Federal Aviation Administration – Regulations and Policies
Aviation Rulemaking Advisory Committee

Rotorcraft Issue Area
Composite Rotorcraft Structure

Task 1 - Revise 14 CFR/JAR 27 and 29

[Federal Register: April 5, 2000 (Volume 65, Number 66)]

[Notices]

[Page **17936**-17938]

From the Federal Register Online via GPO Access [wais.access.gpo.gov]

[DOCID:fr05ap00-111]

DEPARTMENT OF TRANSPORTATION

Federal Aviation Administration

Aviation Rulemaking Advisory Committee Rotorcraft Issues--New Task

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

ACTION: Notice of new task assignments for the Aviation Rulemaking Advisory Committee (ARAC)

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

FOR FURTHER INFORMATION CONTACT: Mark Shilling, Rotorcraft Standards Staff (ASW--119), Federal Aviation Administration, 2601 Meacham Blvd,

Fort Worth, Texas 76137-4298; phone (817) 222-5110; fax (817) 222-5961 email Mark.R.Schilling@**faa**.gov.

[[Page 17937]]

SUPPLEMENTARY INFORMATION:

Background

The **FAA** has established an Aviation Rulemaking Advisory Committee to provide advice and recommendations to the **FAA** Administrator, through the Associate Administrator for Regulation and Certification, on the full range of the **FAA**'s rulemaking activities with respect to aviation-related issues. This includes obtaining advice and recommendations on the **FAA**'s commitment to harmonize the Federal Aviation Regulations (FAR) and practices with its trading partners in Europe and Canada.

The Task

This notice is to inform the public that the **FAA** has asked ARAC to provide advice and recommendations on the following harmonization tasks:

Task No. 1: Damage Tolerance and Fatigue Evaluation of Metallic Rotorcraft Structure

The project is to be a harmonized Joint Aviation

Regulation (JAR)/FAR 27/29 ARAC program.

Evaluate: the European Association of Aerospace Industries and the Aerospace Industry Association's White Paper, the recommendations contained in the Technical Oversight Group for Aging Aircraft letters to the **FAA**, and the ongoing activities and results of rotorcraft damage tolerance research and development.

Identify the information needed to commence rulemaking and define an acceptable means of compliance.

Recommend appropriate changes to FAR/JAR 29 regarding damage tolerance and fatigue evaluation of metallic structure, and recommend appropriate changes to FAR/JAR 27 that would allow damage tolerance as an option. Any recommended changes should be practical and appropriate to the unique characteristics of rotorcraft. Where feasible and appropriate, provide consistency with FAR/JAR 23/25.

Evaluate and revise, as appropriate, the following advisory materials: AC 29-2; AC 27-1; and AC 20-95, Fatigue Evaluation of Rotorcraft Structure; and related guidance.

The recommendation should be forwarded to the Federal Aviation Administration (**FAA**) and the Joint Aviation Authorities (JAA) in the format of a proposed rule.

Although this tasking for metallic structure does not depend on the completion of the composite structure project, the Composite Rotorcraft Structure and Metallic Rotorcraft Structure working groups should communicate to avoid possibly conflicting recommendation to amend the same regulatory sections.

The **FAA** requests that ARAC draft appropriate regulatory documents with supporting economic and other required analyses, and any other related guidance material or collateral documents to support its recommendations. If the recommendation results in one or more notice of

proposed rulemaking (NPRM) published by the **FAA**, the **FAA** may ask ARAC to dispose of any substantive comments the **FAA** receives.

A progress report should be provided at each Joint Harmonization Working Group meeting. The recommendation should be forwarded to the **FAA** and the **FAA** by September 2002.

Task No. 2: Damage Tolerance and Fatigue Evaluation of Composite Rotorcraft Structure

The project is to be a harmonized FAR/JAR 29/29 ARAC program.

Revise current FAR/JAR 27 and 29 to add regulations for composite structure. Consider creating a new FAR/FAR 27/29.573 to address composite structure.

Evaluate and revise, as appropriate, the regulations and the following advisory materials: AC 20-107A, Composite Aircraft Structure; AC 27-1; AC 29-2; and related guidance to achieve the goal of improved tolerance to flaws and defects in composite structure with methodology and procedures which are practical and appropriate to rotorcraft. Where feasible and appropriate, provide consistency with FAR/JAR 23/25.

The recommendation should be forwarded to the **FAA** and JAA in the format of a proposed rule.

Although this tasking for composite structure does not depend on the completion of the metallic structure project, the Composite Rotorcraft Structure and Metallic Rotorcraft Structure working groups should communicate to avoid possibly conflicting recommendations to amend the same regulatory sections.

The **FAA** requests that ARAC draft appropriate regulatory documents with supporting economic and other required analyses, and any other related guidance material or collateral documents to support its recommendations. If the recommendation results in one or more NPRM's published by the **FAA**, the **FAA** may ask ARAC to dispose of any substantive comments the **FAA** receives.

A progress report should be provided at each Joint Harmonization Working Group meeting. The recommendation should be forwarded to the **FAA** and JAA by November 2002.

ARAC Acceptance of Task

ARAC has accepted the tasks and has chosen to establish two new working groups, the Composite Rotorcraft Structure working group and the Metallic Rotorcraft Structure working group. The working groups will serve as staff to ARAC to assist ARAC in the completion of the assigned tasks. Working group recommendations must be reviewed and approved by ARAC. If ARAC accepted the working groups' recommendations, ARAC will forward them to the **FAA** as recommendations.

Working Group Activity

The Composite Rotorcraft Structure working group and the Metallic Rotorcraft Structure working group is expected to comply with the procedures adopted by ARAC. As part of the procedures, the working groups are expected to:

1. Recommend a work plan for completion of the task, including the rationale supporting such a plan, for consideration at the Rotorcraft Issues ARAC meeting held following publication of this notice.

2. Given a detailed conceptual presentation of the proposed recommendations prior to proceeding with the work stated in item 3

below.

3. Draft appropriate regulatory documents with supporting economic and other required analyses, and/or any other related guidance material or collateral documents the working group determines to be appropriate; or, if new or revised requirements or compliance methods are not recommended, a draft report stating the rationale for not making such recommendations.

4. Provide a status report at each meeting of ARAC held to consider rotorcraft issues.

Participation in the Working Group

The Composite Rotorcraft Structure working group and the Metallic Rotorcraft Structure working group will be composed of technical experts having an interest in the assigned tasks. A working group member need not be a representative of a member of the full committee.

An individual who has expertise in the subject matter and wishes to become a member of the working group should write to the person listed under the caption FOR FURTHER INFORMATION CONTACT expressing that desire, describing his or her interest in the task, and stating the expertise he or she would bring to the working group. All requests to participate must be received no later than April 10, 2000. The requests will be reviewed by the

[[Page 17938]]

assistant chair and the assistant executive director, and the individuals will be advised whether or not the request can be

accommodated.

Individuals chosen for membership on the working group will be expected to represent their aviation community segment and participate actively in the working group (e.g., attend all meetings, provide written comments when requested to do so, etc.). They also will be expected to devote the resources necessary to ensure the working group meets any assigned deadline(s). Members are expected to keep their management chain advised of working group activities and decisions to ensure that the agreed technical solutions do not conflict with their sponsoring organization's position when the subject is presented to ARAC for a vote.

Once the working group has begun deliberations, members will not be added or substituted without the approval of the assistant chair, the assistant executive director, and the working group chair.

The Secretary of Transportation has determined that the formation and use of ARAC are necessary and in the public interest in connection with the performance of duties imposed on the **FAA** by law.

Meetings of ARAC will be open to the public. Meetings of the working groups 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 March 28, 2000.

Anthony F. Fazio,

Executive Director, Aviation Rulemaking Advisory Committee.

[FR Doc. 00-8382 Filed 4-4-00; 8:45 am]

BILLING CODE 4910-13-M

February 24, 2008

Mr. Nicholas A. Sabatini
Associate Administrator for Regulation and Certification
FAA National Headquarters, AVS-1
800 Independence Ave., SW
Washington DC 20591

Dear Mr. Sabatini:

The Aviation Rulemaking Advisory Committee (ARAC) Working Group activity associated with the Damage Tolerance and Fatigue Evaluation of Composite Rotorcraft Structure Task has been completed. The results of their efforts were submitted to ARAC for review and approval. The ARAC examined the proposed rulemaking submittal in a public meeting on March 1, 2007 in Orlando, FL. The associated advisory circular material was not available at that time, and it was reviewed in a public meeting today in Houston, TX. Both packages were approved by ARAC.

Accordingly, the ARAC hereby submits, without change, the proposed NPRM package and associated advisory circular material developed by the Working Group with a recommendation that it be processed for publication. The advisory circular material is formatted for inclusion in AC 27-1, and AC 29-2.

With this submittal the Rotorcraft Issues Group has completed all of the tasks assigned to it. This seems like a good time for me to step aside. The Helicopter Association International, however, is interested in continuing its role in support of ARAC and, in particular, the Executive Committee. HAI recommends as my replacement Mr. David York, the Helicopter Association International's Vice-President for Regulations and International Affairs. Please contact Mr. York at HAI if you need any further information from him. I appreciate all the good support I received over the last 10-plus years from all of your people.

Very Truly Yours,

John D. Swihart, Jr.
ARAC Assistant Chair for Rotorcraft Issues

Attachments

cc:
Mr. Craig Bolt, ARAC Chair
Mr. Norman Joseph, ARAC Vice Chair
Ms. Pam Hamilton, ARAC Executive Director
Mr. Scott Horn, ARAC Asst. Executive Director
Mr. David York, HAI
Mrs. Gerri Robinson, ARM-200
Mr. Nicanor Davidson, ARM-200 w/attachments
Mr. Ted Jones, ASW-111

U.S. Department
of Transportation
**Federal Aviation
Administration**

MAY 19 2008

Mr. David York
Assistant Chair, Rotorcraft
Issues Group
Helicopter Association International
1635 Prince Street
Alexandria, VA 22314

Dear Mr. York:

This is in reply to Mr. Swihart's February 24, 2008 letter. That letter sent the recommendations from the Aviation Rulemaking Advisory Committee's (ARAC) Working Group activity associated with the Damage Tolerance and Fatigue Evaluation of Composite Rotorcraft Structure Task. I understand the Rotorcraft Issues Group approved the Working Group's documents during your meeting on February 24, 2008 without dissent.

I wish to thank the Aviation Rulemaking Advisory Committee, the members associated with Rotorcraft Issues and the Composite Rotorcraft Structure Working Group who provided resources to develop the report and recommendation. The report will be placed on the ARAC website at: http://www.faa.gov/regulations_policies/rulemaking/committees/arac/.

We consider your submittal of the Composite Rotorcraft Structure Working Group's report as completion of the original tasking issued on March 28, 2000 (65 FR 17936, April 5, 2000), which asked for an evaluation and revision of parts 27 and 29, and appropriate advisory materials. We understand that this submission completes all outstanding tasks assigned to the Rotorcraft Issues Group.

Sincerely,

Pamela Hamilton-Powell
Director, Office of Rulemaking

AC 29.573. §29.573 (Amendment 29-XX) DAMAGE TOLERANCE AND FATIGUE EVALUATION OF COMPOSITE ROTORCRAFT STRUCTURE

a. Purpose. This advisory material provides an acceptable means of compliance with the provisions of § 29.573, Amendment XX, of the Federal Aviation Regulations (FAR) dealing with the damage tolerance and fatigue evaluation of transport category composite rotorcraft structure. Paragraph f.(6) specifically addresses the advisory guidance applying to damage tolerance and fatigue evaluation as required by §29.573, Amendment 29-XX. Some information contained in AC 29-2C, MG8 (Amendment 29-42) is repeated and updated as appropriate to preserve the "building block" approach for analyses of composite rotorcraft structure for compliance to §29.573, Amendment 29-XX . (Supplemental guidance can be found in AC 20-107A, "Composite Aircraft Structure.") These procedures address the substantiation requirements for composite material system constituents, composite material systems, and composite structures common to rotorcraft. A uniform approach to composite structural substantiation is desirable, but it is recognized that in a continually developing technical area which has diverse industrial roots, both in aerospace and in other industries, variations and deviations from the procedures described here may be necessary. Significant deviations from this advisory material should be coordinated in advance with the Rotorcraft Directorate.

b. Special Considerations. Since rotorcraft structure is configured uniquely and is inherently subjected to severe cyclic stresses, special consideration is required for the substantiation of all rotorcraft structure, including composites. This special consideration is necessary to ensure that the level of safety intended by the current regulations are attained during the type certification process for all structure with special emphasis on composite structure because of its unique structural characteristics, manufacturing quality and operational considerations, and failure mechanisms.

c. Background.

(1) Historically, rotorcraft have required unique, conservative structural substantiation because of unique configuration effects, unique loading considerations, severe fatigue spectrum effects, and the specialized comprehensive fatigue testing required by these effects. Rotorcraft structural static strength substantiation for both metal and composite structure is essentially identical to that for fixed wing structure once basic loads have been determined. However, rotorcraft structural fatigue substantiation for metals is significantly different from fixed wing fatigue substantiation. Since AC 20-107A, as developed, applies to both fixed wing aircraft and rotorcraft, it, of necessity, was finalized in a broad generic form. Accordingly, a need to supplement AC 20-107A for rotorcraft was recognized during type certification programs. One significant difference in traditional rotorcraft fatigue substantiation programs and fixed wing fatigue programs is the use of multiple component fatigue tests for rotorcraft programs rather than just one full-scale test. Also, constant amplitude, accelerated load

tests are typically used rather than spectrum tests because of the high frequency loads common to rotorcraft operations. These rotorcraft fatigue tests have traditionally involved the generation of stress versus life or cycle (S-N) curves for each critical part (most of which are subjected to the cyclic loading of the main or tail rotor system) using a monotonic (sinusoidal) fatigue spectrum based on maximum and minimum service stress values. Unless configuration differences or flight usage data dictate otherwise, the monotonic fatigue spectrum's period is typically based on six ground-air-ground (GAG) cycles for each flight hour of operation. The S-N curves for the substantiation of each detailed part are typically generated by plotting a curved line through three data points (reference AC 29-2C, AC 29 MG 11, "Fatigue Evaluation of Transport Category Rotorcraft Structure (Including Flaw Tolerance)"). The three data points selected are a short specimen life (low-cycle fatigue), an intermediate specimen life and a long specimen life (high-cycle fatigue). Each raw data point is generated by monotonically fatigue testing at least two full-scale parts to failure or run out for each data point on the S-N curve. The raw data point values are then reduced by an acceptable statistical method to a single value for plotting to ensure proper reliability of the associated S-N curve. Order 8110.9, "Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and Other Power Transmission Systems" and AC 27-1B, AC 27 MG11, "Fatigue Evaluation of Rotorcraft Structure", contain comprehensive discussions of the S-N curve generation process. The rotorcraft S-N curve process contrasts sharply with the fixed wing process of using a single full-scale fatigue article (usually an entire wing or airframe, which constitutes a single full-scale assembly data point), generic material or full-scale assembly S-N data (e.g., Metallic Materials Properties Development and Standardization (MMPDS) formerly the MIL-HDBK-5 for metals, MIL-HDBK-17 for composites, or AC 23-13, "Fatigue, Fail-Safe, and Damage Tolerance Evaluation of Metallic Structure for Normal, Utility, Acrobatic, and Commuter Airplanes", which replaced AFS-120-73-2 for full-scale assemblies), a non-monotonic spectrum and relatively large scatter factors to verify or determine the design fatigue life of the full-scale airplane.

(2) Also, rotorcraft have employed and mass-produced composite designs in primary structure (typically main and tail rotor blades) since the early 1950's. This was 10 or more years before composites were type certificated for primary fixed-wing structure in either military or civil aircraft applications (with some notable limited production exceptions, such as the Windecker fixed wing aircraft). In any case, the early 1950 period was well before a clear, detailed understanding of composite structural behavior (especially in the areas of macroscopic and microscopic failure mechanisms and modes) was relatively common and readily available in a usable format for the average engineer working in this field. It also predated the initial issuance of AC 20-107. Currently, much composite design information is proprietary, either to government, industry or both, and many data gathering methods have not been completely standardized. Consequently, a significant variation from laboratory to laboratory in material property value determination methods and results can exist. The early rotor blade designs (as well as current designs) are by nature relatively low strain, tension structure designs. Also, by nature, these designs are not damage or flaw critical. Thus by circumstance as much as design, early composite rotor blade and other

composite rotorcraft designs incorporated an acceptable fatigue tolerance level of safety. In the 1980's, more test data, analytical knowledge, and analytical methodology became available to more completely substantiate a composite design. Current 14 CFR parts 27 and 29 contain many sections to be considered in substantiating composite rotorcraft structure. This advisory material provides the current or updated information from AC 29-2C, MG 8, Amendment 29-42 to supplement the general guidance of AC 20-107A and provides compliance guidance for the requirements of §29.573 Amendment 29-XX for rotorcraft composite structure.

d. Definitions. The following basic definitions are provided as a convenient reading reference. MIL-HDBK-17, and other sources, contain more complete glossaries of definitions.

(1) AUTOCLAVE. A closed apparatus usually equipped with variable conditions of vacuum, pressure, and temperature. Used for bonding, compressing or curing materials.

(2) ALLOWABLES. Both A-basis and B-basis values statistically derived and used for a particular composite design.

(3) BALANCED LAMINATE. A composite laminate in which all laminae at angles other than 0° occur only in ± pairs (not necessarily adjacent).

(4) A-BASIS ALLOWABLE. The "A" mechanical property value is the value above which at least 99 percent of the population of values is expected to fall, with a confidence level of 95 percent.

(5) B-BASIS ALLOWABLE. The "B" mechanical property value is the value above which at least 90 percent of the population of values is expected to fall, with a confidence level of 95 percent.

(6) BOND. The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent.

(7) COCURE. The process of curing several different materials in a single step. Examples include the curing of various compatible resin system pre-pregs, using the same cure cycle, to produce hybrid composite structure or the curing of compatible composite materials and structural adhesives, using the same cure cycle, to produce sandwich structure or skins with integrally molded fittings.

(8) CURE. To change the properties of a thermosetting resin irreversibly by chemical reaction; i.e., condensation, ring closure, or addition. Cure may be accomplished by addition of curing (crosslinking) agents, with or without a catalyst, and with or without heat.

(9) DELAMINATION. The separation of the layers of material in a laminate.

(10) DISBOND. A lack of proper adhesion in a bonded joint. This may be isolated or may cover a majority of the bond area. It may occur at any time in the cure or subsequent life of the bond area and may arise from a wide variety of causes.

(11) FIBER. A single homogeneous strand of material, essentially one-dimensional in the macro-behavior sense, used as a principal constituent in advanced composites because of its high axial strength and modulus.

(12) FIBER VOLUME. The volume of fiber present in the composite. This is usually expressed as a percentage volume fraction or weight fraction of the composite.

(13) FILL. The 90° yarns in a fabric, also called the woof or weft.

(14) GLASS TRANSITION. The reversible change in an amorphous polymer or in amorphous regions of a partially crystalline polymer from (or to) a viscous or rubbery condition to (or from) a hard and relatively brittle one.

(15) GLASS TRANSITION TEMPERATURE. The approximate midpoint of the temperature range over which the glass transition takes place.

(16) HYBRID. Any mixture of fiber types (e.g., graphite and glass).

(17) IMPREGNATE. An application of resin onto fibers or fabrics by several processes: hot melt, solution coat, or hand lay-up.

(18) LAMINA. A single ply or layer in a laminate in which all fibers have the same fiber orientation.

(19) LAMINATE. A product made by bonding together two or more layers or laminae of material or materials.

(20) LOW STRAIN LEVEL. As used herein, is defined as a principal, elastic axial gross strain level that for a given composite structure provides for no flaw growth and thus provides damage tolerance of the maximum defects allowed during the certification process using the approved design fatigue spectrum.

(21) MATERIAL SYSTEM CONSTITUENT. A single constituent (ingredient) chosen for a material system (e.g., a fiber, a resin).

(22) MATERIAL SYSTEM. The combination of single constituents chosen (e.g., fiber and resin).

(23) MATRIX. The essentially homogeneous material in which the fibers or filaments of a composite are embedded in resins which are mainly thermoset polymers in aircraft structure.

(24) MAXIMUM STRUCTURAL TEMPERATURE. The temperature of a part, panel or structural element due to service parameters such as incident heat fluxes, temperature, and air flow at the time of occurrence of any critical load case, (i.e., each critical load case has an associated maximum structural temperature). This term is synonymous with the term “maximum panel temperature.”

(25) POROSITY. A condition of trapped pockets of air, gas, or void within a solid materials, usually expressed as a percentage of the total nonsolid volume to the total volume (solid + nonsolid) of a unit quantity of material.

