local load events that are not part of the overall aircraft spectrum should be included (e.g. flutter damper loads during pre-flight control surface checks).

The initial crack size and shape and subsequent cracking scenario to be followed are problem dependent. Guidance on this is given in Appendix 4.

Applicable stress intensity solutions may be available in the public domain or may need to be developed. Many references exist which provide technical guidance for application and development of stress intensity solutions. Care should be taken to ensure that the reference stress used for the spectrum load and stress intensity solution are compatible.

Crack growth algorithms used in predicting crack extension range from simple linear models to complex ones that can account for crack growth retardation and acceleration. It is generally accepted that the use of a linear model will result in conservative results. A non-linear model on the other hand can be conservative or unconservative and generally requires a higher level of validation and analysis/test correlation to adequately validate the accuracy of the algorithm. Coupon testing should be performed using representative materials and spectra types (e.g. wing lower cover, pylon support lug, horizontal stabilizer upper cover) that will be encountered in the course of the overall aircraft crack growth evaluation.

Crack growth rate data (e.g. da/dN vs. ΔK vs. R, da/dN vs. ΔK_{eff}) for many common aerospace materials is available in the public domain. Additionally, testing standards (e.g. ASTM) exist for performing tests to gather this data. Generally accepted practice is to use typical or average representation of this data for performing crack growth evaluations.

(2) *Tests*. Crack growth testing using coupons is typically performed to generate crack growth rate data and validate crack growth algorithms used for analyses. Simple specimens are generally used that have well established stress intensity solutions for the characteristic cracking that can be expected. The primary issue for these tests is the precracking required to achieve a well-behaved fatigue crack before data is collected. Effective precracking procedures (e.g. "load shedding") have been established and are described in the public domain. Care must be taken to insure that subsequent crack growth is not affected by the prior precracking.

In order to minimize the test time for actual structural components and/or full scale test articles, the test loading spectrum may be modified by eliminating small magnitude load events or by replacing them with a fewer number of larger load events that give equivalent crack growth.

Crack growth behavior may be obtained from actual structural components and/or full-scale test articles. However, inducing active fatigue cracks of the desired initial size and at the desired locations can be extremely difficult. Past success in obtaining useful data has been achieved on an opportunistic basis when natural fatigue cracks have developed in the course of normal cyclic testing. Naturally occurring and artificially induced fatigue cracks may be monitored and data collected for at least a portion of the overall crack growth period to be used for setting inspection requirements. This data can be extremely useful in supplementing and validating the analytical predictions, in some cases it may be the sole basis for establishment of inspection requirements. Where fatigue test crack growth data is used, the results should be corrected to address expected operational environmental conditions.

Appendix 2 – Freedom From Widespread Fatigue Damage

Freedom from Widespread Fatigue Damage. FAR 25.571(b) requires the effectiveness of the provisions to preclude the possibility of widespread fatigue damage occurring within the limits of validity of the maintenance program to be demonstrated with sufficient full-scale fatigue test evidence. The determination of what constitutes "sufficient full-scale test evidence" requires a considerable amount of engineering judgment and is a matter that should be discussed and agreed to between an applicant and the FAA early in the planning stage for a certification project. In general, sufficient full-scale test evidence consists of fatigue testing to two or more times the DSG, followed by specific inspections and analyses to determine that widespread fatigue damage has not occurred. The following factors should be considered in determining the sufficiency of the evidence:

Factor 1: The comparability of the load spectrum between the test and the projected usage of the airplane.

Factor 2: The comparability of the airframe materials, design and build standards between the test article and the certificated airplane.

Factor 3: The extent of post-test teardown inspection and analysis for determining if widespread fatigue cracking has occurred.

Factor 4: The duration of the fatigue testing.

Factor 5: The size and complexity of a design or build standard change. This factor applies to design changes made to a model that has already been certificated and for which full-scale fatigue test evidence for the original structure should have already been determined to be sufficient. Small, simple design changes, comparable to the original structure, could be analytically determined to be equivalent to the original structure in their propensity for WFD. In such cases, additional full-scale fatigue test evidence should not be necessary.

Factor 6: The age of an airplane being modified. This factor applies to airplanes that have already accumulated a portion of their design service goal prior to being modified. An applicant should only be required to demonstrate freedom from WFD up to the LOV for the original airplane.