(26) PRE-PREG, PREIMPREGNATED. A combination of mat, fabric, nonwoven material, tape, or roving already impregnated with resin, usually partially cured, and ready for manufacturing use in a final product which will involve complete curing. Pre-preg is usually drapable, tacky and can be easily handled.

(27) RESIN. An organic material with indefinite and usually high molecular weight and no sharp melting point.

(28) RESIN CONTENT. The amount of matrix present in a composite either by percent weight or percent volume.

(29) SECONDARY BONDING. The joining together, by the process of adhesive bonding, of two or more already-cured composite parts, during which the only chemical or thermal reaction occurring is the curing of the adhesive itself. The joining together of one already-cured composite part to an uncured composite part, through the curing of the resin of the uncured part, is also considered for the purposes of this advisory circular to be a secondary bonding operation. (See COCURING).

(30) SHELF LIFE. The lengths of time a material, substance, product, or reagent can be stored under specified environmental conditions and continue to meet all applicable specification requirements and remain suitable for its intended function.

(31) STRAIN LEVEL. As used herein, is defined as the principal axial gross strain of a part or component due to the principal load or combinations of loads applied by a critical load case considered in the structural analysis (e.g., tension, bending, bending-tension, etc.). Strain level is generally measured in thousandths of an inch per unit inch of part or microinches/per inch (e.g., .003 in/in equals 3000 microinches/inch).

(32) SYMMETRICAL LAMINATE. A composite laminate in which the ply orientation is symmetrical about the laminate midplane.

(33) TAPE. Hot melt impregnated fibers forming unidirectional pre-preg.

(34) THERMOPLASTIC. A plastic that repeatedly can be softened by heating and hardened by cooling through a temperature range characteristic of the plastic, and when in the softened stage, can be shaped by flow into articles by molding or extrusion.

(35) THERMOSET (OR CHEMSET). A plastic that once set or molded cannot be re-set or remolded because it undergoes a chemical change; (i.e., it is substantially infusible and insoluble after having been cured by heat or other means).

(36) WARP. Yarns extended along the length of the fabric (in the 0° direction) and being crossed by the fill yarns (90° fibers).

(37) WORK LIFE. The period during which a compound, after mixing with a catalyst, solvent, or other compounding constituents, remains suitable for its intended use.

(38) Damage Tolerance. The attribute of the structure that permits it to retain its required residual strength for a period of use after the structure has sustained a given level of fatigue, corrosion, accidental or discrete source damage.

(39) Catastrophic Failure. An event that could prevent continued safe flight and landing.

(40) Damage. A generic term for structural anomalies caused by manufacturing (processing, fabrication, assembly or handling) or service usage. Trimming, fastener installation, or foreign object impact are potential sources of damage, along with fatigue and environmental effects.

(41) Damage Tolerant Safe Life. Capability of structure with damage present to survive expected repeated loads of variable magnitude without detectable damage growth and to maintain ultimate load capability throughout service life of the rotorcraft.

(42) Damage Tolerant Fail-Safe. The capability of structure remaining after a partial failure to withstand design limit loads without catastrophic failure within an inspection period.

(43) Multiple Load Path. Structure providing two or more separate and distinct paths of structure that will carry limit load after complete failure of one of the members.

(44) Active Multiple Load Path. Structure providing two or more load paths that are all loaded during operation to a similar load spectrum.

(45) Passive Multiple Load Path. Structure providing load paths with one or more of the members (or areas of a member) relatively unloaded until failure of the other member or members.

(46) Accidental Damage. Discrete damage which may occur in service use or in manufacturing due to impacts or collisions, such as dents, scratches, gouges, abrasions, disbonds, splintering, and delaminations.

(47) Intrinsic or discrete manufacturing defects. Intrinsic or discrete imperfections or flaws related to manufacturing operations, processing or assembly such as voids, gaps, porosity, inclusions, fiber dislocation, disbonds, and delaminations.

(48) Fatigue or Environmental Damage. Structural damage related to fatigue or environmental effects such as delaminations, disbonds, splintering, or cracking.

(49) Design Limit Loads. The maximum loads to be expected in service, as defined by § 29.301(a).

(50) As-Manufactured. Product or component that has passed the applicable quality control process and has been found to conform to the approved design within the allowable tolerances.

(51) Residual Strength. The strength retained for some period of unrepaired use after a failure or partial failure due to fatigue, accidental, or discrete source of damage.

(52) Principal Structural Element (PSE). A structural element that contributes significantly to the carrying of flight or ground loads and whose failure can lead to catastrophic failure of the rotorcraft.

(53) Coupon. A small test specimen (e.g., usually a flat laminate) for evaluation of basic lamina or laminate properties or properties of generic structural features (e.g., bonded or mechanically fastened joints).

(54) Point Design. An element or detail of a specific design which is not considered generically applicable to other structure for the purpose of substantiation (e.g., lugs and major joints). Such a design element or detail can be qualified by test or by a combination of test and analysis.

(55) Element. A generic element of a more complex structural member (e.g., skin, stringers, shear panels, sandwich panels, joints, or splices).

(56) Detail. A non-generic structural element of a more complex structural member (e.g., specific design configured joints, splices, stringers, stringer runouts, or major access holes).

(57) Subcomponent. A major three-dimensional structure which can provide complete structural representation of a section of the full structure (e.g., stub box, section of a spar, wing panel, wing rib, body panel, or frames).

(58) Component. A major section of the airframe structure (e.g., wing, fin, body, horizontal stabilizer) which can be tested as a complete unit to qualify the structure.

(59) Environment. External, non accidental conditions (excluding mechanical loading), separately or in combination, that can be expected in service and which may affect the structure (e.g., temperature, moisture, UV radiation, and fuel).

e. Related Regulatory and Guidance Material.

Document	Title
FAA Order 8110.9	Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and other Power Transmission Systems
AC 27-1B, MG 11	"Fatigue Evaluation of Rotorcraft Structure"
AC 20-107A	"Composite Aircraft Structure"
AC 21-26	"Quality Control for the Manufacture of Composite Materials"
MIL-HDBK-17	"Composite Material Handbooks"
AC 29-2C, MG 11	"Fatigue Tolerance Evaluation of Transport Category Rotorcraft Metallic Structure"
DOT/FAA/CT-86/39	Whitehead, R.S., Kan, H.P., Cordero, R., and Seather, R., "Certification Testing Methodology for Composite Structures", October 1986.

f. Procedures for Substantiation of Rotorcraft Composite Structure. The composite structures evaluation has been divided into eight basic regulatory areas to provide focus on relevant regulatory requirements. These eight areas are: (1) fabrication requirements; (2) basic constituent, pre-preg and laminate material acceptance requirements and material property determination requirements; (3) protection of structure; (4) lightning protection; (5) static strength evaluation; (6) damage tolerance and fatigue evaluation; (7) dynamic loading and response evaluation; and (8) special repair and continued airworthiness requirements. Original as well as alternate or substitute material system constituents (e.g., fibers, resins, etc.), material systems (combinations of constituents and adhesives), and composite designs (laminates, co-cured assemblies, bonded assemblies, etc.) should be qualified in accordance with the methodology presented in the following paragraphs. Each regulatory area will be addressed in turn. It is important to remember that proper

certification of a composite structure is an incremental, building block process which involves phased FAA/AUTHORITY involvement and incremental approval in each of the various areas outlined herein. It is recommended that a FAA/AUTHORITY certification team approach be used for composite structural substantiation. The team should consist of FAA/AUTHORITY engineering and cognizant aircraft evaluation group inspector(s), the manufacturing inspector(s), the associated Designated Engineering Representatives (DER's), the associated Designated Manufacturing Inspection Representatives (DMIR's), and cognizant members of the applicant's organization. Personnel who are composites specialists (or are otherwise knowledgeable in the subject) should be primary team member candidates.

Once selected, it is recommended that team meetings be held periodically (possibly in conjunction with type boards) during certification to ensure the building block certification process is accomplished as intended. The team should assure that permanent documentation in the form of reports or other Authority-acceptable documents are included in the certification data package. The documentation includes but is not limited to the structural substantiation reports (both analysis and test), manufacturing processes and quality control, and Instructions for Continued Airworthiness (maintenance, overhaul, and repair manuals). The Airworthiness Limitations Section of the Instructions for Continued Airworthiness is approved by FAA/AUTHORITY engineering. Engineering practices for many of the areas identified below are available in Mil-HDBK-17.

(1) The <u>first</u> area is the <u>fabrication requirements</u> of § 29.605:

(i) The quality control system should be developed considering the critical engineering, manufacturing, and quality requirements and a guidance standard such as AC 21-26, "Quality Control for the Manufacture of Composite Materials." This ensures that all special engineering, or manufacturing quality instructions for composites are presented, evaluated, documented, and approved, using drawings, process and manufacturing specifications, standards, or other equivalent means. This should be one of the early phases of a composite structure certification program, since this represents a major building block for sequential substantiation work. Some important concepts of AC 21-26 are included below.

(ii) Specific allowable defect limits on, for example, fiber waviness, warp defects, fill defects, porosity, hole edge effects, edge defects, resin content, large area debonds, and delaminations, etc., for a particular material system component, laminate design, detailed part, or assembly should be jointly established by engineering, manufacturing, and quality and the associated inspection programs for defect detection created, validated, and approved. Each critical engineering design should consider the variability of the manufacturing process to determine the worse case effects (maximum waviness, disbonds, delaminations, and other critical defects) allowed by the reliability limitations of the approved inspection program.

(iii) If bonds or bond lines such as those typical of rotorcraft rotor blade structure are used, special inspection methods, special fabrication methods or other approved verification methods (e.g., engineering proof tests, reference paragraph g(5)) should be provided to detect and limit disbonds or understrength bonds.

(iv) Structurally critical composite construction fabrication process and procurement specifications, for fabricating reproducible and reliable structure, must be provided and FAA/AUTHORITY approved early during the certification process and should, as a minimum, cover the following:

(A) Vendor and Qualified Parts List (QPL) Control. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both the manufacturing inspection district office (MIDO) and FAA/AUTHORITY engineering) at any time, that their quality control systems ensure on a continuous basis, that only qualified suppliers provide the basic material constituents or material systems (e.g., pre-pregs) that meet approved material specifications. Recommended guidelines for qualification of alternate material systems and suppliers are contained in MIL-HDBK-17. These methods can also be used, periodically for qualification status renewals of existing material systems and suppliers.

(B) Receiving Inspection and In-Process Inspection. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their receiving and in-process quality control systems provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, so that the necessary statistical reliability and confidence levels for the items being inspected (that are specified by engineering) are continuously maintained. This will require periodic standard inspections and engineering characterization tests on basic constituent and material system samples which should be conducted, as a minimum, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY witnessed.

(C) Material System Component Storage and Handling. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their composite material system (or constituent) storage and handling procedures and specifications provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, so that the necessary statistical reliability and confidence levels for the items being inspected (which are specified by engineering) are continuously maintained. This should require, as a minimum, periodic inspections to ensure that proper records are kept on critical parameters (e.g., room temperature “bench” exposure, shelf life, etc.) and that periodic basic constituent and material system characterization tests are conducted, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard

should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY witnessed.

(D) Statistical Validation Level. It is necessary to maintain the minimum required statistical validation level of the quality control system (which should be specified for each critical item or constituent by the approved quality and engineering specifications). The statistical validation level should be defined and approved early in certification. Also, approval and proper usage should be continuously maintained during the entire procurement and manufacturing cycles.

(v) Alternate fabrication and process techniques should be approved and must comply with § 29.605. Any alternate techniques should provide at least the same level of quality and safety as the original technique. Any changes should be presented for FAA/AUTHORITY approval well in advance of the change's production effectivity.

(2) The second area is the basic raw constituent, pre-preg, and laminate material acceptance requirements and material property determination requirements of §§ 29.603 and 29.613. These criteria require application of the critical environmental limits such as temperature, humidity, and exposure to aircraft fluids (such as fuel, oils, and hydraulic fluids), to determine their effect on the performance of each composite material system. Temperature and humidity effects are commonly considered by coupon and component tests utilizing preconditioned test specimens for each material system selected. Material "A" and "B" basis allowable strength values and other basic material properties (based on MIL-HDBK-17 or equivalent procedures) are typically determined by small scale tests, such as coupon tests, for use in certification work. In the case of composites, determination of these basic constituent and material system properties will almost invariably involve the submittal, acceptance, and use of company standards. This is currently necessary because the FAA (new managers of MIL-HDBK-17) has not completed development of "B" basis allowables for inclusion in the handbook. Also, test methods vary somewhat from manufacturer to manufacturer; therefore, individual company results will exhibit some scatter in final material property values. Any company standard that is used should meet or exceed related MIL-HDBK-17 requirements. Material structural acceptance criteria and property determination should, as a minimum, include the following:

(i) Property characterization requirements of all material systems (e.g., pre-pregs, adhesives, etc.) and constituents (e.g., fibers, resins, etc.) should be identified, documented, and approved. These requirements, once approved, should be placed in all appropriate procedures and specifications such as those in paragraph f.(1).

(ii) Moisture conditioning of test coupons, parts, subassemblies, or assemblies should be accomplished in accordance with MIL-HDBK-17, other similar approved methods or per FAA/AUTHORITY approved programs.

(iii) The maximum and minimum temperatures expected in service (as derived from test measurements, thermal analyses on panels and other parts, experience, or a combination) should be determined and accounted for in static and fatigue strength (including damage tolerance) substantiation programs considering associated humidity-induced effects.

(iv) The wet glass transition temperature, Tg, is an important characteristic parameter of amorphous polymers, such as epoxies. It is the temperature below which the polymer behaves like a “glassy” solid and above which it behaves like a “rubbery” solid, i.e., it is the temperature at which there is a very rapid change in physical properties. The change from a hard polymeric material to a rubbery material takes place over a narrow temperature range. A composite material will experience a drastic reduction in matrix-controlled mechanical material properties when loaded in this temperature range. Since the resin is the critical structural constituent in a composite matrix and since Tg exceedance is critical to structural integrity; Tg determination is necessary. The Tg margin methodology of MIL-HDBK-17 should be implemented, i.e., the wet glass transition temperature (Tg) should be 50° F higher than the maximum structural temperature (see definition). For any type of resin or adhesive, an acceptable temperature margin using MIL-HDBK-17 techniques (e.g., consideration of limited high temperature excursions) or equivalent methodologies based on tests or experience, or both should be established and approved early in the certification process.

(v) Local design values should be established by analysis and characterization tests and approved for specific structural configurations (point designs) which include the effects of stress risers (e.g., holes, notches, etc.) and structural discontinuities (e.g., joints, splices, etc.). Proper determination of these values for full-scale design and test should be considered one of the most critical building blocks in substantiating and evaluating a composite structure. These transitional load transfer areas typically produce the highest stresses (and strains) and serve as the initiation sites for many of the failures (including those due to the relatively low interlaminar strength of composites) that occur in service in a full-scale part or assembly. Small scales tests (such as coupon, element, and subcomponent tests), or equivalent approved testing programs, and analytical techniques should be carefully designed, prepared, and approved to evaluate potential “hot spots” and provide accurate simulations and representations of full-scale article stresses and strains in the critical transition areas. Proper certification work in this area will ensure initial safety and continued airworthiness in full-scale production articles.

(vi) The design strain level for each major component and material system should be established so that specified impact damage considerations are defined and properly limited. The effects of the strain levels may be established for each composite material using small-scale characterization tests and then the results should be used to establish or verify the maximum allowable design strain level for each full-scale article. The maximum allowable design strain values selected should also take into account the reliability and confidence levels established for the relevant portions of the quality control system. This methodology is necessary because the

amount and size of flaws in the production article may restrict the allowable level of design strain. In a no-flaw-growth design, the maximum specified impact damage and manufacturing flaw size at the most critical location on the part will be a major factor in determining the maximum allowable elastic strain. This design approach is currently selected for nearly all civil and most military applications; since, under normal conditions, only visual inspections are required in the field (unless unusual external damage circumstances such as a hail storm occur) to maintain the initial level of airworthiness (safety). However, many military applications, because of their demanding missions, employ scheduled field non-destructive inspection (NDI) maintenance, (such as comparative ultrasonics) to ensure that flaw growth either does not occur, is controlled by approved structural repair, or by replacement of affected parts. To date, civil applications have not been presented that desire a flaw growth, phased NDI approach. Therefore, selection of the full-scale article's design strain limit based on small-scale tests for a no flaw growth design is extremely important.

(vii) Composite and adhesive properties should be determined so that detrimental structural creep does not occur under the sustained loads and environments expected in service. Small-scale characterization tests (such as coupon, element, and subcomponent tests) and analysis, which verify and establish the full-scale design criteria and parameters necessary to ensure that detrimental structural creep in full-scale structure does not occur in service, should be conducted early in certification and should be FAA/AUTHORITY-approved.

(viii) Material allowable strength values for full-scale design and testing should be developed using the coupon procedures presented in MIL-HDBK-17 or equivalent. The intent is to represent the material variability including the effects that can occur in multiple batches of material and process runs. At least three batches of material samples should be used in material allowable strength testing. Company standards should be prepared, evaluated and FAA/AUTHORITY-approved early in certification (as part of the building block process), that reflect the material property determination considerations recommended in MIL-HDBK-17 on an equal to or better than basis.

(3) The <u>third</u> area is the <u>protection of structure</u> as required by § 29.609. Protection against thermal and humidity effects and other environmental effects (e.g., weathering, abrasion, fretting, hail, ultraviolet radiation, chemical effects, accidental damage, etc.) should be provided, or the structural substantiation should consider the results of those effects for which total protection is impractical. Determination and approval of worst-case or most conservative operating limits, and damage scenarios should be accomplished. Appropriate flammability and fire-resistance requirements should also be considered in selecting and protecting composite structure. Usually a threat analysis is conducted early in the certification process that identifies the various threats and threat levels for which protection must be provided. This data is then used to construct and submit for approval the methods-of-compliance necessary to provide proper structural protection.

(4) The fourth area is the lightning protection requirements of § 29.610. Protection should be provided and substantiated in accordance with analysis and with tests such as those of AC 20-53B, “Protection of Aircraft Fuel Systems Against Fuel Vapor Ignition Caused by Lightning” and FAA Report DOT/FAA/CT-86/8. For composite structure projects involving rotorcraft certificated to earlier certification bases (which do not automatically include the lightning protection requirements of § 29.610), these requirements should be imposed as special conditions. The design should be reviewed early in the certification process to ensure proper protection is present. The substantiation test program should also be established, reviewed and approved early to ensure proper substantiation.

(5) The fifth area is the static strength evaluation requirements of §§ 29.305 and 29.307 for composite structure. Structural static strength substantiation of a composite design should consider all critical load cases and associated failure modes, including effects of environment, material and process variability, and defects or service damage that are not detectable or allowed by the quality control, manufacturing acceptance criteria, or maintenance documents of the end product. The static strength demonstration should include a program of component ultimate load tests, unless experience exists to demonstrate the adequacy of the analysis, supported by subcomponent tests or component tests to accepted lower load levels. The necessary experience to validate an analysis should include previous component ultimate load tests with similar designs, material systems, and load cases.

(i) The effects of repeated loading and environmental exposure, both of which may result in material property degradation, should be addressed in the static strength evaluation. This can be shown by analysis supported by test evidence, by tests at the coupon, element or subcomponent levels, or alternatively by existing data. Earlier discussions in this AC address the effects of environment on material properties (reference paragraphs f. (2)) and protection of structure (reference paragraphs f.(3)). Static strength tests should be conducted for substantiation of new structure. For the critical loading conditions, two approaches to account for prior repeated loading or environmental exposure for structural substantiation exist.

- In the first approach, the large-scale static test should be conducted on structure with prior repeated loading and conditioned to simulate the environmental exposure and then tested in that environment.

- The second approach relies upon coupon, element, and sub-component test data to assess the possible degradation of static strength after application of repeated loading and environmental exposure. The degradation characterized by these tests should then be accounted for in the static strength demonstration test (e.g., load enhancement), or in the analysis of these results (e.g., showing a positive margin of safety with allowables that include the degrading effects of environment and repeated load).

In practice, the two approaches may be combined to get the desired result (e.g., a large-scale static test may be performed at a temperature with a load enhancement factor to account for moisture absorbed over the aircraft structure's life).

(ii) The strength of the composite structure should be statistically established, incrementally, through a program of analysis and tests at the coupon, element, subcomponent, or component levels. As part of the evaluation, building block tests and analyses at the coupon, element, or subcomponent levels can be used to address the issues of variability, environment, structural discontinuity (e.g., joints, cut-outs or other stress risers), damage, manufacturing defects, and design or process-specific details. Figure AC 29.573-1 provides a conceptual schematic of tests included in the building block approach. The material stress-strain curve should be clearly established, at least through the ultimate design load, for each composite design. As shown in Figure AC 29.573-1, the large quantity of tests needed to provide a statistical basis comes from the lowest levels (coupons and elements) and the performance of structural details are validated in a lesser number of sub-component and component tests. The static strength substantiation program should also consider all critical loading conditions for all critical structure including residual strength and stiffness requirements after a predetermined length of service, e.g., end of life (EOL) (which takes into account damage and other degradation due to the service period).

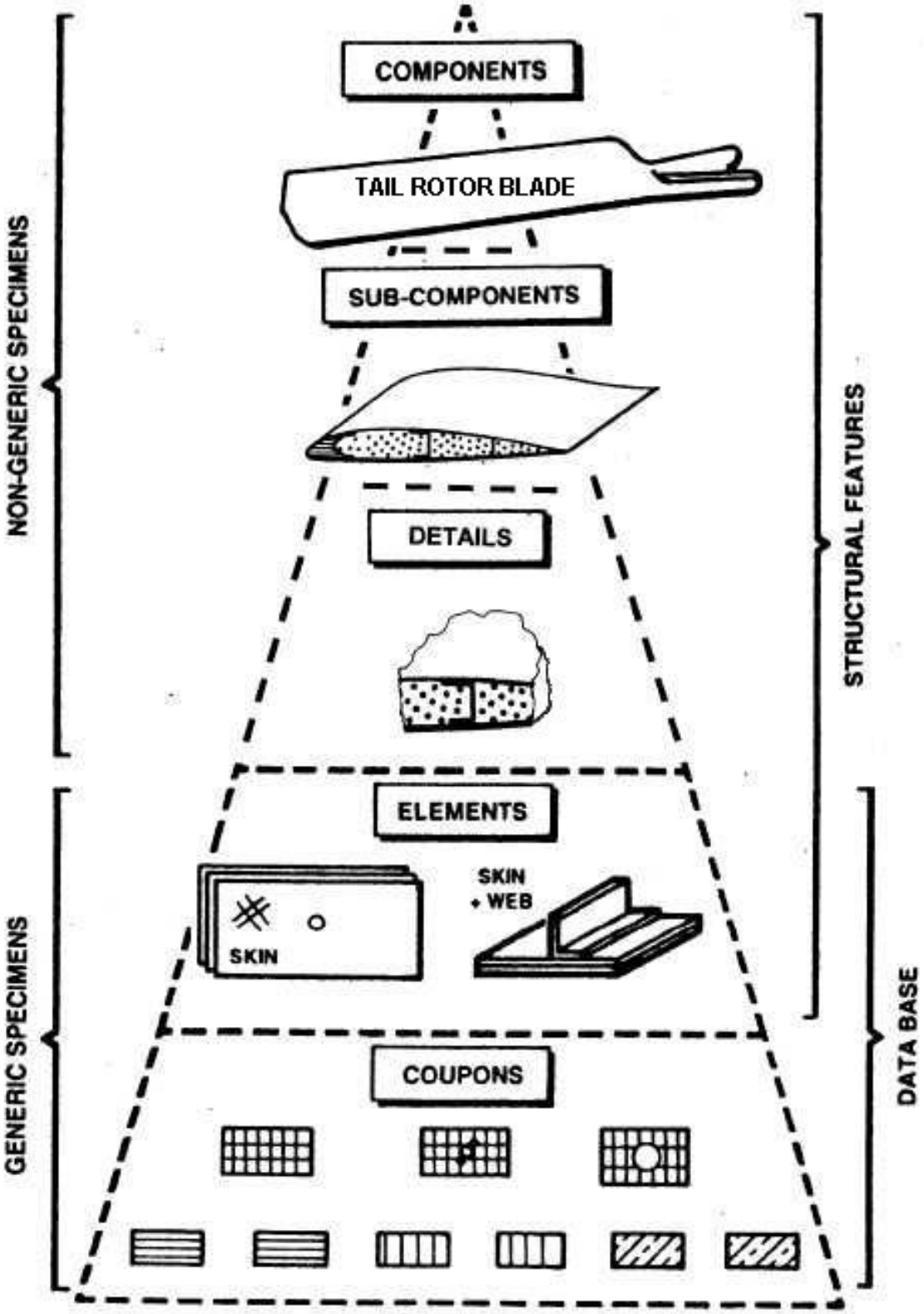

Figure AC 29.573-1. Schematic diagram of building block tests

(iii) Allowables should be evaluated and used as specified in § 29.613. These allowables may be generated at the lamina, laminate, or specific design feature level (e.g. filled hole, lap joint, stringer run-out, etc.), provided they accurately reflect the actual value and variability of the structural strength for the critical failure modes being considered, at each point design where margins need to be established.

(iv) The static test articles should be fabricated and assembled in accordance with production specifications and processes so that they are representative of production structure including defects consistent with the limits established by manufacturing acceptance criteria.

(v) The material and processing variability of the composite structure should be considered in the static strength substantiation. This can be achieved by establishing sufficient process and quality controls to manufacture structure and reliably substantiate the required strength in tests and analyses, which support a building block approach. If sufficient process and quality controls cannot be achieved, it may be necessary to account for greater variability with special factors (§ 29.619) applied to the design. Such factors should be accounted for in the component static tests or analysis.

(vi) It should be shown that impact damage (or other minor discrete source damage) that can be realistically expected from manufacturing and service, but not more than the established threshold of detectability for the selected inspection procedure, will not reduce the structural strength below ultimate load capability. This static strength capability can be shown by analysis supported by test evidence, or by a combination of tests at the coupon, element, subcomponent, and component levels. Later discussions in this AC address the issues associated with damage in excess of that considered in f(5) and drops in residual strength below ultimate load capability (reference paragraph f(6)) below.

(6) The <u>sixth</u> area is the <u>damage tolerance and fatigue evaluation</u> requirements of § 29.573.

(i) BACKGROUND. The static strength determination required by §§ 29.305 and 29.307 establishes the ultimate load capability for composite structures that are manufactured, operated, and maintained with established procedures and conditions. The damage tolerance and fatigue evaluation required by § 29.573 mandates procedures that allow the composite structure to retain the intended ultimate load capability when subjected to expected fatigue loads and conditions during its operational life. The requirements established for the damage tolerance and fatigue evaluation include component replacement times, inspection intervals, or other procedures as necessary to avoid catastrophic failure. These evaluations assume that the baseline ultimate strength capability might be compromised by damage caused by fatigue, environmental effects, intrinsic or discrete flaws, or accidental damage. This damage includes flaws or defects which may occur in manufacturing or maintenance

and which are used to set the ultimate strength capability and establish the manufacturing acceptance criteria. The damage tolerance assessment establishes standards that allow the static strength capability to degrade below the ultimate strength capability assuming such damage occurs within the operational life of the structure. However, when this damage occurs, the remaining structure must withstand expected loads without failure or excessive structural deformations until the damage is detected and the component is either repaired to restore ultimate strength capability or retired.

(ii) GENERAL - The nature and extent of the required analysis or tests on complete structures and portions of the primary structure can be based on applicable previous fatigue or damage tolerant designs, construction, tests, and service experience on similar structures. In the absence of experience with similar designs, Airworthiness Authorities-approved structural development tests of components, subcomponents, and elements should be performed. The following considerations are unique to the use of composite material systems and should be observed for the method of substantiation selected by the applicant.

Rotorcraft structure provides a broad range of composite applications that are quite different in terms of functionality, geometry and inspectability. These include the rotors, the drive shafts, the fuselage, control system components (e.g. push-pull rods), and the control surfaces. When selecting the approach, attention should be given to the composite application under evaluation, the type of potential damage and degradation of the structural design details, the materials used and margin over flight loads. Whatever the approach that may be selected, the following considerations will apply for tests and analysis:

(A) The test articles should be fabricated and assembled in accordance with production specifications and processes so that the test articles are representative of production structure.

(B) The test articles should include material imperfections whose extent is not less than the limits established under the inspection and acceptance criteria used during the manufacturing process and consistent with the inspection techniques used in service (e.g. visual, ultrasonic, X-ray). The initial extent of these imperfections should be discussed and agreed with the FAA/AUTHORITY, taking into account experience in manufacturing and routine in-service inspections. Typical defects to be considered include but are not limited to the following:

(1) Disbonds and weak bonds (considered as disbonds)
(2) Delaminations, fiber waviness, porosity, voids
(3) Scratches, gouges, and penetrations
(4) Impact damage

All of the damages identified in the preceding paragraph (B) above should be derived from the threat assessment described in the following paragraph (C).

(C) Threat Assessment. For each PSE, a threat assessment must be made of the probable locations, types, and sizes of damage

considering fatigue, environmental effects, intrinsic and discrete flaws, and impact or other accidental damage. This determination must be submitted with accompanying rationale to the certifying authority for approval. This rationale may include experience with similar materials, designs, processes (manufacturing, maintenance, and overhaul), structural details, or structure, and may also include service failure evaluations, manufacturing records, overhauls and repair reports, field service reports, incident and accident investigations, service impact surveys, inspectability surveys, and engineering judgment.

Consideration should also be given to factors that:

- Reduce scatter and deviations from nominal structures, such as “frozen processes”, Flight Critical Parts programs, and materials and manufacturing processes to mitigate intrinsic flaws (inclusions and defects).
- Preclude a type of damage by use of a specific design feature (material selection, surface treatment, protective coating, or shielding), a specific stress level (for fatigue damage), or a specific manufacturing inspection process (if it can be shown to be highly reliable, well-controlled and documented, and systematically required).

The assessment should include:

- A systematic evaluation of all the location, types, and sizes of damage and their estimated probability of occurrence.
- A selection or elimination of this damage based on the above estimate
- A verification that the inspection method selected is capable of detecting the damage at the size and location determined.

The types of damage to consider include:

(1) Intrinsic Flaws (imperfections) which are probable to exist in an as-manufactured structure based on the evaluation of the details and potential sensitivities of the specific manufacturing work processes used. The types of flaws to be considered include voids, disbonds, inclusions, foreign objects, resin-rich and resin-starved areas, and improper ply orientation or ply ending. The sizes of the intrinsic flaws considered should be based on the limits established under the manufacturing inspection and acceptance criteria and are expected to remain in service for the life of the structure.

(2) Impact Damage which may occur during manufacturing and in service based on an evaluation of the threats by means of an impact survey and/or service experiences. This type of damage can include dents, penetrations, gouges, abrasions, and scratches. A threat assessment is needed to identify impact damage severity and detectability for design and maintenance. A threat assessment usually includes damage data collected from service plus an impact survey. An impact survey consists of impact tests performed with configured structure, which is subjected to boundary conditions characteristic of the real structure. Many different impact scenarios and locations are typically considered in the survey, which has a goal

of identifying the most critical impacts (i.e., those causing the most serious damage but are least detectable). When simulating accidental impact damage, blunt or sharp impactors should be selected to represent the maximum criticality versus detectability, according to the load conditions (e.g., tension, compression or shear). Until sufficient service experience exists to make good engineering judgments on energy and impactor variables, impact surveys should consider a wide range of conceivable impacts, including runway or ground debris, hail, tool drops, and vehicle collisions. Service data collected over time, can better define impact surveys and design criteria for subsequent products, as well as establish more rational inspection intervals and maintenance practice. Refer to paragraph f(6)(ii)(H) for various combinations of detectability and energy levels to be considered in the damage tolerance and fatigue evaluation.

(3) Discrete Source Damage. The structure should be able to withstand limit static loads (considered as ultimate loads) and fatigue loads which are reasonably expected during a completion of a flight on which damage resulting from obvious discrete source occurs (e.g., hail damage, bird strike, uncontained engine failure, and uncontained high energy rotating machinery failure). The extent of damage should be based on a rational assessment of service mission and potential damage relating to each discrete source.

(D) The use of composite secondary bonding in manufacturing or maintenance requires strict process and quality controls to achieve the reliability needed to use such technology in critical structures (reference AC 21-26). Assuming good process and quality controls, service history has shown that additional damage tolerant design considerations are also needed to ensure the safety of structure with secondary bonds (i.e., random, but an unacceptable number of weak bonds discovered in service). Unless the ultimate strength of each critical bonded joint can be reliably substantiated in production by NDI techniques (or other equivalent, approved techniques), then the limit load capability should be ensured by any or a combination of the following:

(1) Consider isolated disbonds and weak bonds (represented by zero bond strength) in structural elements that use secondary bonding for primary load transfer. The associated disbond size should be up to the limitations provided by redundant design features (i.e., mechanical fasteners or a separate bonding detail). The structure containing such damage should be shown to carry limit load by tests, analyses, or some combination of both. For purposes of test or analysis demonstration, each disbond should be considered separately as a random occurrence (i.e., it is not necessary to demonstrate residual strength with all structural elements disbonded simultaneously).

(2) Each critical bonded joint on each production article should be proof-tested to the critical limit load.

(3) Critical bonded joints that have high static margins of safety (e.g., some rotor blades) may be accepted based on satisfactory service history of like or similar components.

(E) The fatigue load spectrum developed for fatigue testing and analysis purposes should be representative of the anticipated service usage. Low amplitude load levels that can be shown not to contribute to fatigue damage may be omitted (truncated). Reducing maximum load levels (clipping) is generally not accepted.

(F) Environmental effects (temperature and humidity representative of the expected service usage) on the static and fatigue behavior and damage growth should be considered. Unless tested in the environment, appropriate environmental knock down factors for the static and the fatigue test articles should be derived and applied in the evaluation.

(G) Variability in fatigue behavior should be covered by appropriate load or life scatter factors and these factors should take into account the number of specimens tested.

(H) The following Figure AC 29.573-2 illustrates the extent of the impact damage that needs to be considered in the damage tolerance and fatigue evaluation.

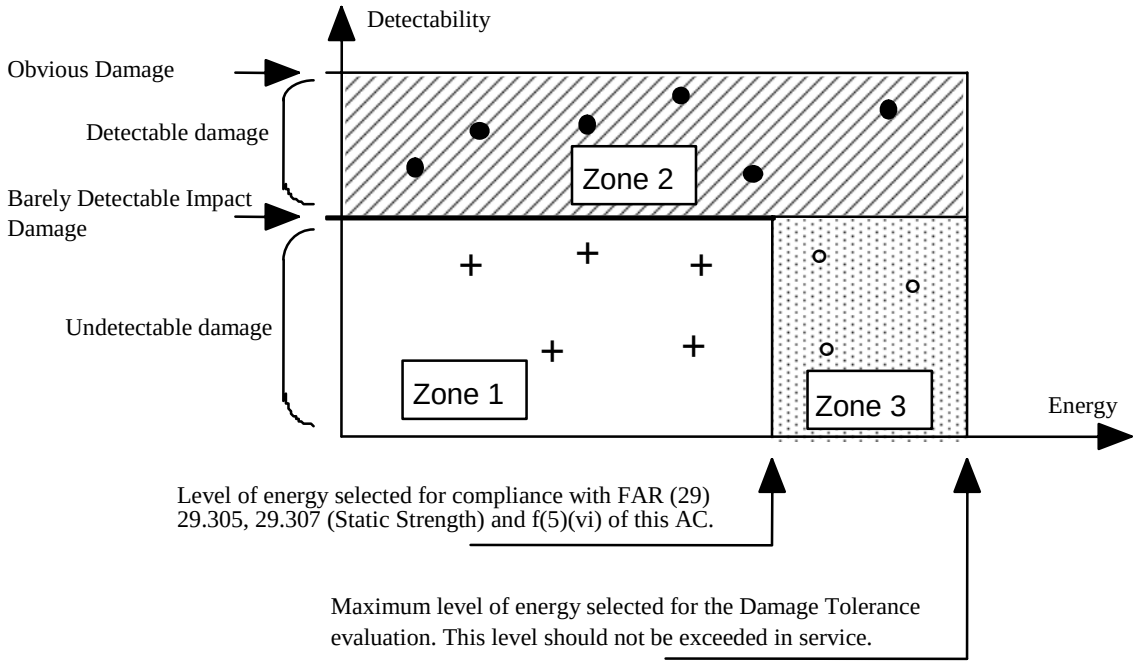

Figure AC 29.573-2. Characterization of Impact Damage

(1) Both the energy level associated with the static strength demonstration and the maximum energy level associated with the damage tolerance evaluation (depicted in Figure AC 29.573-2) are dependent on the part of the structure under evaluation and a threat assessment.

(2) Obvious impact damage is used to define the threshold from which damage is readily detectable and appropriate actions may be taken before the next flight.

(3) Barely Detectable Impact Damage is the state of damage at the threshold of detectability for the approved inspection procedure. Barely Visible Impact Damage (BVID) is that threshold of visually detectable damage associated with a detailed visual inspection procedure.

(4) Detectable Damage is the state of damage that can be reliably detected at scheduled inspection intervals. Visible Impact Damage (VID) is that threshold associated with the type of damage that should be detectable during a detailed visual inspection.

(5) Three Zones are depicted by this figure:

Zone 1: Since the damage is not detectable, Ultimate Load capability is required. The provisions of paragraph f(5) provide a means of compliance.

Zone 2: Since the damage can be detected at a scheduled inspection, Limit Load (considered as Ultimate load) capability is the minimum requirement for this damage.

Zone 3: Since the damage is not detectable with the proposed in-service inspection procedures, ultimate load capability is required, unless an alternate procedure can show an equivalent level of safety. For example, residual strength lower than ultimate may be used in association with improved inspection procedures or with a probabilistic approach showing that the occurrence of energy levels is low enough so that an acceptable level of safety can be achieved.

Of the three zones, only Zone 3 may have a residual strength requirement that can vary with alternate procedures or the probability of damage occurrence or both. In either case, any compromise for residual strength requirements less than the ultimate load requirement should only be considered when pursuing one of the options under the damage tolerant fail-safe means of compliance, as described in the following section, f(6)(iii)(B).

One example of the use of alternate procedures is for the rare damage threat from a high energy, blunt impact (e.g., service vehicle collision). Depending on the selected maintenance inspection scheme, such damage may fall under Zone 3. When considering such damage in the design of a part, it may be shown to be damage

tolerant fail safe, even though the damage is not detectable, based on a very low probability of occurrence. As a result, the design may have sufficiently high residual strength (e.g., below Ultimate, but well above limit load capability to ensure safety without detection for long periods of time). If it is further determined that such impact events usually occur with the knowledge of maintenance or aircraft service personnel, then the alternate procedures may be added to the Instructions for Continued Airworthiness. For example, advanced inspection methods, which can detect damage from high-energy blunt impacts, may be used as alternate procedures to minimize the risk of catastrophic failure for such Zone 3 damage.

(iii) MEANS OF COMPLIANCE - For each PSE, inspections, replacement times, or other procedures must be established as necessary to avoid catastrophic failure. Compliance with the requirements of § 29.573(e) and (f) should be shown by one, or a combination of, the methods described subsequently. Generally, replacement times are established using Damage Tolerance Safe Life Evaluations and Inspection Intervals are established using Fail Safe Evaluations. From current state-of-the-art rotorcraft applications, it is widely accepted that composite materials have good flaw and damage tolerance capabilities and therefore the supplemental procedures may only be rarely necessary. Damage tolerance evaluations are best suited for composite structures, particularly those with structural redundancy and inherent resistance to damage growth. Damage resulting from anomalous or accidental events must be considered in the damage tolerant evaluations. The damage tolerant evaluation for replacement times and inspection intervals is to be used unless shown that neither can be achieved within the limitations of geometry, inspectability, or good design practice. In that case, supplemental procedures must be established and submitted to the FAA/AUTHORITY for approval. In any case, the FAA/AUTHORITY must approve the methodology used for compliance to § 29.573.

The substantiation method(s) should be chosen so that the structure is protected against catastrophic failure from each of the threats identified in paragraph f(6)(ii)(C) of this AC by a specific procedure (inspection, replacement time, or other procedure). For example, a manufacturing-related void of a specific allowable size could be substantiated by means of a replacement time method with no scheduled inspection. An accidental impact in the same area could be substantiated by an inspection method with no specific replacement time. The result could be one structure with several different inspection requirements (location, method, and interval) and a fixed replacement time as well. This combination of procedures assures that each threat is covered.

The fatigue substantiation should include sufficient coupon, element, sub-element, or component tests to establish the fatigue scatter, curve shapes, and the environmental effects. The substantiation should include full-scale, component, or sub-component fatigue testing but also may be accomplished by analysis supported by test evidence. When spectrum testing is used, the lowest load levels can be eliminated from the spectrum if they can be shown to be non-damaging. The substantiation should include a static strength evaluation to show that the required residual strength and adequate

stiffness, accounting for the effects of environment, are retained for the life of the structure or the appropriate inspection interval. Damage as determined in paragraph f(6)(ii) of this AC for the specific structure being substantiated should be imposed at each critical area of the structure.

(A) Damage Tolerant Safe-Life Evaluation. This is a "No-Growth" method in which the structure, with damage present, is able to withstand repeated loads of variable magnitude without detectable damage growth for the life of the rotorcraft or within a specified replacement time. This evaluation may be used to substantiate any type of damage that will remain in-service for the life of the part.