The following examples offer some guidance on the types of data sets that might constitute "sufficient evidence" for some kinds of certification projects.

(1) *New type certificates:* Normally this type of project would necessitate its own full-scale fatigue test to represent the new structure and it's loading environment. Nevertheless, prior full-scale fatigue test evidence from earlier tests performed by the applicant, or others, may also

be used and could supplement, or in rare instances eliminate the need for, additional tests on the new model. Ultimately, the evidence needs to be sufficient to conclude with confidence that, within the design service goal of the airframe, widespread fatigue damage will not occur. Factors 1 through 4 should be considered in determining the sufficiency of the evidence.

A test duration of a minimum of twice the design service goal for the airplane model would normally be necessary if the loading spectrum is realistic, the design and construction for the test article principal structure is the same as for the certificated airplane, and the post-test teardown is exhaustive. If the conformance to Factors 1 through 3 is less than ideal, a significantly longer test duration would be needed to conclude with confidence that WFD will not occur within the design service goal. Moreover, no amount of fatigue testing will suffice if the conformance to Factors 1 through 3 above is not reasonable.

(2) *Derivative models:* It may be possible to reliably extrapolate the occurrence of widespread fatigue damage for part or all of the derivative model from the data that the applicant generated or assembled during the original certification project. Nevertheless, the evidence needs to be sufficient to allow confidence in the calculations that show that widespread fatigue damage will not occur within the design service goal of the airplane. Factors 1 through 5 should be considered in determining the sufficiency of the evidence for derivative models. For example, a change in structural design concept, a change in aerodynamic contour, or a modification of structure that has a complex internal load distribution might well make analytical extrapolation from the existing full-scale fatigue test evidence very uncertain. Such changes might well necessitate full-scale fatigue testing of the actual derivative principal structure. On the other hand, a typical derivative often involves extending the fuselage by inserting “fuselage plugs” that consist of a copy of the typical semi-monocoque construction for that model with slightly modified material gauges. Normally this type of project would not necessitate its own full-scale fatigue test.

(3) *Type design changes – service bulletins:* Normally this type of project would not necessitate its own full-scale fatigue test because the applicant would have generated, or assembled, sufficient full-scale fatigue test evidence during the original certification project that could be applied to the change. Nevertheless, as cited in the previous example, the evidence needs to be sufficient to allow confidence in the calculations that show that widespread fatigue damage will not occur within the design service goal of the airplane. In addition, Factor 5, ‘The size and complexity of a design change’, should be considered.

(4) *Supplemental type certificates (STC):*

(i) Sufficient full-scale test evidence for structure certified under an STC may necessitate additional full-scale fatigue testing, although the extent of the design change may be small enough to use Factor 5, to establish the sufficiency of the existing full-scale fatigue test evidence. In addition, although the applicant for an STC may not have access to the original equipment manufacturer's full-scale fatigue test data, the applicant may assume the basic structure was shown to comply with the regulation, unless the FAA/JAA has taken, or intends to take, Airworthiness Directive (AD) action to alleviate a WFD condition. This assumption implies that sufficient full-scale fatigue test evidence exists, demonstrating that WFD will not occur within the design service goal of the airplane. For the purpose of the STC applicant's demonstration, it may be assumed that model types certified under JAR/FAR 25.571, as amended by Amendment 25-96, and which are not subject to AD action to alleviate a WFD condition, have received two full design service goals of fatigue testing, under realistic loads, and have received a thorough post-test inspection that did not detect any widespread fatigue damage. With this assumption, and Factors 1 through 5, the STC applicant may be able to demonstrate that WFD will not occur on its modification (or the underlying original structure) within the design service goal. If, however, the modification significantly affects the distribution of stress in the underlying structure, or significantly alters loads in other parts of the airplane, or significantly alters the intended mission for the airplane, or if the modification is significantly different in structural concept from the certificated airplane being modified, additional representative fatigue test evidence would likely be necessary.