No specific inspection requirements are generated from the test program in this method. However, compliance with routine inspections for cracking, delaminations, and service damage and other limitations prescribed in accordance with § 29.1529 are always required. Compliance using full-scale, component, or sub-component fatigue testing can be accomplished by either of the following methods:

(1) S-N Method. This method is based on determining the point where initiation of growth occurs for the damage present at critical locations in the structure. AC 27-1B, Change 1, AC 27 MG-11, provides guidance that may be appropriate for this method. The method utilizes one or more full-scale, component, or sub-component test specimens subjected to constant-amplitude or spectrum loading applied in a distribution on the structure that is representative of critical flight conditions. Any indication of growth of the imposed damage and defects, or structurally significant cracking, disbonding, splintering, or delaminating of the composite, defines the fatigue initiation characteristic of the structure in terms of applied load and cycles. Working S-N curves are established from the mean curve using strength or cycle reductions or both to account for fatigue scatter and environmental effects. Flight loads are compared to this working curve, and if any intercepts occur, a cumulative damage calculation is conducted to establish the component retirement time. Compliance with the ultimate load requirements should be demonstrated at the completion of the fatigue test.

(2) Life-Test Method. This method uses spectrum fatigue testing to verify the absence of damage growth over a large number of cycles that are equivalent to a lifetime of expected usage. The method uses one or more full-scale, component, or sub-component test specimens subjected to spectrum fatigue loading applied in a representative distribution of flight loads, including Ground-Air-Ground (GAG) loads. Fatigue test loads should be increased by factors for environment and fatigue strength scatter. The load may also be increased using an S-N curve approach to reduce the duration of the test. Any significant growth of the imposed damage, or structurally significant cracking, disbonding, splintering, or delamination of the composite during the test constitutes failure to achieve the desired lifetime. However, the equivalent life demonstrated at the time of inception of damage growth or cracking can be used as a retirement time for the component. Compliance with the ultimate load requirements should be demonstrated at the completion of the fatigue test.

(B) Damage Tolerant Fail-Safe (Residual Strength with Detectable Damage) Evaluation. This method establishes inspection intervals to ensure that the structure remaining after a partial failure is able to withstand design limit loads without failure or excessive structural deformations within a specified inspection interval. If the damage is detected in an inspection, the structure should be either replaced or repaired to restore ultimate load capability. Evaluation of Zone 3 damage should have sufficiently high residual strength and, if necessary, supplemental procedures should be established to minimize the risk of catastrophic failure. Full-scale, component, or sub-component testing should be accomplished using one or more specimens subjected to constant amplitude or spectrum loading applied in a manner representative of flight load conditions. The test loads should be increased by factors that account for environment and fatigue strength scatter. The results of the testing can be used to manage the structure in one or a combination of the three methods described subsequently.

(1) No Growth Evaluation. This approach is appropriate for inspectable in-service damage which does not grow in service. See Figure AC 29.573-3. (Damage growth should be substantiated using either method described in f(6)(iii)(B)(2) or f(6)(iii)(B)(3)). Structural details, elements, sub-components, and components of critical structural areas, or full-scale structures, should be tested under repeated loads for validating a no-growth approach to the damage tolerance requirements. The number of cycles applied to validate a no-growth concept should be statistically significant, and may be determined by load or life considerations or both. Residual strength testing or evaluations should be performed after repeated load cycling demonstrating that the residual strength of the structure is equal to or greater than limit load considered as ultimate. Moreover, it should be shown that stiffness properties have not changed beyond acceptable levels. Inspection intervals should be established, considering the residual strength capability associated with the assumed damage. The intent of this is to assure that structure is not exposed to an excessive period of time with static margins less than ultimate, providing a lower safety level than in the typical slow growth situation, as illustrated by the Figure AC 29.573-3. Once the damage is detected, the component is either repaired to restore ultimate load capability or replaced.

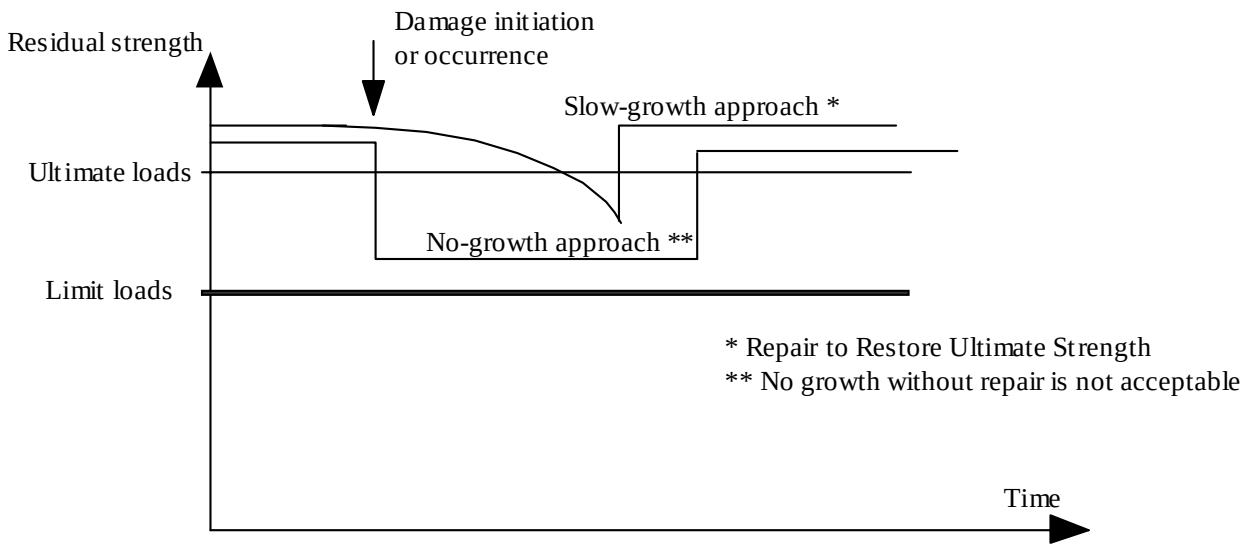

Figure AC 29.573-3. Residual Strength vs. Time

The lower the residual strength of a structure after an accidental damage event, the shorter the inspection interval should be. Considerations of both inspectability and impact surveys (including probability of occurrence) for specific structure may be used to isolate the most critical threats to consider in setting a maintenance inspection interval. Knowledge of the residual strength for a given critical damage is also needed for such an evaluation. If it is known that the design is capable of handling large and clearly detectable damage, while maintaining a residual strength well above limit load, a less rigorous engineering approach may be applied in establishing the inspection interval.

(2) Slow Growth Evaluation. This method is applicable when the damage grows in the test and the growth rate is shown to be slow, stable, and predictable, as illustrated in Figure AC 29.573-4. An inspection program should be developed consisting of the frequency, extent, and methods of inspection for inclusion in the maintenance plan. Inspection intervals should be established so that the damage will have a very high probability of detection between the time it becomes initially inspectable and the time at which the extent of the damage reduces the residual static strength to limit load (considered as ultimate), including the effects of environment. For any damage size that reduces the load capability below ultimate, the component is either repaired to restore ultimate load capability or replaced. Should functional impairment (such as unacceptable loss of stiffness) occur before the damage becomes otherwise critical, this should be accounted for in the development of the inspection program.

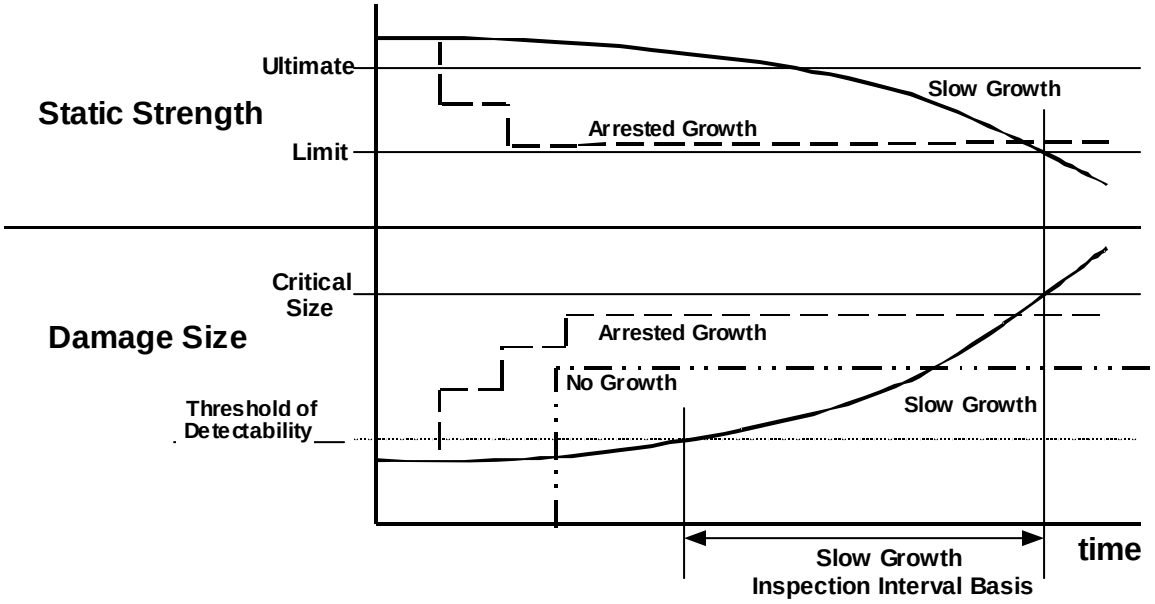

Figure AC 29.573-4. Illustration of Residual Strength and Damage Size Relationships for Fail-Safe Substantiation

(3) Arrested Growth Evaluation. This method is applicable when the damage grows, but the growth is mechanically arrested or terminated before becoming critical (residual static strength reduced to limit load), as illustrated in Figure AC 29.573-4. Arrested Growth may occur due to design features such as a geometry change, reinforcement, thickness change, or a structural joint. This approach is appropriate for inspectable arrested growth damage. Structural details, elements, and sub-components of critical structural areas, or full-scale structures, should be tested under repeated loads for validating an arrested growth approach to the flaw tolerance requirements. The number of cycles applied to validate an arrested growth concept should be statistically significant, and may be determined by load or life considerations or both. Residual strength testing or evaluation should be performed after repeated load cycling and a demonstration that the residual strength of the structure is equal to or greater than limit load considered as ultimate. Moreover, it should be shown that stiffness properties have not changed beyond acceptable levels. Inspection intervals should be established, considering the residual strength capability associated with the arrested growth damage. The intent of this is to assure that structure is not exposed to an excessive period of time with static margins less than ultimate, providing a lower safety level than in the typical slow growth situation, as illustrated by Figure AC 29.573-3. For any damage size that reduces the load capability below ultimate, the component is either repaired to restore ultimate load capability or replaced.

The lower the residual strength of a structure after an arrested growth event, the shorter the inspection interval should be. Considerations of both inspectability and impact surveys (including probability of occurrence) for specific structure may be used to isolate the most critical threats to consider in setting a maintenance inspection interval.

Knowledge of the residual strength for a given critical damage is also needed for such an evaluation. If it is known that the design is capable of handling large and clearly detectable damage, while maintaining a residual strength well above limit load, a less rigorous engineering approach may be applied in establishing the inspection interval.

(C) Combination of Damage Tolerant Safe Life and Fail Safe Evaluations. Generally, it may be appropriate to establish both a replacement time and an inspection program for a given structure as calculated by the Damage Tolerant Safe Life and Fail Safe Evaluations.

(D) Other Procedures. Other procedures are allowed according to paragraph 29.573(d). Such alternative procedures must still provide the same degree of damage tolerance to the same identified threats as the replacement time or inspection interval methods.

One possible alternate approach is the use of indirect damage detection methods instead of the specific mandated inspection procedures that are determined in the Fail Safe Evaluations of f(6)(iii)(B). These indirect detection methods should be documented and shown to have the same degree of reliability, repeatability, and margin provided by a conventional inspection approach. These methods could include: (1) establishing measurable vibration or blade out-of-track conditions and limits, (2) defining indirect inspections which would detect damage, and (3) in-flight detecting of damage by means of monitoring and warning devices.

(E) Supplemental Procedures. If the damage tolerant evaluations as described previously cannot be achieved within the limitations of geometry, inspectability, or good design practice, a fatigue evaluation using supplemental procedures may be proposed to the certifying authority per § 29.573(h). A rationale must be provided as to why the damage tolerance criteria cannot be satisfied for the specific PSE, locations, and threats considered. In addition, the types of damage considered in the evaluations must be identified. Finally, supplemental procedures must be established to minimize the risk of catastrophic failure with the damages considered.

(iv) Additional Considerations for DAMAGE TOLERANCE AND FATIGUE Evaluations.

(A) Experience with the application of methods of fatigue and damage tolerance evaluations indicates that a relevant test background should exist in order to achieve the design objective. It is the general practice within industry to conduct damage tolerance tests for design information and guidance purposes. It is crucial that the critical structure be identified and tested to the proper flight and ground loads. In the fatigue and flaw tolerance evaluation, the following items must be considered:

(B) Identification of the structure to be considered in each evaluation (a failure mode and effects analysis or similar method should be used).

(1) Identification of Principal Structural Elements. Principal structural elements are those that contribute significantly to carrying flight and ground loads and whose failure could result in catastrophic failure of the rotorcraft. Typical examples of such elements are:

(i) Rotor blades and attachment fittings.

(ii) Rotor heads, including hubs, hinges, and some main rotor dampers.

(iii) Control system components subject to repeated loading, including control rods, servo structure, and swashplates.

(iv) Rotor supporting structure (lift path from airframe to rotorhead).

(v) Fuselage, including stabilizers and auxiliary lifting surfaces, airframe provisions for engine and transmission mountings.

(vi) Main fixed or retractable landing gear and fuselage attachment structure.

(2) Identification of Locations Within Principal Structural Elements to be Evaluated. The locations of damage to structure for damage tolerance evaluation can be determined by analysis or by fatigue test on complete structures or subcomponents. However, tests will 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 following should be considered:

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

(ii) Locations where analysis shows high stress or low margins of safety;

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

(iv) Locations of potential fatigue damage identified by fatigue analysis;

(v) Locations where the stresses in adjacent elements will be at a maximum with an element in the location failed;

(vi) Partial fracture locations in an element where high stress concentrations are present in the residual structure;

(vii) Locations where detection would be difficult;

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

(3) In addition, the areas of probable damage from sources such as a severe corrosive or fretting environment, a wear or galling environment, or a high maintenance environment should be determined from a review of the design and past service experience.

(C) The stresses and strains (steady and oscillatory) associated with all representative steady and maneuvering operating conditions expected in service.

(D) The frequency of occurrences of various flight conditions and the corresponding spectrum of loadings and stresses.

(E) The fatigue strength, fatigue crack propagation characteristics of the materials used and of the structure, and the residual strength of the damaged structure.

(F) Inspectability, inspection methods, and detectable flaw sizes.

(G) Variability of the measured stresses of paragraph f(6)(iv)(C), the actual flight condition occurrences of paragraph f(6)(iv)(D), and the fatigue strength material properties of paragraph f(6)(iv)(E).

(v) FLIGHT STRAIN MEASUREMENT PROGRAM.

(A) General. Subsequent to design analysis, in which aircraft loads and associated stresses are derived, the stress level or loads are to be verified by a carefully controlled flight strain measurement program. (This guidance is similar to that of AC 27-1B, MG 11.)

(B) Instrumentation.

(1) The instrumentation system used in the flight strain measurement program should accurately measure and record the critical strains under test conditions associated with normal operation and specific maneuvers. The location

and distribution of the strain gauges should be based on a rational evaluation of the critical stress areas. This may be accomplished by appropriate analytical means supplemented, when deemed necessary, by strain sensitive coatings or photoelastic methods. The distribution and number of strain gauges should cover the load spectrum adequately for each part essential to the safe operation of the rotorcraft as identified in § 29.573(c)(1). Other devices such as accelerometers may be used as appropriate.

(2) The corresponding flight parameters (airspeed, rotor RPM, center-of-gravity accelerations, etc.) should also be recorded simultaneously by appropriate methods. This is necessary to correlate the loads and stresses with the maneuver or operating conditions at which they occurred.

(3) The instrumentation system should be adequately calibrated and checked periodically throughout the flight strain measurement program to ensure consistent and accurate results.

(C) Parts to be Strain-Gauged. Fatigue critical portions of the rotor systems, control systems, landing gear, fuselage, and supporting structure for rotors, transmissions, and engine are to be strain-gauged. For rotorcraft of unusual or unique design, special consideration might be necessary to ensure that all the essential parts are evaluated.

(D) Flight Regimes and Conditions to be investigated.

(1) Typical flight and ground conditions to be investigated in the flight strain measurement program are given in paragraphs c. and d. of AC 27-1B, MG 11.

(2) The determination of flight conditions to be investigated in the flight strain measurement program should be based on the anticipated use of the rotorcraft and, if available, on past service records for similar designs. In any event, the flight conditions considered appropriate for the design and application should be representative of the actual operation in accordance with the rotorcraft flight manual. In the case of multiengine rotorcraft, the flight conditions concerning partial engine-out operation should be considered in addition to complete power-off operation. The flight conditions to be investigated should be submitted in connection with the flight evaluation program.

(3) The severity of the maneuvers investigated during the flight strain survey should be at least as severe as the maneuvers likely to occur in service.

(4) All flight conditions considered appropriate for the particular design are to be investigated over the complete rotor speed, airspeed, center of gravity, altitude, and weight ranges to determine the most critical stress levels associated with each flight condition. The temperature effects on loads as affected by

elastomeric components are to be investigated. To account for data scatter and to determine the stress levels present, a sufficient amount of data points should be obtained at each flight condition. Consideration can be given to the use of scatter factors in determining the sufficiency of data points. In some instances, the critical weight, center of gravity, and altitude ranges for the various maneuvers can be based on past experience with similar design. This procedure is acceptable where adequate flight tests are performed to substantiate such selections. The combinations of flight parameters that produce the most critical stress levels should be used in the evaluation.

(vi) FREQUENCY OF LOADING.

(A) Types of Operation.

(1) The probable types of operation (transport, utility, etc.) for the rotorcraft should be established. The type of operation can have a major influence on the loading environment. In the past, rotorcrafts have been substantiated for the most critical general types of operation with some consideration of special, occasional types of operation. To assure that the most critical types of operation are considered, each major rotorcraft structural component should be substantiated for the most critical types of operation as established by the manufacturer. The types of operation shown below should be considered and, if applicable, used in the substantiation:

(i) Long flights to remote sites (low ground-air-ground cycles but high cruising speeds).

(ii) Typical, general types of operation.

(iii) Short flights as used in logging operations.

(2) One means is to substantiate for the most severe type of operation; however, this method is not always economically feasible.

(3) A second means is to quantify the influence of mission type on fatigue damage by adding to or replacing hour limitations by flight cycle limitations (if properly defined and easily identifiable by the crew, for example: one landing, one load transportation). A special type of flight hour limitation replacement using factorization of flight hours for multiple types of operations may be feasible if continuing manufacturers' technical support is provided and documented; i.e., the manufacturer either provides the factorization analyses or checks them on a continuing basis for each type of rotorcraft operation.

(4) Where one or more operations are not among the general uses intended for the rotorcraft, the rotorcraft flight manual should state in the limitations section that the intended use of the rotorcraft does not include certain missions or repeated maneuvers (i.e., logging with its high number of takeoffs and

landings per hour). A note to this effect should also appear in the rotorcraft airworthiness limitations section of the maintenance manual prepared in accordance with §§ 29.573 and 29.1529.

(5) Should subsequent usage of the rotorcraft encompass a mission for which the original structural substantiation did not account, the effects of this new mission environment on the frequency of loading and structural substantiation should be addressed and where practicable, in the interest of safety, a reassessment made. If this reassessment indicates the necessity for revised retirement times, those new times may be limited to aircraft involved in the added mission provided:

(i) Changes are adopted through the airworthiness directives process, and;

(ii) Proper part re-identification is established;

(iii) A Rotorcraft Flight Manual (RFM) supplement outlining the limitations is approved;

(iv) An airworthiness limitations section (ALS) supplement is approved; or

(v) An appropriate combination of part re-identification, RFM supplement, or airworthiness limitation section supplement is approved.