(ii) In addition, Factor 6, 'The age of the airplane being modified', comes into play for modifications made to older airplanes. The STC applicant should demonstrate freedom from WFD up to the design service goal of the airplane being modified. For example, an applicant for an STC to an airplane that has reached an age equivalent to 75 percent of its design service goal should demonstrate that the modified airplane will be free from WFD for at least the remaining 25 percent of the design service goal. Although an applicant could attempt to demonstrate freedom from WFD for a longer period, this may not be possible unless the original equipment manufacturer cooperates by providing data for the basic structure. A short design service goal for the modification could simplify the demonstration of freedom from WFD for the STC applicant. Nevertheless, the applicant should also be aware that the design service goal of the airplane is not a fixed life; it may be extended as a result of a structural re-evaluation and service action plan, such as has been developed for certain models under the FAA's 'Aging Airplane Program'. Unless the modifier also reevaluates its STC modification, the shorter design service goal for the modification could impede extending the design service goal of the modified airplanes. Because of these unique considerations for STCs, it is especially important that the applicant address this issue with the FAA at an early stage of the STC project.

(5) *Repairs*. New repairs that differ from repairs contained in the FAA-approved original equipment manufacturer's structural repair manual, but that are comparable in design to such repairs, and that meet the JAR/FAR in other respects, would not necessitate full-scale fatigue testing.

Appendix 3 – Methods for Threshold Determination

Different approaches have been used to calculate thresholds, although these are essentially variants of one of two methods, *viz.*

(a) the fatigue (stress-life or strain-life) method, which uses fatigue endurance data collected under constant stress or constant strain conditions, and a linear damage accumulation model (Palmgren-Miner rule)

(b) the crack growth method, which uses crack propagation and residual strength data to calculate the growth from an assumed initial crack size to a critical crack length, according to fracture mechanics principles.

There is no evidence to suggest that any of these approaches is inherently more accurate than others in predicting the life of a structure subjected to representative aircraft spectrum loading, as long as the calculation is supported by appropriate test evidence and service experience. In each case, the production of a reliable estimate of the fatigue life requires that the analysis is correctly 'calibrated' by reference to component and full-scale fatigue tests, in order to account for effects such as residual stresses (in the stress-life and strain-life methods) or crack growth retardation (in the crack growth method).

In lieu of other data, a conservative estimate of a lower bound threshold for inspection is obtained for aluminium airframe structure, if an initial corner crack of radius 0.05" (1.27mm) is assumed at a single typical fastener hole and the total crack growth life is divided by 2.

Regardless of the approach used, the calculated threshold should be substantiated with appropriate fatigue test evidence. The best source of fatigue test evidence is from service experience, augmented by the results of large component full-scale fatigue tests. Large component and full-scale fatigue test specimens are generally constructed using the same manufacturing processes as on the actual aircraft. It may then be assumed that the results of such tests provide sufficient information to reliably establish the lower bound manufacturing quality, especially when those results are combined with service experience. Conversely, the simple test specimens used to generate fatigue endurance and crack growth data, which are typically assembled under laboratory or workshop conditions, may not be representative of the actual range of manufacturing quality in the structure under consideration. Therefore, in the absence of information from full-scale fatigue tests or service experience, consideration should be given to generating fatigue endurance and crack growth data on simple test specimens which include artificial damages that are introduced at the beginning of the test, and are representative of the lower bound manufacturing quality.

Appendix 4 – Methodology for Determining Repeat Inspection Intervals

The applicant may face a number of different configurations that he will be required to evaluate and determine repeat inspection intervals. The reliability of the repeat inspection program (i.e. frequency of inspections and probability of detection) should assure damage detection before the residual strength of the aircraft is compromised. Several of the more common situations are described below.

a. *Single load path structure*. The basis for setting an inspection interval for a single load path structure is illustrated in Figure 1. The inspection interval is based on the time for a detectable crack (a_{DET}) to grow to critical crack size (a_{CRIT}). The inspection interval is based on this time (L).