(B) Loading Spectrum. The spectrum allocating percentage of time or frequencies of occurrence to flight conditions or maneuvers is to be based on the expected usage of the rotorcraft. This spectrum is to be established so that it is unlikely that actual usage will subject the structure to damage beyond that associated with the spectrum. Considerations to be included in developing this spectrum should include prior knowledge based on flight history recorder data, design limitations established in compliance with § 29.309, and recommended operating conditions and limitations specified in the rotorcraft flight manual or instructions for continued airworthiness (ICA). The distribution of times at various forward flight speeds should reflect not only the relation of these speeds to V_{NE} but also the recommended operating conditions in the rotorcraft flight manual or ICA that govern V_c or cruise speed. It is desirable to conduct the flight strain-gauge program by simulating the usage as determined previously, with continuous recording of stresses and loads, thus obtaining directly the stress or load spectra for structural elements.

(7) The seventh major area is the dynamic loading and response requirements of § 29.241, § 29.251, and § 29.629 for vibration and resonance frequency determination and separation for aeroelastic stability and stability margin determination for dynamically critical flight structure. Critical parts, locations, excitation modes, and separations should be identified and substantiated. This substantiation

should consist of analysis supported by tests and tests that account for repeated loading effects and environment exposure effects on critical properties, such as stiffness, mass, and damping. This must be accomplished to assure that the initial stiffness, residual stiffness, proper critical frequency design, and structural damping are provided as necessary to prevent vibration, resonance, and flutter problems.

(A) All vibration and resonance critical composite PSE must be identified and properly evaluated.

(B) All flutter-critical composite PSE must be identified and properly evaluated. This structure must be shown by analysis to be flutter free to 1.1 V_{NE} (or any other critical operating limit, such as V_D, for a VSTOL aircraft) with the extent of damage for which residual strength and stiffness are demonstrated.

(C) Where appropriate, crash impact dynamics considerations should be taken into account to ensure proper crash resistance and a proper level of occupant safety for an otherwise survivable impact.

(8) The eighth area is the special repair and continued airworthiness requirements of §§ 29.611, 29.1529, and 14 CFR part 29 Appendix A, for composite structures. When repair and continued airworthiness procedures are provided in service documents (including approved sections of the maintenance manual or instructions for continued airworthiness), the resulting repairs and maintenance provisions should be shown to provide structure which continually meets the guidance of paragraphs (1) through (7) of this AC. All certification-based repair and continued airworthiness standards, limits, and inspections must be clearly stated and their provisions and limitations clearly documented to ensure continued airworthiness. No composite structural repair should be attempted that is beyond the scope of the applicable approved Structural Repair Manual (SRM) without an engineering design approval by a qualified FAA/AUTHORITY designated representative.

AC 27.573. §27.573 (Amendment 27-XX) DAMAGE TOLERANCE AND FATIGUE EVALUATION OF COMPOSITE ROTORCRAFT STRUCTURE

a. Purpose. This advisory material provides an acceptable means of compliance with the provisions of § 27.573, Amendment XX, of the Federal Aviation Regulations (FAR) dealing with the damage tolerance and fatigue evaluation of transport category composite rotorcraft structure. Paragraph f.(6) specifically addresses the advisory guidance applying to damage tolerance and fatigue evaluation as required by §27.573, Amendment 27-XX. Some information contained in AC 27-1B, MG8 (Amendment 27-30) is repeated and updated as appropriate to preserve the "building block" approach for analyses of composite rotorcraft structure for compliance to §27.573, Amendment 27-XX . (Supplemental guidance can be found in AC 20-107A, "Composite Aircraft Structure.") These procedures address the substantiation requirements for composite material system constituents, composite material systems, and composite structures common to rotorcraft. A uniform approach to composite structural substantiation is desirable, but it is recognized that in a continually developing technical area which has diverse industrial roots, both in aerospace and in other industries, variations and deviations from the procedures described here may be necessary. Significant deviations from this advisory material should be coordinated in advance with the Rotorcraft Directorate.

b. Special Considerations. Since rotorcraft structure is configured uniquely and is inherently subjected to severe cyclic stresses, special consideration is required for the substantiation of all rotorcraft structure, including composites. This special consideration is necessary to ensure that the level of safety intended by the current regulations are attained during the type certification process for all structure with special emphasis on composite structure because of its unique structural characteristics, manufacturing quality and operational considerations, and failure mechanisms.

c. Background.

(1) Historically, rotorcraft have required unique, conservative structural substantiation because of unique configuration effects, unique loading considerations, severe fatigue spectrum effects, and the specialized comprehensive fatigue testing required by these effects. Rotorcraft structural static strength substantiation for both metal and composite structure is essentially identical to that for fixed wing structure once basic loads have been determined. However, rotorcraft structural fatigue substantiation for metals is significantly different from fixed wing fatigue substantiation. Since AC 20-107A, as developed, applies to both fixed wing aircraft and rotorcraft, it, of necessity, was finalized in a broad generic form. Accordingly, a need to supplement AC 20-107A for rotorcraft was recognized during type certification programs. One significant difference in traditional rotorcraft fatigue substantiation programs and fixed wing fatigue programs is the use of multiple component fatigue tests for rotorcraft programs rather than just one full-scale test. Also, constant amplitude, accelerated load

tests are typically used rather than spectrum tests because of the high frequency loads common to rotorcraft operations. These rotorcraft fatigue tests have traditionally involved the generation of stress versus life or cycle (S-N) curves for each critical part (most of which are subjected to the cyclic loading of the main or tail rotor system) using a monotonic (sinusoidal) fatigue spectrum based on maximum and minimum service stress values. Unless configuration differences or flight usage data dictate otherwise, the monotonic fatigue spectrum's period is typically based on six ground-air-ground (GAG) cycles for each flight hour of operation. The S-N curves for the substantiation of each detailed part are typically generated by plotting a curved line through three data points (reference AC 29-2C, AC 29 MG 11, "Fatigue Evaluation of Transport Category Rotorcraft Structure (Including Flaw Tolerance)"). The three data points selected are a short specimen life (low-cycle fatigue), an intermediate specimen life and a long specimen life (high-cycle fatigue). Each raw data point is generated by monotonically fatigue testing at least two full-scale parts to failure or run out for each data point on the S-N curve. The raw data point values are then reduced by an acceptable statistical method to a single value for plotting to ensure proper reliability of the associated S-N curve. Order 8110.9, "Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and Other Power Transmission Systems" and AC 27-1B, AC 27 MG11, "Fatigue Evaluation of Rotorcraft Structure", contain comprehensive discussions of the S-N curve generation process. The rotorcraft S-N curve process contrasts sharply with the fixed wing process of using a single full-scale fatigue article (usually an entire wing or airframe, which constitutes a single full-scale assembly data point), generic material or full-scale assembly S-N data (e.g., Metallic Materials Properties Development and Standardization (MMPDS) formerly the MIL-HDBK-5 for metals, MIL-HDBK-17 for composites, or AC 23-13, "Fatigue, Fail-Safe, and Damage Tolerance Evaluation of Metallic Structure for Normal, Utility, Acrobatic, and Commuter Airplanes", which replaced AFS-120-73-2 for full-scale assemblies), a non-monotonic spectrum and relatively large scatter factors to verify or determine the design fatigue life of the full-scale airplane.

(2) Also, rotorcraft have employed and mass-produced composite designs in primary structure (typically main and tail rotor blades) since the early 1950's. This was 10 or more years before composites were type certificated for primary fixed-wing structure in either military or civil aircraft applications (with some notable limited production exceptions, such as the Windecker fixed wing aircraft). In any case, the early 1950 period was well before a clear, detailed understanding of composite structural behavior (especially in the areas of macroscopic and microscopic failure mechanisms and modes) was relatively common and readily available in a usable format for the average engineer working in this field. It also predated the initial issuance of AC 20-107. Currently, much composite design information is proprietary, either to government, industry or both, and many data gathering methods have not been completely standardized. Consequently, a significant variation from laboratory to laboratory in material property value determination methods and results can exist. The early rotor blade designs (as well as current designs) are by nature relatively low strain, tension structure designs. Also, by nature, these designs are not damage or flaw critical. Thus by circumstance as much as design, early composite rotor blade and other

composite rotorcraft designs incorporated an acceptable fatigue tolerance level of safety. In the 1980's, more test data, analytical knowledge, and analytical methodology became available to more completely substantiate a composite design. Current 14 CFR parts 27 and 29 contain many sections to be considered in substantiating composite rotorcraft structure. This advisory material provides the current or updated information from AC 27-1B, MG 8, Amendment 27-30 to supplement the general guidance of AC 20-107A and provides compliance guidance for the requirements of §27.573 Amendment 27-XX for rotorcraft composite structure.

d. Definitions. The following basic definitions are provided as a convenient reading reference. MIL-HDBK-17, and other sources, contain more complete glossaries of definitions.

(1) AUTOCLAVE. A closed apparatus usually equipped with variable conditions of vacuum, pressure, and temperature. Used for bonding, compressing or curing materials.

(2) ALLOWABLES. Both A-basis and B-basis values statistically derived and used for a particular composite design.

(3) BALANCED LAMINATE. A composite laminate in which all laminae at angles other than 0° occur only in ± pairs (not necessarily adjacent).

(4) A-BASIS ALLOWABLE. The "A" mechanical property value is the value above which at least 99 percent of the population of values is expected to fall, with a confidence level of 95 percent.

(5) B-BASIS ALLOWABLE. The "B" mechanical property value is the value above which at least 90 percent of the population of values is expected to fall, with a confidence level of 95 percent.

(6) BOND. The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent.

(7) COCURE. The process of curing several different materials in a single step. Examples include the curing of various compatible resin system pre-pregs, using the same cure cycle, to produce hybrid composite structure or the curing of compatible composite materials and structural adhesives, using the same cure cycle, to produce sandwich structure or skins with integrally molded fittings.

(8) CURE. To change the properties of a thermosetting resin irreversibly by chemical reaction; i.e., condensation, ring closure, or addition. Cure may be accomplished by addition of curing (crosslinking) agents, with or without a catalyst, and with or without heat.

(9) DELAMINATION. The separation of the layers of material in a laminate.

(10) DISBOND. A lack of proper adhesion in a bonded joint. This may be isolated or may cover a majority of the bond area. It may occur at any time in the cure or subsequent life of the bond area and may arise from a wide variety of causes.

(11) FIBER. A single homogeneous strand of material, essentially one-dimensional in the macro-behavior sense, used as a principal constituent in advanced composites because of its high axial strength and modulus.

(12) FIBER VOLUME. The volume of fiber present in the composite. This is usually expressed as a percentage volume fraction or weight fraction of the composite.

(13) FILL. The 90° yarns in a fabric, also called the woof or weft.

(14) GLASS TRANSITION. The reversible change in an amorphous polymer or in amorphous regions of a partially crystalline polymer from (or to) a viscous or rubbery condition to (or from) a hard and relatively brittle one.

(15) GLASS TRANSITION TEMPERATURE. The approximate midpoint of the temperature range over which the glass transition takes place.

(16) HYBRID. Any mixture of fiber types (e.g., graphite and glass).

(17) IMPREGNATE. An application of resin onto fibers or fabrics by several processes: hot melt, solution coat, or hand lay-up.

(18) LAMINA. A single ply or layer in a laminate in which all fibers have the same fiber orientation.

(19) LAMINATE. A product made by bonding together two or more layers or laminae of material or materials.

(20) LOW STRAIN LEVEL. As used herein, is defined as a principal, elastic axial gross strain level that for a given composite structure provides for no flaw growth and thus provides damage tolerance of the maximum defects allowed during the certification process using the approved design fatigue spectrum.

(21) MATERIAL SYSTEM CONSTITUENT. A single constituent (ingredient) chosen for a material system (e.g., a fiber, a resin).

(22) MATERIAL SYSTEM. The combination of single constituents chosen (e.g., fiber and resin).

(23) MATRIX. The essentially homogeneous material in which the fibers or filaments of a composite are embedded in resins which are mainly thermoset polymers in aircraft structure.

(24) MAXIMUM STRUCTURAL TEMPERATURE. The temperature of a part, panel or structural element due to service parameters such as incident heat fluxes, temperature, and air flow at the time of occurrence of any critical load case, (i.e., each critical load case has an associated maximum structural temperature). This term is synonymous with the term “maximum panel temperature.”

(25) POROSITY. A condition of trapped pockets of air, gas, or void within a solid materials, usually expressed as a percentage of the total nonsolid volume to the total volume (solid + nonsolid) of a unit quantity of material.

(26) PRE-PREG, PREIMPREGNATED. A combination of mat, fabric, nonwoven material, tape, or roving already impregnated with resin, usually partially cured, and ready for manufacturing use in a final product which will involve complete curing. Pre-preg is usually drapable, tacky and can be easily handled.

(27) RESIN. An organic material with indefinite and usually high molecular weight and no sharp melting point.

(28) RESIN CONTENT. The amount of matrix present in a composite either by percent weight or percent volume.

(29) SECONDARY BONDING. The joining together, by the process of adhesive bonding, of two or more already-cured composite parts, during which the only chemical or thermal reaction occurring is the curing of the adhesive itself. The joining together of one already-cured composite part to an uncured composite part, through the curing of the resin of the uncured part, is also considered for the purposes of this advisory circular to be a secondary bonding operation. (See COCURING).

(30) SHELF LIFE. The lengths of time a material, substance, product, or reagent can be stored under specified environmental conditions and continue to meet all applicable specification requirements and remain suitable for its intended function.

(31) STRAIN LEVEL. As used herein, is defined as the principal axial gross strain of a part or component due to the principal load or combinations of loads applied by a critical load case considered in the structural analysis (e.g., tension, bending, bending-tension, etc.). Strain level is generally measured in thousandths of an inch per unit inch of part or microinches/per inch (e.g., .003 in/in equals 3000 microinches/inch).

(32) SYMMETRICAL LAMINATE. A composite laminate in which the ply orientation is symmetrical about the laminate midplane.

(33) TAPE. Hot melt impregnated fibers forming unidirectional pre-preg.

(34) THERMOPLASTIC. A plastic that repeatedly can be softened by heating and hardened by cooling through a temperature range characteristic of the plastic, and when in the softened stage, can be shaped by flow into articles by molding or extrusion.

(35) THERMOSET (OR CHEMSET). A plastic that once set or molded cannot be re-set or remolded because it undergoes a chemical change; (i.e., it is substantially infusible and insoluble after having been cured by heat or other means).

(36) WARP. Yarns extended along the length of the fabric (in the 0° direction) and being crossed by the fill yarns (90° fibers).

(37) WORK LIFE. The period during which a compound, after mixing with a catalyst, solvent, or other compounding constituents, remains suitable for its intended use.

(38) Damage Tolerance. The attribute of the structure that permits it to retain its required residual strength for a period of use after the structure has sustained a given level of fatigue, corrosion, accidental or discrete source damage.

(39) Catastrophic Failure. An event that could prevent continued safe flight and landing.

(40) Damage. A generic term for structural anomalies caused by manufacturing (processing, fabrication, assembly or handling) or service usage. Trimming, fastener installation, or foreign object impact are potential sources of damage, along with fatigue and environmental effects.

(41) Damage Tolerant Safe Life. Capability of structure with damage present to survive expected repeated loads of variable magnitude without detectable damage growth and to maintain ultimate load capability throughout service life of the rotorcraft.

(42) Damage Tolerant Fail-Safe. The capability of structure remaining after a partial failure to withstand design limit loads without catastrophic failure within an inspection period.

(43) Multiple Load Path. Structure providing two or more separate and distinct paths of structure that will carry limit load after complete failure of one of the members.

(44) Active Multiple Load Path. Structure providing two or more load paths that are all loaded during operation to a similar load spectrum.

(45) Passive Multiple Load Path. Structure providing load paths with one or more of the members (or areas of a member) relatively unloaded until failure of the other member or members.

(46) Accidental Damage. Discrete damage which may occur in service use or in manufacturing due to impacts or collisions, such as dents, scratches, gouges, abrasions, disbonds, splintering, and delaminations.

(47) Intrinsic or discrete manufacturing defects. Intrinsic or discrete imperfections or flaws related to manufacturing operations, processing or assembly such as voids, gaps, porosity, inclusions, fiber dislocation, disbonds, and delaminations.

(48) Fatigue or Environmental Damage. Structural damage related to fatigue or environmental effects such as delaminations, disbonds, splintering, or cracking.

(49) Design Limit Loads. The maximum loads to be expected in service, as defined by § 29.301(a).

(50) As-Manufactured. Product or component that has passed the applicable quality control process and has been found to conform to the approved design within the allowable tolerances.

(51) Residual Strength. The strength retained for some period of unrepaired use after a failure or partial failure due to fatigue, accidental, or discrete source of damage.

(52) Principal Structural Element (PSE). A structural element that contributes significantly to the carrying of flight or ground loads and whose failure can lead to catastrophic failure of the rotorcraft.

(53) Coupon. A small test specimen (e.g., usually a flat laminate) for evaluation of basic lamina or laminate properties or properties of generic structural features (e.g., bonded or mechanically fastened joints).

(54) Point Design. An element or detail of a specific design which is not considered generically applicable to other structure for the purpose of substantiation (e.g., lugs and major joints). Such a design element or detail can be qualified by test or by a combination of test and analysis.

(55) Element. A generic element of a more complex structural member (e.g., skin, stringers, shear panels, sandwich panels, joints, or splices).

(56) Detail. A non-generic structural element of a more complex structural member (e.g., specific design configured joints, splices, stringers, stringer runouts, or major access holes).

(57) Subcomponent. A major three-dimensional structure which can provide complete structural representation of a section of the full structure (e.g., stub box, section of a spar, wing panel, wing rib, body panel, or frames).

(58) Component. A major section of the airframe structure (e.g., wing, fin, body, horizontal stabilizer) which can be tested as a complete unit to qualify the structure.

(59) Environment. External, non accidental conditions (excluding mechanical loading), separately or in combination, that can be expected in service and which may affect the structure (e.g., temperature, moisture, UV radiation, and fuel).

e. Related Regulatory and Guidance Material.

Document	Title
FAA Order 8110.9	Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and other Power Transmission Systems
AC 27-1B, MG 11	"Fatigue Evaluation of Rotorcraft Structure"
AC 20-107A	"Composite Aircraft Structure"
AC 21-26	"Quality Control for the Manufacture of Composite Materials"
MIL-HDBK-17	"Composite Material Handbooks"
AC 29-2C, MG 11	"Fatigue Tolerance Evaluation of Transport Category Rotorcraft Metallic Structure"
DOT/FAA/CT-86/39	Whitehead, R.S., Kan, H.P., Cordero, R., and Seather, R., "Certification Testing Methodology for Composite Structures", October 1986.

f. Procedures for Substantiation of Rotorcraft Composite Structure. The composite structures evaluation has been divided into eight basic regulatory areas to provide focus on relevant regulatory requirements. These eight areas are: (1) fabrication requirements; (2) basic constituent, pre-preg and laminate material acceptance requirements and material property determination requirements; (3) protection of structure; (4) lightning protection; (5) static strength evaluation; (6) damage tolerance and fatigue evaluation; (7) dynamic loading and response evaluation; and (8) special repair and continued airworthiness requirements. Original as well as alternate or substitute material system constituents (e.g., fibers, resins, etc.), material systems (combinations of constituents and adhesives), and composite designs (laminates, co-cured assemblies, bonded assemblies, etc.) should be qualified in accordance with the methodology presented in the following paragraphs. Each regulatory area will be addressed in turn. It is important to remember that proper

certification of a composite structure is an incremental, building block process which involves phased FAA/AUTHORITY involvement and incremental approval in each of the various areas outlined herein. It is recommended that a FAA/AUTHORITY certification team approach be used for composite structural substantiation. The team should consist of FAA/AUTHORITY engineering and cognizant aircraft evaluation group inspector(s), the manufacturing inspector(s), the associated Designated Engineering Representatives (DER's), the associated Designated Manufacturing Inspection Representatives (DMIR's), and cognizant members of the applicant's organization. Personnel who are composites specialists (or are otherwise knowledgeable in the subject) should be primary team member candidates.

Once selected, it is recommended that team meetings be held periodically (possibly in conjunction with type boards) during certification to ensure the building block certification process is accomplished as intended. The team should assure that permanent documentation in the form of reports or other Authority-acceptable documents are included in the certification data package. The documentation includes but is not limited to the structural substantiation reports (both analysis and test), manufacturing processes and quality control, and Instructions for Continued Airworthiness (maintenance, overhaul, and repair manuals). The Airworthiness Limitations Section of the Instructions for Continued Airworthiness is approved by FAA/AUTHORITY engineering. Engineering practices for many of the areas identified below are available in Mil-HDBK-17.

(1) The <u>first</u> area is the <u>fabrication requirements</u> of § 27.605:

(i) The quality control system should be developed considering the critical engineering, manufacturing, and quality requirements and a guidance standard such as AC 21-26, "Quality Control for the Manufacture of Composite Materials." This ensures that all special engineering, or manufacturing quality instructions for composites are presented, evaluated, documented, and approved, using drawings, process and manufacturing specifications, standards, or other equivalent means. This should be one of the early phases of a composite structure certification program, since this represents a major building block for sequential substantiation work. Some important concepts of AC 21-26 are included below.

(ii) Specific allowable defect limits on, for example, fiber waviness, warp defects, fill defects, porosity, hole edge effects, edge defects, resin content, large area debonds, and delaminations, etc., for a particular material system component, laminate design, detailed part, or assembly should be jointly established by engineering, manufacturing, and quality and the associated inspection programs for defect detection created, validated, and approved. Each critical engineering design should consider the variability of the manufacturing process to determine the worse case effects (maximum waviness, disbonds, delaminations, and other critical defects) allowed by the reliability limitations of the approved inspection program.

(iii) If bonds or bond lines such as those typical of rotorcraft rotor blade structure are used, special inspection methods, special fabrication methods or other approved verification methods (e.g., engineering proof tests, reference paragraph g(5)) should be provided to detect and limit disbonds or understrength bonds.

(iv) Structurally critical composite construction fabrication process and procurement specifications, for fabricating reproducible and reliable structure, must be provided and FAA/AUTHORITY approved early during the certification process and should, as a minimum, cover the following:

(A) Vendor and Qualified Parts List (QPL) Control. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both the manufacturing inspection district office (MIDO) and FAA/AUTHORITY engineering) at any time, that their quality control systems ensure on a continuous basis, that only qualified suppliers provide the basic material constituents or material systems (e.g., pre-pregs) that meet approved material specifications. Recommended guidelines for qualification of alternate material systems and suppliers are contained in MIL-HDBK-17. These methods can also be used, periodically for qualification status renewals of existing material systems and suppliers.

(B) Receiving Inspection and In-Process Inspection. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their receiving and in-process quality control systems provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, so that the necessary statistical reliability and confidence levels for the items being inspected (that are specified by engineering) are continuously maintained. This will require periodic standard inspections and engineering characterization tests on basic constituent and material system samples which should be conducted, as a minimum, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY witnessed.

(C) Material System Component Storage and Handling. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their composite material system (or constituent) storage and handling procedures and specifications provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, so that the necessary statistical reliability and confidence levels for the items being inspected (which are specified by engineering) are continuously maintained. This should require, as a minimum, periodic inspections to ensure that proper records are kept on critical parameters (e.g., room temperature “bench” exposure, shelf life, etc.) and that periodic basic constituent and material system characterization tests are conducted, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard

should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY witnessed.

(D) Statistical Validation Level. It is necessary to maintain the minimum required statistical validation level of the quality control system (which should be specified for each critical item or constituent by the approved quality and engineering specifications). The statistical validation level should be defined and approved early in certification. Also, approval and proper usage should be continuously maintained during the entire procurement and manufacturing cycles.

(v) Alternate fabrication and process techniques should be approved and must comply with § 27.605. Any alternate techniques should provide at least the same level of quality and safety as the original technique. Any changes should be presented for FAA/AUTHORITY approval well in advance of the change's production effectivity.

(2) The second area is the basic raw constituent, pre-preg, and laminate material acceptance requirements and material property determination requirements of §§ 27.603 and 27.613. These criteria require application of the critical environmental limits such as temperature, humidity, and exposure to aircraft fluids (such as fuel, oils, and hydraulic fluids), to determine their effect on the performance of each composite material system. Temperature and humidity effects are commonly considered by coupon and component tests utilizing preconditioned test specimens for each material system selected. Material "A" and "B" basis allowable strength values and other basic material properties (based on MIL-HDBK-17 or equivalent procedures) are typically determined by small scale tests, such as coupon tests, for use in certification work. In the case of composites, determination of these basic constituent and material system properties will almost invariably involve the submittal, acceptance, and use of company standards. This is currently necessary because the FAA (new managers of MIL-HDBK-17) has not completed development of "B" basis allowables for inclusion in the handbook. Also, test methods vary somewhat from manufacturer to manufacturer; therefore, individual company results will exhibit some scatter in final material property values. Any company standard that is used should meet or exceed related MIL-HDBK-17 requirements. Material structural acceptance criteria and property determination should, as a minimum, include the following:

(i) Property characterization requirements of all material systems (e.g., pre-pregs, adhesives, etc.) and constituents (e.g., fibers, resins, etc.) should be identified, documented, and approved. These requirements, once approved, should be placed in all appropriate procedures and specifications such as those in paragraph f.(1).

(ii) Moisture conditioning of test coupons, parts, subassemblies, or assemblies should be accomplished in accordance with MIL-HDBK-17, other similar approved methods or per FAA/AUTHORITY approved programs.

(iii) The maximum and minimum temperatures expected in service (as derived from test measurements, thermal analyses on panels and other parts, experience, or a combination) should be determined and accounted for in static and fatigue strength (including damage tolerance) substantiation programs considering associated humidity-induced effects.

(iv) The wet glass transition temperature, Tg, is an important characteristic parameter of amorphous polymers, such as epoxies. It is the temperature below which the polymer behaves like a “glassy” solid and above which it behaves like a “rubbery” solid, i.e., it is the temperature at which there is a very rapid change in physical properties. The change from a hard polymeric material to a rubbery material takes place over a narrow temperature range. A composite material will experience a drastic reduction in matrix-controlled mechanical material properties when loaded in this temperature range. Since the resin is the critical structural constituent in a composite matrix and since Tg exceedance is critical to structural integrity; Tg determination is necessary. The Tg margin methodology of MIL-HDBK-17 should be implemented, i.e., the wet glass transition temperature (Tg) should be 50° F higher than the maximum structural temperature (see definition). For any type of resin or adhesive, an acceptable temperature margin using MIL-HDBK-17 techniques (e.g., consideration of limited high temperature excursions) or equivalent methodologies based on tests or experience, or both should be established and approved early in the certification process.

(v) Local design values should be established by analysis and characterization tests and approved for specific structural configurations (point designs) which include the effects of stress risers (e.g., holes, notches, etc.) and structural discontinuities (e.g., joints, splices, etc.). Proper determination of these values for full-scale design and test should be considered one of the most critical building blocks in substantiating and evaluating a composite structure. These transitional load transfer areas typically produce the highest stresses (and strains) and serve as the initiation sites for many of the failures (including those due to the relatively low interlaminar strength of composites) that occur in service in a full-scale part or assembly. Small scales tests (such as coupon, element, and subcomponent tests), or equivalent approved testing programs, and analytical techniques should be carefully designed, prepared, and approved to evaluate potential “hot spots” and provide accurate simulations and representations of full-scale article stresses and strains in the critical transition areas. Proper certification work in this area will ensure initial safety and continued airworthiness in full-scale production articles.

(vi) The design strain level for each major component and material system should be established so that specified impact damage considerations are defined and properly limited. The effects of the strain levels may be established for each composite material using small-scale characterization tests and then the results should be used to establish or verify the maximum allowable design strain level for each full-scale article. The maximum allowable design strain values selected should also take into account the reliability and confidence levels established for the relevant portions of the quality control system. This methodology is necessary because the

amount and size of flaws in the production article may restrict the allowable level of design strain. In a no-flaw-growth design, the maximum specified impact damage and manufacturing flaw size at the most critical location on the part will be a major factor in determining the maximum allowable elastic strain. This design approach is currently selected for nearly all civil and most military applications; since, under normal conditions, only visual inspections are required in the field (unless unusual external damage circumstances such as a hail storm occur) to maintain the initial level of airworthiness (safety). However, many military applications, because of their demanding missions, employ scheduled field non-destructive inspection (NDI) maintenance, (such as comparative ultrasonics) to ensure that flaw growth either does not occur, is controlled by approved structural repair, or by replacement of affected parts. To date, civil applications have not been presented that desire a flaw growth, phased NDI approach. Therefore, selection of the full-scale article's design strain limit based on small-scale tests for a no flaw growth design is extremely important.

(vii) Composite and adhesive properties should be determined so that detrimental structural creep does not occur under the sustained loads and environments expected in service. Small-scale characterization tests (such as coupon, element, and subcomponent tests) and analysis, which verify and establish the full-scale design criteria and parameters necessary to ensure that detrimental structural creep in full-scale structure does not occur in service, should be conducted early in certification and should be FAA/AUTHORITY-approved.

(viii) Material allowable strength values for full-scale design and testing should be developed using the coupon procedures presented in MIL-HDBK-17 or equivalent. The intent is to represent the material variability including the effects that can occur in multiple batches of material and process runs. At least three batches of material samples should be used in material allowable strength testing. Company standards should be prepared, evaluated and FAA/AUTHORITY-approved early in certification (as part of the building block process), that reflect the material property determination considerations recommended in MIL-HDBK-17 on an equal to or better than basis.

(3) The <u>third</u> area is the <u>protection of structure</u> as required by § 27.609. Protection against thermal and humidity effects and other environmental effects (e.g., weathering, abrasion, fretting, hail, ultraviolet radiation, chemical effects, accidental damage, etc.) should be provided, or the structural substantiation should consider the results of those effects for which total protection is impractical. Determination and approval of worst-case or most conservative operating limits, and damage scenarios should be accomplished. Appropriate flammability and fire-resistance requirements should also be considered in selecting and protecting composite structure. Usually a threat analysis is conducted early in the certification process that identifies the various threats and threat levels for which protection must be provided. This data is then used to construct and submit for approval the methods-of-compliance necessary to provide proper structural protection.

(4) The fourth area is the lightning protection requirements of § 27.610. Protection should be provided and substantiated in accordance with analysis and with tests such as those of AC 20-53B, “Protection of Aircraft Fuel Systems Against Fuel Vapor Ignition Caused by Lightning” and FAA Report DOT/FAA/CT-86/8. For composite structure projects involving rotorcraft certificated to earlier certification bases (which do not automatically include the lightning protection requirements of § 27.610), these requirements should be imposed as special conditions. The design should be reviewed early in the certification process to ensure proper protection is present. The substantiation test program should also be established, reviewed and approved early to ensure proper substantiation.

(5) The fifth area is the static strength evaluation requirements of §§ 27.305 and 27.307 for composite structure. Structural static strength substantiation of a composite design should consider all critical load cases and associated failure modes, including effects of environment, material and process variability, and defects or service damage that are not detectable or allowed by the quality control, manufacturing acceptance criteria, or maintenance documents of the end product. The static strength demonstration should include a program of component ultimate load tests, unless experience exists to demonstrate the adequacy of the analysis, supported by subcomponent tests or component tests to accepted lower load levels. The necessary experience to validate an analysis should include previous component ultimate load tests with similar designs, material systems, and load cases.

(i) The effects of repeated loading and environmental exposure, both of which may result in material property degradation, should be addressed in the static strength evaluation. This can be shown by analysis supported by test evidence, by tests at the coupon, element or subcomponent levels, or alternatively by existing data. Earlier discussions in this AC address the effects of environment on material properties (reference paragraphs f. (2)) and protection of structure (reference paragraphs f.(3)). Static strength tests should be conducted for substantiation of new structure. For the critical loading conditions, two approaches to account for prior repeated loading or environmental exposure for structural substantiation exist.

- In the first approach, the large-scale static test should be conducted on structure with prior repeated loading and conditioned to simulate the environmental exposure and then tested in that environment.

- The second approach relies upon coupon, element, and sub-component test data to assess the possible degradation of static strength after application of repeated loading and environmental exposure. The degradation characterized by these tests should then be accounted for in the static strength demonstration test (e.g., load enhancement), or in the analysis of these results (e.g., showing a positive margin of safety with allowables that include the degrading effects of environment and repeated load).

In practice, the two approaches may be combined to get the desired result (e.g., a large-scale static test may be performed at a temperature with a load enhancement factor to account for moisture absorbed over the aircraft structure's life).

(ii) The strength of the composite structure should be statistically established, incrementally, through a program of analysis and tests at the coupon, element, subcomponent, or component levels. As part of the evaluation, building block tests and analyses at the coupon, element, or subcomponent levels can be used to address the issues of variability, environment, structural discontinuity (e.g., joints, cut-outs or other stress risers), damage, manufacturing defects, and design or process-specific details. Figure AC 27.573-1 provides a conceptual schematic of tests included in the building block approach. The material stress-strain curve should be clearly established, at least through the ultimate design load, for each composite design. As shown in Figure AC 27.573-1, the large quantity of tests needed to provide a statistical basis comes from the lowest levels (coupons and elements) and the performance of structural details are validated in a lesser number of sub-component and component tests. The static strength substantiation program should also consider all critical loading conditions for all critical structure including residual strength and stiffness requirements after a predetermined length of service, e.g., end of life (EOL) (which takes into account damage and other degradation due to the service period).

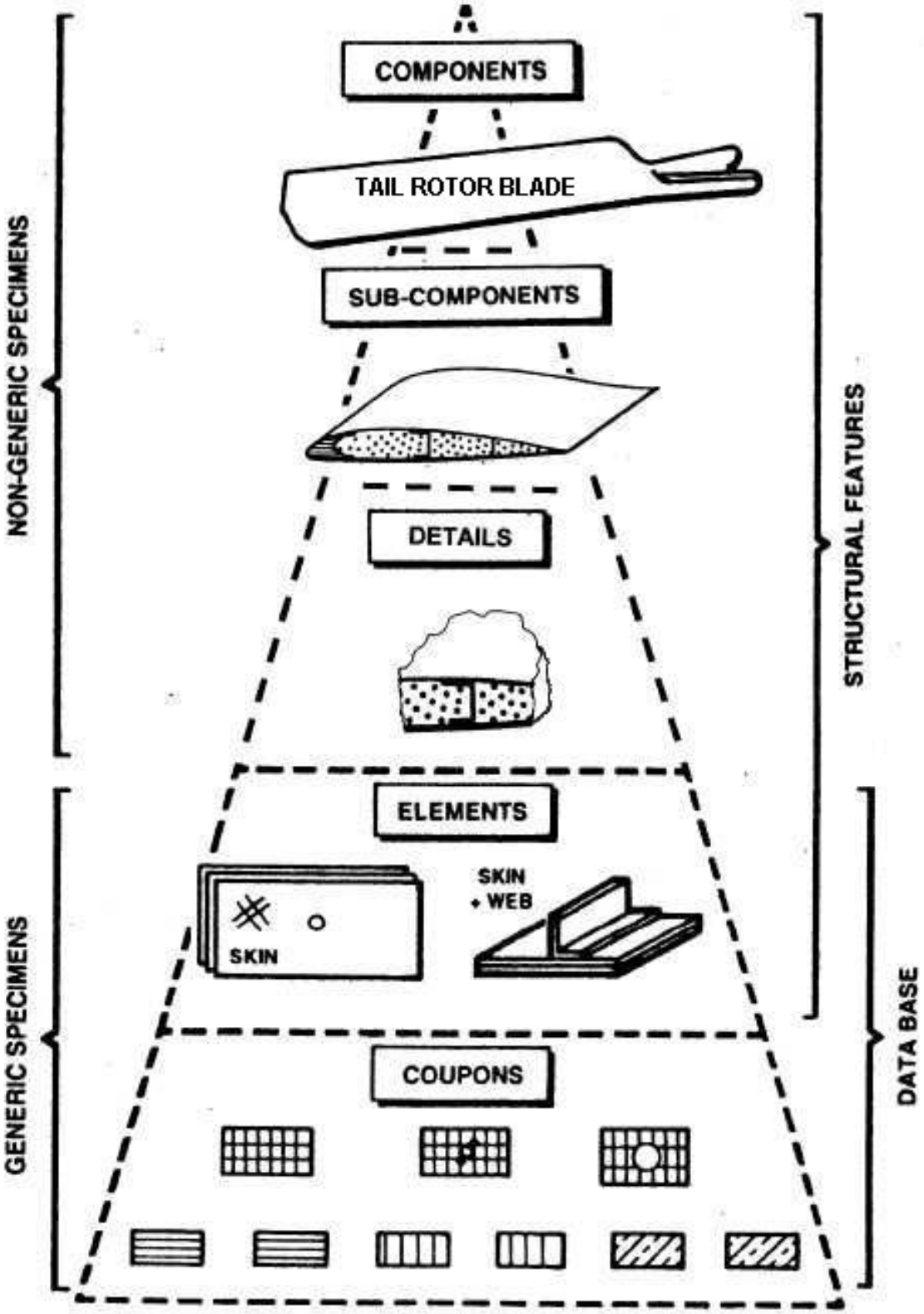

Figure AC 27.573-1. Schematic diagram of building block tests

(iii) Allowables should be evaluated and used as specified in § 27.613. These allowables may be generated at the lamina, laminate, or specific design feature level (e.g. filled hole, lap joint, stringer run-out, etc.), provided they accurately reflect the actual value and variability of the structural strength for the critical failure modes being considered, at each point design where margins need to be established.

(iv) The static test articles should be fabricated and assembled in accordance with production specifications and processes so that they are representative of production structure including defects consistent with the limits established by manufacturing acceptance criteria.

(v) The material and processing variability of the composite structure should be considered in the static strength substantiation. This can be achieved by establishing sufficient process and quality controls to manufacture structure and reliably substantiate the required strength in tests and analyses, which support a building block approach. If sufficient process and quality controls cannot be achieved, it may be necessary to account for greater variability with special factors (§ 27.619) applied to the design. Such factors should be accounted for in the component static tests or analysis.

(vi) It should be shown that impact damage (or other minor discrete source damage) that can be realistically expected from manufacturing and service, but not more than the established threshold of detectability for the selected inspection procedure, will not reduce the structural strength below ultimate load capability. This static strength capability can be shown by analysis supported by test evidence, or by a combination of tests at the coupon, element, subcomponent, and component levels. Later discussions in this AC address the issues associated with damage in excess of that considered in f(5) and drops in residual strength below ultimate load capability (reference paragraph f(6)) below.

(6) The <u>sixth</u> area is the <u>damage tolerance and fatigue evaluation</u> requirements of § 27.573.

(i) BACKGROUND. The static strength determination required by §§ 27.305 and 27.307 establishes the ultimate load capability for composite structures that are manufactured, operated, and maintained with established procedures and conditions. The damage tolerance and fatigue evaluation required by § 27.573 mandates procedures that allow the composite structure to retain the intended ultimate load capability when subjected to expected fatigue loads and conditions during its operational life. The requirements established for the damage tolerance and fatigue evaluation include component replacement times, inspection intervals, or other procedures as necessary to avoid catastrophic failure. These evaluations assume that the baseline ultimate strength capability might be compromised by damage caused by fatigue, environmental effects, intrinsic or discrete flaws, or accidental damage. This damage includes flaws or defects which may occur in manufacturing or maintenance

and which are used to set the ultimate strength capability and establish the manufacturing acceptance criteria. The damage tolerance assessment establishes standards that allow the static strength capability to degrade below the ultimate strength capability assuming such damage occurs within the operational life of the structure. However, when this damage occurs, the remaining structure must withstand expected loads without failure or excessive structural deformations until the damage is detected and the component is either repaired to restore ultimate strength capability or retired.

(ii) GENERAL - The nature and extent of the required analysis or tests on complete structures and portions of the primary structure can be based on applicable previous fatigue or damage tolerant designs, construction, tests, and service experience on similar structures. In the absence of experience with similar designs, Airworthiness Authorities-approved structural development tests of components, subcomponents, and elements should be performed. The following considerations are unique to the use of composite material systems and should be observed for the method of substantiation selected by the applicant.

Rotorcraft structure provides a broad range of composite applications that are quite different in terms of functionality, geometry and inspectability. These include the rotors, the drive shafts, the fuselage, control system components (e.g. push-pull rods), and the control surfaces. When selecting the approach, attention should be given to the composite application under evaluation, the type of potential damage and degradation of the structural design details, the materials used and margin over flight loads. Whatever the approach that may be selected, the following considerations will apply for tests and analysis:

(A) The test articles should be fabricated and assembled in accordance with production specifications and processes so that the test articles are representative of production structure.

(B) The test articles should include material imperfections whose extent is not less than the limits established under the inspection and acceptance criteria used during the manufacturing process and consistent with the inspection techniques used in service (e.g. visual, ultrasonic, X-ray). The initial extent of these imperfections should be discussed and agreed with the FAA/AUTHORITY, taking into account experience in manufacturing and routine in-service inspections. Typical defects to be considered include but are not limited to the following:

(1) Disbonds and weak bonds (considered as disbonds)
(2) Delaminations, fiber waviness, porosity, voids
(3) Scratches, gouges, and penetrations
(4) Impact damage

All of the damages identified in the preceding paragraph (B) above should be derived from the threat assessment described in the following paragraph (C).

(C) Threat Assessment. For each PSE, a threat assessment must be made of the probable locations, types, and sizes of damage

considering fatigue, environmental effects, intrinsic and discrete flaws, and impact or other accidental damage. This determination must be submitted with accompanying rationale to the certifying authority for approval. This rationale may include experience with similar materials, designs, processes (manufacturing, maintenance, and overhaul), structural details, or structure, and may also include service failure evaluations, manufacturing records, overhauls and repair reports, field service reports, incident and accident investigations, service impact surveys, inspectability surveys, and engineering judgment.