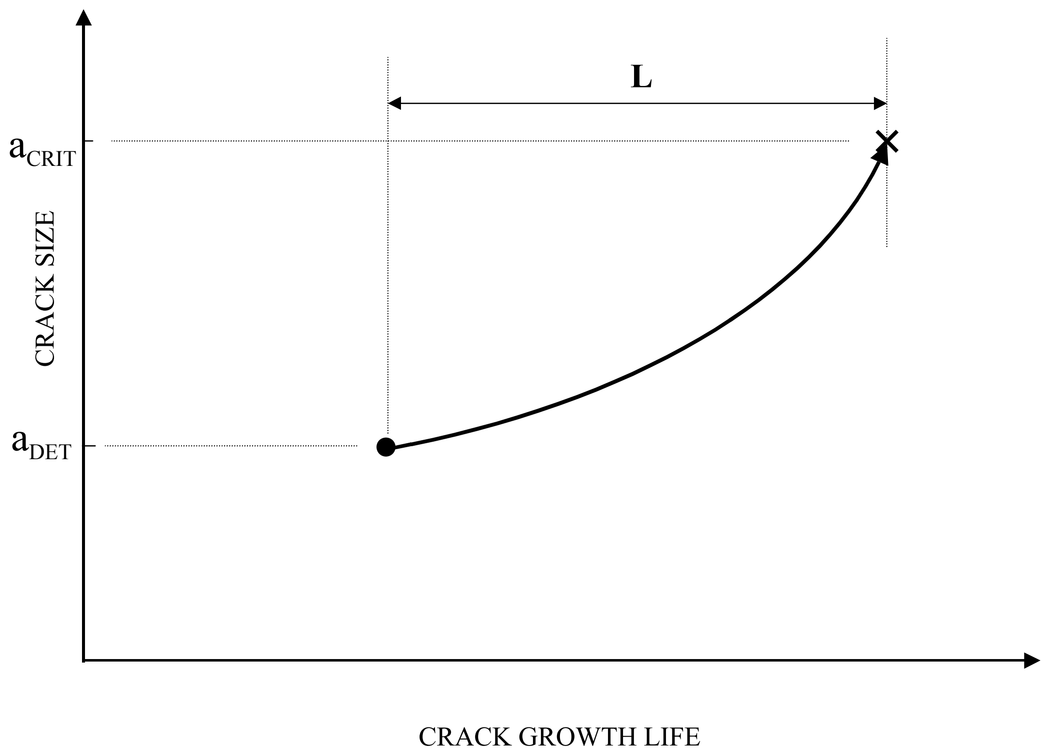

Figure 1 Single Load Path Structure Basis for Inspection Interval

b. *Multiple load path structure*. The safety of 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 consistent with section a above. On the other hand, depending on inspectablity considerations and residual life characteristics of the structure, subsequent to a load path failure, it may be beneficial to take advantage of its redundancy. The inspection for multiple load path structure may be for a completely failed load path or for less than a load path failure, in either case the residual life of the secondary load path(s) subsequent to damage being detected in the primary load path(s), is used to determine the inspection interval.

The inspection interval can be determined by test or analysis for complete or partial load path failure as described below.

(1) *Complete Load Path Failure Evaluation by Test* Figure 2 illustrates some key points if an inspection for complete load path failure is to be developed based on testing only. The inspection interval is based on the test demonstrated residual life (L_r) subsequent to load path failure. Since the residual life decreases with the time accumulated prior to load path failure L_r will be dependent on the time at which load path failure is simulated, (N_D).

The test article should consist of as manufactured production parts. Representative loading for the intact structure should be applied for some predetermined period of time, (N_D). At the end of this period the load path failure should be simulated (e.g. saw cutting, attachment(s) removal, member removal). The test should then be restarted with representative damage condition loading. (Note that the external loads may be the same as for the intact structure if the damage simulation results in the correct "failed" condition internal load redistribution.) The test should continue until the desired residual life has been achieved, or to critical crack size, whichever is less, (N_0).