Consideration should also be given to factors that:

- Reduce scatter and deviations from nominal structures, such as “frozen processes”, Flight Critical Parts programs, and materials and manufacturing processes to mitigate intrinsic flaws (inclusions and defects).
- Preclude a type of damage by use of a specific design feature (material selection, surface treatment, protective coating, or shielding), a specific stress level (for fatigue damage), or a specific manufacturing inspection process (if it can be shown to be highly reliable, well-controlled and documented, and systematically required).

The assessment should include:

- A systematic evaluation of all the location, types, and sizes of damage and their estimated probability of occurrence.
- A selection or elimination of this damage based on the above estimate
- A verification that the inspection method selected is capable of detecting the damage at the size and location determined.

The types of damage to consider include:

(1) Intrinsic Flaws (imperfections) which are probable to exist in an as-manufactured structure based on the evaluation of the details and potential sensitivities of the specific manufacturing work processes used. The types of flaws to be considered include voids, disbonds, inclusions, foreign objects, resin-rich and resin-starved areas, and improper ply orientation or ply ending. The sizes of the intrinsic flaws considered should be based on the limits established under the manufacturing inspection and acceptance criteria and are expected to remain in service for the life of the structure.

(2) Impact Damage which may occur during manufacturing and in service based on an evaluation of the threats by means of an impact survey and/or service experiences. This type of damage can include dents, penetrations, gouges, abrasions, and scratches. A threat assessment is needed to identify impact damage severity and detectability for design and maintenance. A threat assessment usually includes damage data collected from service plus an impact survey. An impact survey consists of impact tests performed with configured structure, which is subjected to boundary conditions characteristic of the real structure. Many different impact scenarios and locations are typically considered in the survey, which has a goal

of identifying the most critical impacts (i.e., those causing the most serious damage but are least detectable). When simulating accidental impact damage, blunt or sharp impactors should be selected to represent the maximum criticality versus detectability, according to the load conditions (e.g., tension, compression or shear). Until sufficient service experience exists to make good engineering judgments on energy and impactor variables, impact surveys should consider a wide range of conceivable impacts, including runway or ground debris, hail, tool drops, and vehicle collisions. Service data collected over time, can better define impact surveys and design criteria for subsequent products, as well as establish more rational inspection intervals and maintenance practice. Refer to paragraph f(6)(ii)(H) for various combinations of detectability and energy levels to be considered in the damage tolerance and fatigue evaluation.

(3) Discrete Source Damage. The structure should be able to withstand limit static loads (considered as ultimate loads) and fatigue loads which are reasonably expected during a completion of a flight on which damage resulting from obvious discrete source occurs (e.g., hail damage, bird strike, uncontained engine failure, and uncontained high energy rotating machinery failure). The extent of damage should be based on a rational assessment of service mission and potential damage relating to each discrete source.

(D) The use of composite secondary bonding in manufacturing or maintenance requires strict process and quality controls to achieve the reliability needed to use such technology in critical structures (reference AC 21-26). Assuming good process and quality controls, service history has shown that additional damage tolerant design considerations are also needed to ensure the safety of structure with secondary bonds (i.e., random, but an unacceptable number of weak bonds discovered in service). Unless the ultimate strength of each critical bonded joint can be reliably substantiated in production by NDI techniques (or other equivalent, approved techniques), then the limit load capability should be ensured by any or a combination of the following:

(1) Consider isolated disbonds and weak bonds (represented by zero bond strength) in structural elements that use secondary bonding for primary load transfer. The associated disbond size should be up to the limitations provided by redundant design features (i.e., mechanical fasteners or a separate bonding detail). The structure containing such damage should be shown to carry limit load by tests, analyses, or some combination of both. For purposes of test or analysis demonstration, each disbond should be considered separately as a random occurrence (i.e., it is not necessary to demonstrate residual strength with all structural elements disbonded simultaneously).

(2) Each critical bonded joint on each production article should be proof-tested to the critical limit load.

(3) Critical bonded joints that have high static margins of safety (e.g., some rotor blades) may be accepted based on satisfactory service history of like or similar components.

(E) The fatigue load spectrum developed for fatigue testing and analysis purposes should be representative of the anticipated service usage. Low amplitude load levels that can be shown not to contribute to fatigue damage may be omitted (truncated). Reducing maximum load levels (clipping) is generally not accepted.

(F) Environmental effects (temperature and humidity representative of the expected service usage) on the static and fatigue behavior and damage growth should be considered. Unless tested in the environment, appropriate environmental knock down factors for the static and the fatigue test articles should be derived and applied in the evaluation.

(G) Variability in fatigue behavior should be covered by appropriate load or life scatter factors and these factors should take into account the number of specimens tested.

(H) The following Figure AC 27.573-2 illustrates the extent of the impact damage that needs to be considered in the damage tolerance and fatigue evaluation.

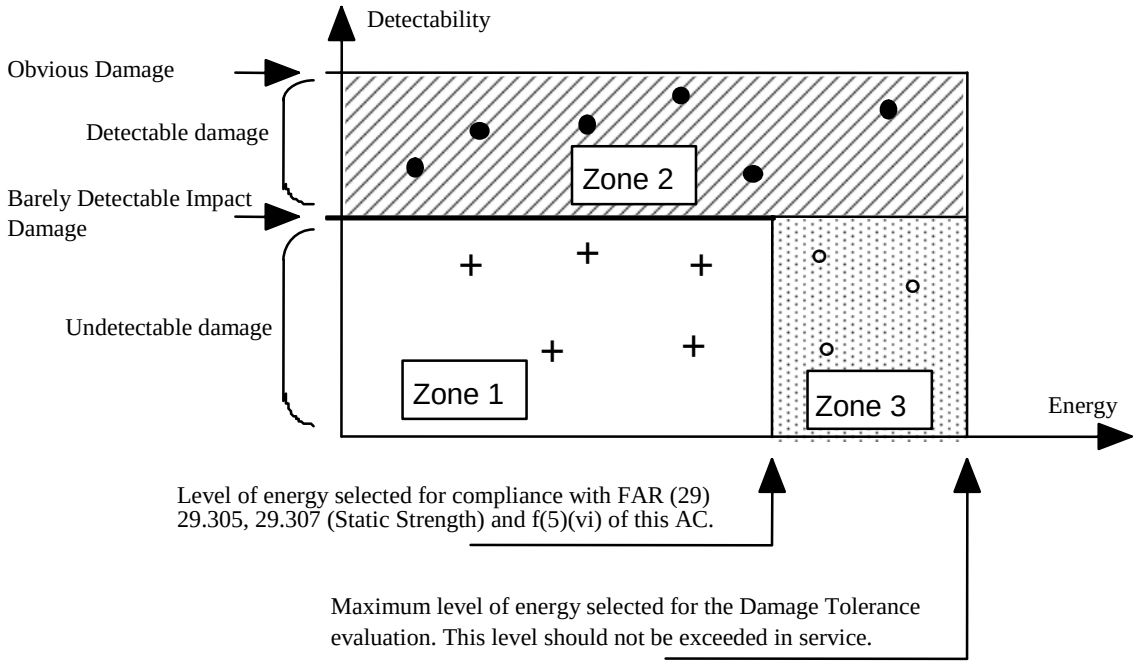

Figure AC 27.573-2. Characterization of Impact Damage

(1) Both the energy level associated with the static strength demonstration and the maximum energy level associated with the damage tolerance evaluation (depicted in Figure AC 27.573-2) are dependent on the part of the structure under evaluation and a threat assessment.

(2) Obvious impact damage is used to define the threshold from which damage is readily detectable and appropriate actions may be taken before the next flight.

(3) Barely Detectable Impact Damage is the state of damage at the threshold of detectability for the approved inspection procedure. Barely Visible Impact Damage (BVID) is that threshold of visually detectable damage associated with a detailed visual inspection procedure.

(4) Detectable Damage is the state of damage that can be reliably detected at scheduled inspection intervals. Visible Impact Damage (VID) is that threshold associated with the type of damage that should be detectable during a detailed visual inspection.

(5) Three Zones are depicted by this figure:

Zone 1: Since the damage is not detectable, Ultimate Load capability is required. The provisions of paragraph f(5) provide a means of compliance.

Zone 2: Since the damage can be detected at a scheduled inspection, Limit Load (considered as Ultimate load) capability is the minimum requirement for this damage.

Zone 3: Since the damage is not detectable with the proposed in-service inspection procedures, ultimate load capability is required, unless an alternate procedure can show an equivalent level of safety. For example, residual strength lower than ultimate may be used in association with improved inspection procedures or with a probabilistic approach showing that the occurrence of energy levels is low enough so that an acceptable level of safety can be achieved.

Of the three zones, only Zone 3 may have a residual strength requirement that can vary with alternate procedures or the probability of damage occurrence or both. In either case, any compromise for residual strength requirements less than the ultimate load requirement should only be considered when pursuing one of the options under the damage tolerant fail-safe means of compliance, as described in the following section, f(6)(iii)(B).

One example of the use of alternate procedures is for the rare damage threat from a high energy, blunt impact (e.g., service vehicle collision). Depending on the selected maintenance inspection scheme, such damage may fall under Zone 3. When considering such damage in the design of a part, it may be shown to be damage

tolerant fail safe, even though the damage is not detectable, based on a very low probability of occurrence. As a result, the design may have sufficiently high residual strength (e.g., below Ultimate, but well above limit load capability to ensure safety without detection for long periods of time). If it is further determined that such impact events usually occur with the knowledge of maintenance or aircraft service personnel, then the alternate procedures may be added to the Instructions for Continued Airworthiness. For example, advanced inspection methods, which can detect damage from high-energy blunt impacts, may be used as alternate procedures to minimize the risk of catastrophic failure for such Zone 3 damage.

(iii) MEANS OF COMPLIANCE - For each PSE, inspections, replacement times, or other procedures must be established as necessary to avoid catastrophic failure. Compliance with the requirements of § 27.573(e) and (f) should be shown by one, or a combination of, the methods described subsequently. Generally, replacement times are established using Damage Tolerance Safe Life Evaluations and Inspection Intervals are established using Fail Safe Evaluations. From current state-of-the-art rotorcraft applications, it is widely accepted that composite materials have good flaw and damage tolerance capabilities and therefore the supplemental procedures may only be rarely necessary. Damage tolerance evaluations are best suited for composite structures, particularly those with structural redundancy and inherent resistance to damage growth. Damage resulting from anomalous or accidental events must be considered in the damage tolerant evaluations. The damage tolerant evaluation for replacement times and inspection intervals is to be used unless shown that neither can be achieved within the limitations of geometry, inspectability, or good design practice. In that case, supplemental procedures must be established and submitted to the FAA/AUTHORITY for approval. In any case, the FAA/AUTHORITY must approve the methodology used for compliance to § 27.573.

The substantiation method(s) should be chosen so that the structure is protected against catastrophic failure from each of the threats identified in paragraph f(6)(ii)(C) of this AC by a specific procedure (inspection, replacement time, or other procedure). For example, a manufacturing-related void of a specific allowable size could be substantiated by means of a replacement time method with no scheduled inspection. An accidental impact in the same area could be substantiated by an inspection method with no specific replacement time. The result could be one structure with several different inspection requirements (location, method, and interval) and a fixed replacement time as well. This combination of procedures assures that each threat is covered.

The fatigue substantiation should include sufficient coupon, element, sub-element, or component tests to establish the fatigue scatter, curve shapes, and the environmental effects. The substantiation should include full-scale, component, or sub-component fatigue testing but also may be accomplished by analysis supported by test evidence. When spectrum testing is used, the lowest load levels can be eliminated from the spectrum if they can be shown to be non-damaging. The substantiation should include a static strength evaluation to show that the required residual strength and adequate

stiffness, accounting for the effects of environment, are retained for the life of the structure or the appropriate inspection interval. Damage as determined in paragraph f(6)(ii) of this AC for the specific structure being substantiated should be imposed at each critical area of the structure.

(A) Damage Tolerant Safe-Life Evaluation. This is a "No-Growth" method in which the structure, with damage present, is able to withstand repeated loads of variable magnitude without detectable damage growth for the life of the rotorcraft or within a specified replacement time. This evaluation may be used to substantiate any type of damage that will remain in-service for the life of the part.

No specific inspection requirements are generated from the test program in this method. However, compliance with routine inspections for cracking, delaminations, and service damage and other limitations prescribed in accordance with § 27.1529 are always required. Compliance using full-scale, component, or sub-component fatigue testing can be accomplished by either of the following methods:

(1) S-N Method. This method is based on determining the point where initiation of growth occurs for the damage present at critical locations in the structure. AC 27-1B, Change 1, AC 27 MG-11, provides guidance that may be appropriate for this method. The method utilizes one or more full-scale, component, or sub-component test specimens subjected to constant-amplitude or spectrum loading applied in a distribution on the structure that is representative of critical flight conditions. Any indication of growth of the imposed damage and defects, or structurally significant cracking, disbonding, splintering, or delaminating of the composite, defines the fatigue initiation characteristic of the structure in terms of applied load and cycles. Working S-N curves are established from the mean curve using strength or cycle reductions or both to account for fatigue scatter and environmental effects. Flight loads are compared to this working curve, and if any intercepts occur, a cumulative damage calculation is conducted to establish the component retirement time. Compliance with the ultimate load requirements should be demonstrated at the completion of the fatigue test.

(2) Life-Test Method. This method uses spectrum fatigue testing to verify the absence of damage growth over a large number of cycles that are equivalent to a lifetime of expected usage. The method uses one or more full-scale, component, or sub-component test specimens subjected to spectrum fatigue loading applied in a representative distribution of flight loads, including Ground-Air-Ground (GAG) loads. Fatigue test loads should be increased by factors for environment and fatigue strength scatter. The load may also be increased using an S-N curve approach to reduce the duration of the test. Any significant growth of the imposed damage, or structurally significant cracking, disbonding, splintering, or delamination of the composite during the test constitutes failure to achieve the desired lifetime. However, the equivalent life demonstrated at the time of inception of damage growth or cracking can be used as a retirement time for the component. Compliance with the ultimate load requirements should be demonstrated at the completion of the fatigue test.

(B) Damage Tolerant Fail-Safe (Residual Strength with Detectable Damage) Evaluation. This method establishes inspection intervals to ensure that the structure remaining after a partial failure is able to withstand design limit loads without failure or excessive structural deformations within a specified inspection interval. If the damage is detected in an inspection, the structure should be either replaced or repaired to restore ultimate load capability. Evaluation of Zone 3 damage should have sufficiently high residual strength and, if necessary, supplemental procedures should be established to minimize the risk of catastrophic failure. Full-scale, component, or sub-component testing should be accomplished using one or more specimens subjected to constant amplitude or spectrum loading applied in a manner representative of flight load conditions. The test loads should be increased by factors that account for environment and fatigue strength scatter. The results of the testing can be used to manage the structure in one or a combination of the three methods described subsequently.

(1) No Growth Evaluation. This approach is appropriate for inspectable in-service damage which does not grow in service. See Figure AC 27.573-3. (Damage growth should be substantiated using either method described in f(6)(iii)(B)(2) or f(6)(iii)(B)(3)). Structural details, elements, sub-components, and components of critical structural areas, or full-scale structures, should be tested under repeated loads for validating a no-growth approach to the damage tolerance requirements. The number of cycles applied to validate a no-growth concept should be statistically significant, and may be determined by load or life considerations or both. Residual strength testing or evaluations should be performed after repeated load cycling demonstrating that the residual strength of the structure is equal to or greater than limit load considered as ultimate. Moreover, it should be shown that stiffness properties have not changed beyond acceptable levels. Inspection intervals should be established, considering the residual strength capability associated with the assumed damage. The intent of this is to assure that structure is not exposed to an excessive period of time with static margins less than ultimate, providing a lower safety level than in the typical slow growth situation, as illustrated by the Figure AC 27.573-3. Once the damage is detected, the component is either repaired to restore ultimate load capability or replaced.

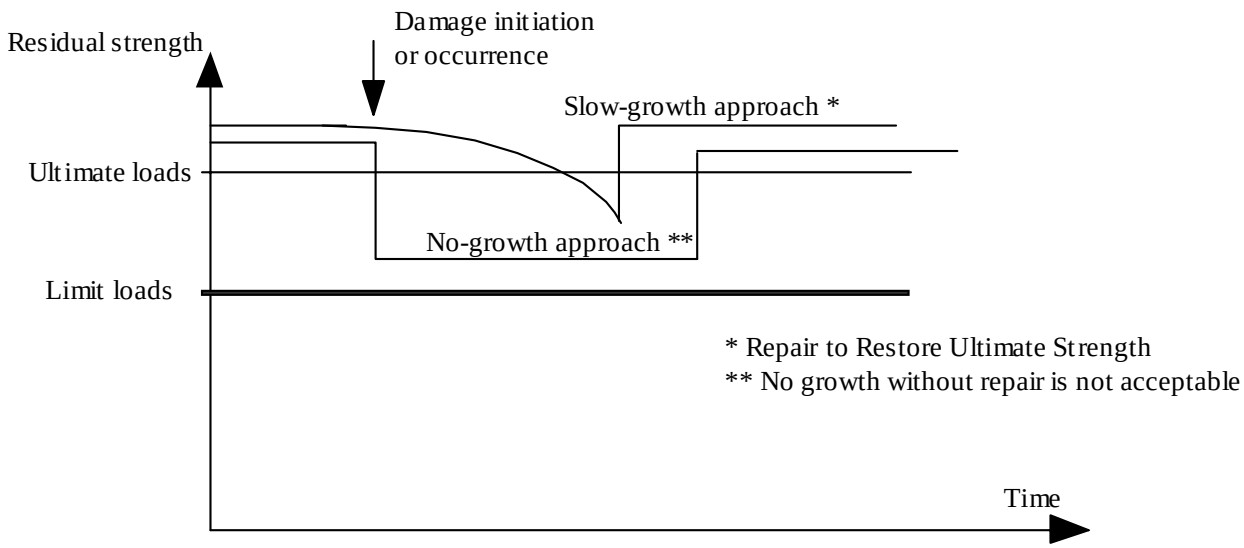

Figure AC 27.573-3. Residual Strength vs. Time

The lower the residual strength of a structure after an accidental damage event, the shorter the inspection interval should be. Considerations of both inspectability and impact surveys (including probability of occurrence) for specific structure may be used to isolate the most critical threats to consider in setting a maintenance inspection interval. Knowledge of the residual strength for a given critical damage is also needed for such an evaluation. If it is known that the design is capable of handling large and clearly detectable damage, while maintaining a residual strength well above limit load, a less rigorous engineering approach may be applied in establishing the inspection interval.

(2) Slow Growth Evaluation. This method is applicable when the damage grows in the test and the growth rate is shown to be slow, stable, and predictable, as illustrated in Figure AC 27.573-4. An inspection program should be developed consisting of the frequency, extent, and methods of inspection for inclusion in the maintenance plan. Inspection intervals should be established so that the damage will have a very high probability of detection between the time it becomes initially inspectable and the time at which the extent of the damage reduces the residual static strength to limit load (considered as ultimate), including the effects of environment. For any damage size that reduces the load capability below ultimate, the component is either repaired to restore ultimate load capability or replaced. Should functional impairment (such as unacceptable loss of stiffness) occur before the damage becomes otherwise critical, this should be accounted for in the development of the inspection program.