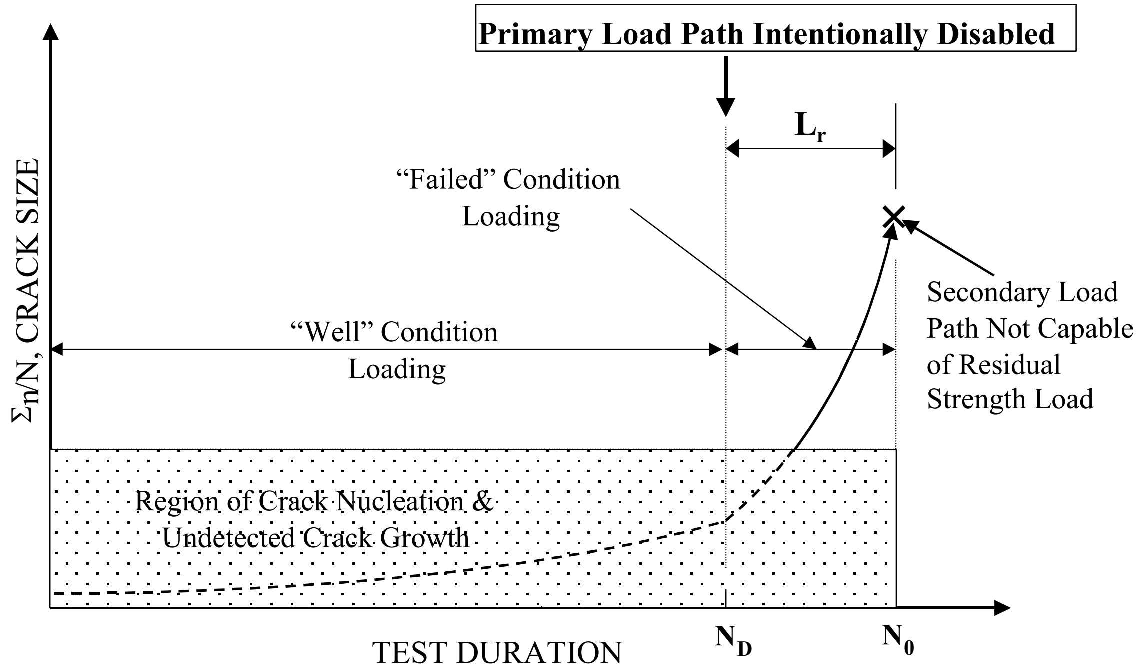

Figure 2. Multiple Load Path Structure Evaluation by Test to Support Inspection for Failed Load Path

(2) Evaluation by analysis. Figure 3 illustrates how inspection intervals could be established on the basis of crack growth and residual strength evaluation. The figure is for less than load path failure scenario but can also be used to discuss the case of a complete load path failure.

The inspection interval is based on the life of the secondary load path(s) (L_r) subsequent to primary load path(s) failure at N_F plus the time (L_P) for a detectable crack (a_{DET}) in the primary load path(s) to grow to critical size under in-service loads. 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 normal manufacturing quality. Damage accumulated prior to load path failure is accounted for by calculating the amount of growth, (Δa_i), that occurs between time zero and N_F using "well" condition loading. 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).

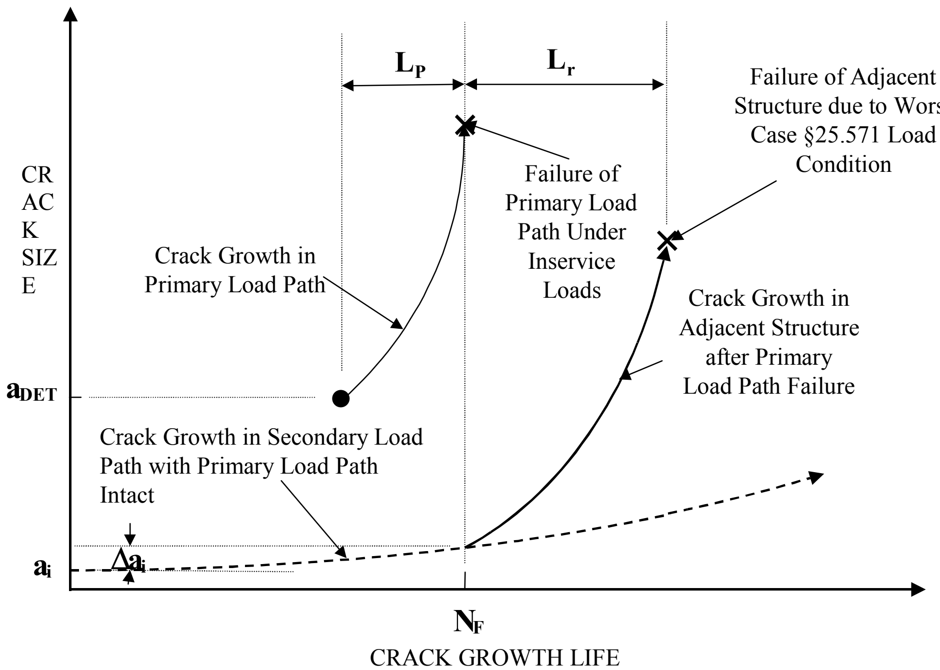

Figure 3 Inspection Interval Determination for Multiple Load Path Structure Based on Crack Growth and Residual Strength Evaluations

Appendix 5 – Quality Assurance and Control for Single Load Path Critical Parts

1. *Definition -* Critical parts are those parts whose failure during ground or flight operation could cause catastrophic failure of the structure.

2. *Objective -* The objective of a critical parts plan is to ensure that critical parts are controlled during design, manufacture, and throughout their service life so that the risk of failure in service is minimized by ensuring that the critical parts maintain the characteristics on which certification is based.

3. *Procedures* - The applicant should establish a critical parts plan. The policies and procedures which constitute that plan should be such as to ensure:

    a) That all critical parts of the structure are identified by means of a failure analysis, and a critical parts list established.
    b) That documentation draws the attention of the personnel involved in the design, manufacture, and maintenance of a critical part to the special nature of the part. For example, all drawings, work sheets, inspection documents, etc. should be prominently annotated with the words “Critical Part” or equivalent and the Instructions for Continued Airworthiness should clearly identify critical parts, and detail the relevant special instructions.
    c) The details of the manufacturing procedures and processes for critical parts (for example manufacture and source, forging procedures, machining operations and sequence, and inspection techniques and acceptance and rejection criteria) are defined. The parts (including test articles) on which certification is based should be produced in accordance with the above manufacturing procedures. Procedures for changing these manufacturing procedures should also be established.
    d) That any changes to the design or manufacturing procedures of a critical part, or to its operating environment or loading spectrum, are considered to establish their effects on the fatigue and damage tolerance evaluation of the part. This evaluation should involve further fatigue testing, unless it can be shown that testing is not necessary.
    e) That materials review procedures for critical parts (i.e. procedures for determining the disposition of parts having manufacturing errors or material flaws) are in accordance with paragraphs (c) and (d) above.
    f) That critical parts are marked (by serial number) and records relating to the marking are maintained, such that it is possible to establish the relevant manufacturing modification and service history of the individual parts (or batches in the case of parts too small to be individually marked or without a limited service life).

g) That all critical parts produced in whole or in part under sub-contracting or partnership arrangements are subject to the critical parts plan at all stages of production.

14 - How does the proposed standard compare to the current ICAO standard? The ICAO standard is much more general.

The proposed standard complies with the general requirement of ICAO.

15 - Does the proposed standard affect other HWG's?

Other HWG's are not affected.

16 - What is the cost impact of complying with the proposed standard?

The proposed rulemaking would not impose any additional costs on manufacturers or operators of part 25 airplanes.

The new requirement to include a limit of validity for the structural maintenance program required by (a)(3) in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness does not represent a significant cost to the airplane manufacturers. The sub-paragraph (b) full scale fatigue test evidence requirement of both the proposed rule and the existing rule are essentially the same, since the design service goal of the airplane and the limit of validity of the maintenance program are generally the same in duration.

The new requirement of paragraph (f), for Structural for Damage Capability, will generally not result in additional costs for airplane manufacturers since they typically design most of their structure to tolerate a single element failure (fail-safe design). In the case of a single load path designs, the additional requirements of (f)(2) do not impose a significant additional cost to the airplane manufacturers over what they would spend even if there were no regulatory requirement for such critical components.

The new requirement of paragraph (g) will not result in additional costs for airplane manufacturers, since this requirement only discusses the fact that the inspectability of an article of structure must be addressed when setting the thresholds for damage-tolerance inspections.

Other changes made to FAR 25.571, for the purpose of harmonizing it with JAR 25.571, did not result in any substantive change in requirements over what is already required by FAR part 25. Harmonization of the FAR and JAR regulations is a benefit to industry and the FAA because it simplifies the certification evaluation of new type designs and eliminates duplicate documentation for manufacturers. Therefore, there is no increased cost to the airplane manufacturers.

17. - If advisory or interpretive material is to be submitted, document the advisory or interpretive guidelines. If disagreement exists, document the disagreement.

See Item 13 above

18. - Does the HWG wish to answer any supplementary questions specific to this project?

Yes

19. – Does the HWG want to review the draft NPRM at “Phase 4” prior to publication in the Federal Register?

Yes

20. – In light of the information provided in this report, does the HWG consider that the “Fast Track” process is appropriate for this rulemaking project, or is the project too complex or controversial for the Fast Track Process? Explain.

Yes

FAA Action: Placed on the AVS “Do By Other Means” list, dated June 14, 2005.