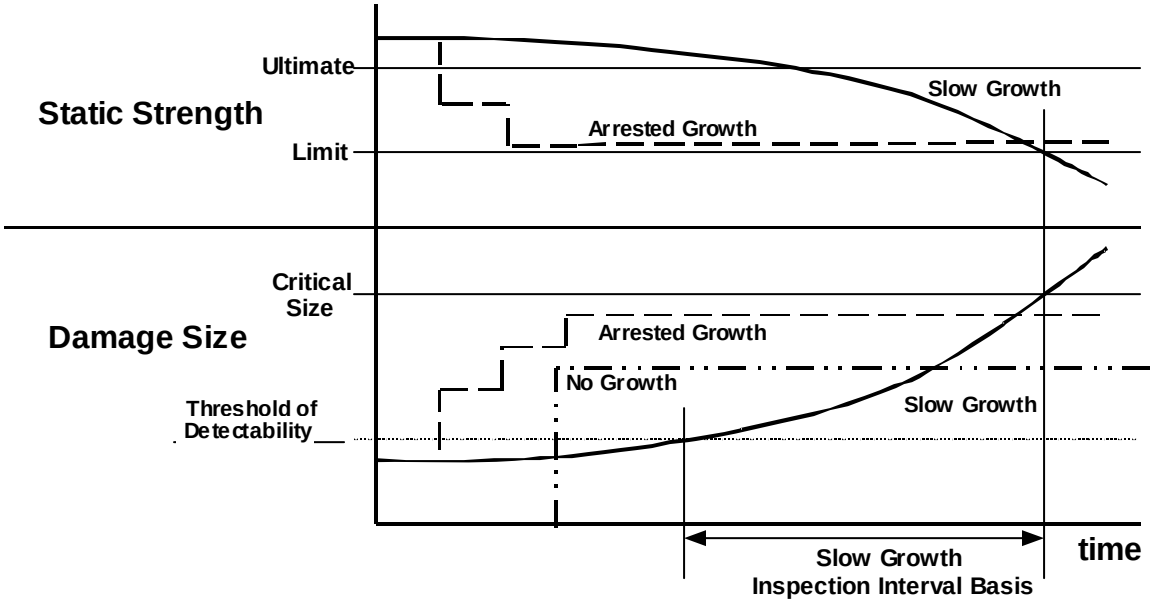

Figure AC 27.573-4. Illustration of Residual Strength and Damage Size Relationships for Fail-Safe Substantiation

(3) Arrested Growth Evaluation. This method is applicable when the damage grows, but the growth is mechanically arrested or terminated before becoming critical (residual static strength reduced to limit load), as illustrated in Figure AC 27.573-4. Arrested Growth may occur due to design features such as a geometry change, reinforcement, thickness change, or a structural joint. This approach is appropriate for inspectable arrested growth damage. Structural details, elements, and sub-components of critical structural areas, or full-scale structures, should be tested under repeated loads for validating an arrested growth approach to the flaw tolerance requirements. The number of cycles applied to validate an arrested growth concept should be statistically significant, and may be determined by load or life considerations or both. Residual strength testing or evaluation should be performed after repeated load cycling and a demonstration that the residual strength of the structure is equal to or greater than limit load considered as ultimate. Moreover, it should be shown that stiffness properties have not changed beyond acceptable levels. Inspection intervals should be established, considering the residual strength capability associated with the arrested growth damage. The intent of this is to assure that structure is not exposed to an excessive period of time with static margins less than ultimate, providing a lower safety level than in the typical slow growth situation, as illustrated by Figure AC 27.573-3. For any damage size that reduces the load capability below ultimate, the component is either repaired to restore ultimate load capability or replaced.

The lower the residual strength of a structure after an arrested growth event, the shorter the inspection interval should be. Considerations of both inspectability and impact surveys (including probability of occurrence) for specific structure may be used to isolate the most critical threats to consider in setting a maintenance inspection interval.

Knowledge of the residual strength for a given critical damage is also needed for such an evaluation. If it is known that the design is capable of handling large and clearly detectable damage, while maintaining a residual strength well above limit load, a less rigorous engineering approach may be applied in establishing the inspection interval.

(C) Combination of Damage Tolerant Safe Life and Fail Safe Evaluations. Generally, it may be appropriate to establish both a replacement time and an inspection program for a given structure as calculated by the Damage Tolerant Safe Life and Fail Safe Evaluations.

(D) Other Procedures. Other procedures are allowed according to paragraph 27.573(d). Such alternative procedures must still provide the same degree of damage tolerance to the same identified threats as the replacement time or inspection interval methods.

One possible alternate approach is the use of indirect damage detection methods instead of the specific mandated inspection procedures that are determined in the Fail Safe Evaluations of f(6)(iii)(B). These indirect detection methods should be documented and shown to have the same degree of reliability, repeatability, and margin provided by a conventional inspection approach. These methods could include: (1) establishing measurable vibration or blade out-of-track conditions and limits, (2) defining indirect inspections which would detect damage, and (3) in-flight detecting of damage by means of monitoring and warning devices.

(E) Supplemental Procedures. If the damage tolerant evaluations as described previously cannot be achieved within the limitations of geometry, inspectability, or good design practice, a fatigue evaluation using supplemental procedures may be proposed to the certifying authority per § 27.573(h). A rationale must be provided as to why the damage tolerance criteria cannot be satisfied for the specific PSE, locations, and threats considered. In addition, the types of damage considered in the evaluations must be identified. Finally, supplemental procedures must be established to minimize the risk of catastrophic failure with the damages considered.

(iv) Additional Considerations for DAMAGE TOLERANCE AND FATIGUE Evaluations.

(A) Experience with the application of methods of fatigue and damage tolerance evaluations indicates that a relevant test background should exist in order to achieve the design objective. It is the general practice within industry to conduct damage tolerance tests for design information and guidance purposes. It is crucial that the critical structure be identified and tested to the proper flight and ground loads. In the fatigue and flaw tolerance evaluation, the following items must be considered:

(B) Identification of the structure to be considered in each evaluation (a failure mode and effects analysis or similar method should be used).

(1) Identification of Principal Structural Elements. Principal structural elements are those that contribute significantly to carrying flight and ground loads and whose failure could result in catastrophic failure of the rotorcraft. Typical examples of such elements are:

(i) Rotor blades and attachment fittings.

(ii) Rotor heads, including hubs, hinges, and some main rotor dampers.

(iii) Control system components subject to repeated loading, including control rods, servo structure, and swashplates.

(iv) Rotor supporting structure (lift path from airframe to rotorhead).

(v) Fuselage, including stabilizers and auxiliary lifting surfaces, airframe provisions for engine and transmission mountings.

(vi) Main fixed or retractable landing gear and fuselage attachment structure.

(2) Identification of Locations Within Principal Structural Elements to be Evaluated. The locations of damage to structure for damage tolerance evaluation can be determined by analysis or by fatigue test on complete structures or subcomponents. However, tests will 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 following should be considered:

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

(ii) Locations where analysis shows high stress or low margins of safety;

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

(iv) Locations of potential fatigue damage identified by fatigue analysis;

(v) Locations where the stresses in adjacent elements will be at a maximum with an element in the location failed;

(vi) Partial fracture locations in an element where high stress concentrations are present in the residual structure;

(vii) Locations where detection would be difficult;

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

(3) In addition, the areas of probable damage from sources such as a severe corrosive or fretting environment, a wear or galling environment, or a high maintenance environment should be determined from a review of the design and past service experience.

(C) The stresses and strains (steady and oscillatory) associated with all representative steady and maneuvering operating conditions expected in service.

(D) The frequency of occurrences of various flight conditions and the corresponding spectrum of loadings and stresses.

(E) The fatigue strength, fatigue crack propagation characteristics of the materials used and of the structure, and the residual strength of the damaged structure.

(F) Inspectability, inspection methods, and detectable flaw sizes.

(G) Variability of the measured stresses of paragraph f(6)(iv)(C), the actual flight condition occurrences of paragraph f(6)(iv)(D), and the fatigue strength material properties of paragraph f(6)(iv)(E).

(v) FLIGHT STRAIN MEASUREMENT PROGRAM.

(A) General. Subsequent to design analysis, in which aircraft loads and associated stresses are derived, the stress level or loads are to be verified by a carefully controlled flight strain measurement program. (This guidance is similar to that of AC 27-1B, MG 11.)

(B) Instrumentation.

(1) The instrumentation system used in the flight strain measurement program should accurately measure and record the critical strains under test conditions associated with normal operation and specific maneuvers. The location

and distribution of the strain gauges should be based on a rational evaluation of the critical stress areas. This may be accomplished by appropriate analytical means supplemented, when deemed necessary, by strain sensitive coatings or photoelastic methods. The distribution and number of strain gauges should cover the load spectrum adequately for each part essential to the safe operation of the rotorcraft as identified in § 27.573(c)(1). Other devices such as accelerometers may be used as appropriate.

(2) The corresponding flight parameters (airspeed, rotor RPM, center-of-gravity accelerations, etc.) should also be recorded simultaneously by appropriate methods. This is necessary to correlate the loads and stresses with the maneuver or operating conditions at which they occurred.

(3) The instrumentation system should be adequately calibrated and checked periodically throughout the flight strain measurement program to ensure consistent and accurate results.

(C) Parts to be Strain-Gauged. Fatigue critical portions of the rotor systems, control systems, landing gear, fuselage, and supporting structure for rotors, transmissions, and engine are to be strain-gauged. For rotorcraft of unusual or unique design, special consideration might be necessary to ensure that all the essential parts are evaluated.

(D) Flight Regimes and Conditions to be investigated.

(1) Typical flight and ground conditions to be investigated in the flight strain measurement program are given in paragraphs c. and d. of AC 27-1B, MG 11.

(2) The determination of flight conditions to be investigated in the flight strain measurement program should be based on the anticipated use of the rotorcraft and, if available, on past service records for similar designs. In any event, the flight conditions considered appropriate for the design and application should be representative of the actual operation in accordance with the rotorcraft flight manual. In the case of multiengine rotorcraft, the flight conditions concerning partial engine-out operation should be considered in addition to complete power-off operation. The flight conditions to be investigated should be submitted in connection with the flight evaluation program.

(3) The severity of the maneuvers investigated during the flight strain survey should be at least as severe as the maneuvers likely to occur in service.

(4) All flight conditions considered appropriate for the particular design are to be investigated over the complete rotor speed, airspeed, center of gravity, altitude, and weight ranges to determine the most critical stress levels associated with each flight condition. The temperature effects on loads as affected by

elastomeric components are to be investigated. To account for data scatter and to determine the stress levels present, a sufficient amount of data points should be obtained at each flight condition. Consideration can be given to the use of scatter factors in determining the sufficiency of data points. In some instances, the critical weight, center of gravity, and altitude ranges for the various maneuvers can be based on past experience with similar design. This procedure is acceptable where adequate flight tests are performed to substantiate such selections. The combinations of flight parameters that produce the most critical stress levels should be used in the evaluation.

(vi) FREQUENCY OF LOADING.

(A) Types of Operation.

(1) The probable types of operation (transport, utility, etc.) for the rotorcraft should be established. The type of operation can have a major influence on the loading environment. In the past, rotorcrafts have been substantiated for the most critical general types of operation with some consideration of special, occasional types of operation. To assure that the most critical types of operation are considered, each major rotorcraft structural component should be substantiated for the most critical types of operation as established by the manufacturer. The types of operation shown below should be considered and, if applicable, used in the substantiation:

(i) Long flights to remote sites (low ground-air-ground cycles but high cruising speeds).

(ii) Typical, general types of operation.

(iii) Short flights as used in logging operations.

(2) One means is to substantiate for the most severe type of operation; however, this method is not always economically feasible.

(3) A second means is to quantify the influence of mission type on fatigue damage by adding to or replacing hour limitations by flight cycle limitations (if properly defined and easily identifiable by the crew, for example: one landing, one load transportation). A special type of flight hour limitation replacement using factorization of flight hours for multiple types of operations may be feasible if continuing manufacturers' technical support is provided and documented; i.e., the manufacturer either provides the factorization analyses or checks them on a continuing basis for each type of rotorcraft operation.

(4) Where one or more operations are not among the general uses intended for the rotorcraft, the rotorcraft flight manual should state in the limitations section that the intended use of the rotorcraft does not include certain missions or repeated maneuvers (i.e., logging with its high number of takeoffs and

landings per hour). A note to this effect should also appear in the rotorcraft airworthiness limitations section of the maintenance manual prepared in accordance with §§ 27.573 and 27.1529.

(5) Should subsequent usage of the rotorcraft encompass a mission for which the original structural substantiation did not account, the effects of this new mission environment on the frequency of loading and structural substantiation should be addressed and where practicable, in the interest of safety, a reassessment made. If this reassessment indicates the necessity for revised retirement times, those new times may be limited to aircraft involved in the added mission provided:

(i) Changes are adopted through the airworthiness directives process, and;

(ii) Proper part re-identification is established;

(iii) A Rotorcraft Flight Manual (RFM) supplement outlining the limitations is approved;

(iv) An airworthiness limitations section (ALS) supplement is approved; or

(v) An appropriate combination of part re-identification, RFM supplement, or airworthiness limitation section supplement is approved.

(B) Loading Spectrum. The spectrum allocating percentage of time or frequencies of occurrence to flight conditions or maneuvers is to be based on the expected usage of the rotorcraft. This spectrum is to be established so that it is unlikely that actual usage will subject the structure to damage beyond that associated with the spectrum. Considerations to be included in developing this spectrum should include prior knowledge based on flight history recorder data, design limitations established in compliance with § 27.309, and recommended operating conditions and limitations specified in the rotorcraft flight manual or instructions for continued airworthiness (ICA). The distribution of times at various forward flight speeds should reflect not only the relation of these speeds to V_{NE} but also the recommended operating conditions in the rotorcraft flight manual or ICA that govern V_c or cruise speed. It is desirable to conduct the flight strain-gauge program by simulating the usage as determined previously, with continuous recording of stresses and loads, thus obtaining directly the stress or load spectra for structural elements.

(7) The seventh major area is the dynamic loading and response requirements of § 27.241, § 27.251, and § 27.629 for vibration and resonance frequency determination and separation for aeroelastic stability and stability margin determination for dynamically critical flight structure. Critical parts, locations, excitation modes, and separations should be identified and substantiated. This substantiation

should consist of analysis supported by tests and tests that account for repeated loading effects and environment exposure effects on critical properties, such as stiffness, mass, and damping. This must be accomplished to assure that the initial stiffness, residual stiffness, proper critical frequency design, and structural damping are provided as necessary to prevent vibration, resonance, and flutter problems.

(A) All vibration and resonance critical composite PSE must be identified and properly evaluated.

(B) All flutter-critical composite PSE must be identified and properly evaluated. This structure must be shown by analysis to be flutter free to 1.1 V_{NE} (or any other critical operating limit, such as V_D, for a VSTOL aircraft) with the extent of damage for which residual strength and stiffness are demonstrated.

(C) Where appropriate, crash impact dynamics considerations should be taken into account to ensure proper crash resistance and a proper level of occupant safety for an otherwise survivable impact.

(8) The eighth area is the special repair and continued airworthiness requirements of §§ 27.611, 27.1529, and 14 CFR part 27 Appendix A, for composite structures. When repair and continued airworthiness procedures are provided in service documents (including approved sections of the maintenance manual or instructions for continued airworthiness), the resulting repairs and maintenance provisions should be shown to provide structure which continually meets the guidance of paragraphs (1) through (7) of this AC. All certification-based repair and continued airworthiness standards, limits, and inspections must be clearly stated and their provisions and limitations clearly documented to ensure continued airworthiness. No composite structural repair should be attempted that is beyond the scope of the applicable approved Structural Repair Manual (SRM) without an engineering design approval by a qualified FAA/AUTHORITY designated representative.

File:O:Rule/Draft Rules/Composites/AC 27573 DT Composites SM Comments.doc:July2007
ASW-7 cmts incorporated 10/17/07

[4910-13]

DEPARTMENT OF TRANSPORTATION

Federal Aviation Administration

14 CFR Parts 27 and 29

[Docket No. FAA-YYYY- ; Notice No.]

RIN 2120- [If a RIN has been assigned to the project it can be found in the Semiannual Regulatory Agenda. However, if a RIN has not been assigned, the ARM analyst obtains the RIN from AGC–200 6 months prior to publication.]

Title: Damage Tolerance and Fatigue Evaluation of Composite Rotorcraft Structure

AGENCY: Federal Aviation Administration (FAA), DOT.

ACTION: Notice of proposed rulemaking (NPRM).

SUMMARY: This document proposes to add type certification requirements for both normal and transport category rotorcraft. This proposal would add airworthiness standards to evaluate the fatigue strength and residual static strength of composite rotorcraft structure using a damage tolerance evaluation and if that is impractical, a fatigue evaluation. The current regulations written for metallic structure do not provide adequate certification standards for composite materials and structures. This proposal would add safety requirements for composite rotorcraft structure to address advances in composite structures technology. Further, this proposal would increase safety by providing internationally harmonized standards for use in evaluating the fatigue strength of composite rotorcraft structure.

DATES: Send your comments on or before [Insert date 90 days after date of publication in the Federal Register.]

ADDRESSES: You may send comments [identified by Docket Number *[***Washington DC inserts this)** *Insert docket number, for example, FAA-200X-XXXXX]*] using any of the following methods:

- DOT Docket web site: Go to http://dms.dot.gov and follow the instructions for sending your comments electronically.
- Government-wide rulemaking web site: Go to http://www.regulations.gov and follow the instructions for sending your comments electronically.
- Mail: Docket Management Facility; US Department of Transportation, 400 Seventh Street, S.W., Nassif Building, Room PL-401, Washington, DC 20590.
- Fax: 1-202-493-2251.
- Hand Delivery: Room PL-401 on the plaza level of the Nassif Building, 400 Seventh Street, S.W., Washington, DC, between 9 a.m. and 5 p.m., Monday through Friday, except Federal holidays.

For more information on the rulemaking process, see the SUPPLEMENTARY INFORMATION section of this document.

Privacy*:* We will post all comments we receive, without change, to http://dms.dot.gov, including any personal information you provide. For more information, see the Privacy Act discussion in the SUPPLEMENTARY INFORMATION section of this document.

Docket*:* To read background documents or comments received, go to http://dms.dot.gov at any time or to Room PL-401 on the plaza level of the Nassif Building, 400 Seventh Street, S.W., Washington, DC, between 9 a.m. and 5 p.m.,

Monday through Friday, except Federal holidays.

FOR FURTHER INFORMATION CONTACT: Charles C. Harrison, Safety Management Group, Rotorcraft Directorate, ASW-112, Federal Aviation Administration, Fort Worth, Texas 76193-0112; telephone (817) 222-5128; facsimile (817) 222-5961; e-mail charles.c.harrison@ faa.gov.

SUPPLEMENTARY INFORMATION:

Comments Invited

The FAA invites interested persons to take part in this rulemaking by sending written comments, data, or views. We also invite comments pertaining to the economic, environmental, energy, or federalism impacts that might result from adopting the proposals in this document. The most helpful comments will reference a specific portion of the proposal, explain the reason for any recommended change, and include supporting data. We ask that you send us two copies of written comments.

We will file in the docket all comments we receive, as well as a report summarizing each substantive public contact with FAA personnel concerning this proposed rulemaking. The docket is available for public inspection before and after the comment closing date. If you wish to review the docket in person, go to the address listed in the ADDRESSES section of this document between 9:00 a.m. and 5:00 p.m., Monday through Friday, except Federal holidays. You may also review the docket using the Internet at the web address in the ADDRESSES section.

Privacy Act*:* Using the search function of our docket web site, anyone can find and read the comments received into any of our dockets, including the name of the individual sending the comment (or signing the comment on behalf of an association, business,

labor union, etc.). You may review DOT's complete Privacy Act Statement in the Federal Register published on April 11, 2000 (65 FR 19477-78), or you may visit http://dms.dot.gov.

Before acting on this proposal, we will consider all comments we receive on or before the closing date for comments. We will consider comments filed late if it is possible to do so without incurring additional expense or delay. We may change these proposals based on the comments we receive.

If you want the FAA to acknowledge receipt of your mailed comments on this proposal, include with your comments a preaddressed, stamped postcard on which the docket number appears. We will stamp the date on the postcard and mail it to you.

Availability of Rulemaking Documents

You can get an electronic copy using the Internet by:

(1) Searching the Department of Transportation's electronic Docket Management System (DMS) web page at http://dms.dot.gov/search;

(2) Visiting the Office of Rulemaking's web page at http://www.faa.gov/avr/arm/index.cfm; or

(3) Accessing the Government Printing Office's web page at http://www.gpoaccess.gov/fr/index.html.

You can also get a copy by sending a request to the Federal Aviation Administration, Office of Rulemaking, ARM-1, 800 Independence Avenue S.W, Washington, DC 20591, or by calling (202) 267-9680. Make sure to identify the docket number, notice number, or amendment number of this rulemaking.

Authority for this Rulemaking

The FAA's authority to issue rules regarding aviation safety is found in Title 49 of the United States code. Subtitle I, Section 106 describes the authority of the FAA Administrator. Subtitle VII, Aviation Programs, describes in more detail the scope of the agency's authority.

This rulemaking is promulgated under the authority described in Subtitle VII, Part A, Subpart III, Section 44701, "General requirements," Section 44702, "Issuance of Certificates," and Section 44704, "Type Certificates, production certificates, and airworthiness certificates." Under Section 44701, the FAA is charged with prescribing regulations and minimum standards for practices, methods, and procedures the Administrator finds necessary for safety in air commerce. Under Section 44702, the FAA may issue various certificates including type certificates, production certificates, air agency certificates, and airworthiness certificates. Under Section 44704, the FAA shall issue type certificates for aircraft, aircraft engines, propellers, and specified appliances when the FAA finds the product is properly designed and manufactured, performs properly, and meets the regulations and minimum prescribed standards. This regulation is within the scope of these authorities because it would promote safety by updating the existing minimum prescribed standards, used during the type certification process, to address advances in composite structural fatigue substantiation technology. It would also harmonize this standard with international standards for evaluating the fatigue strength of normal and transport category composite rotorcraft primary structural elements.

Background

Statement of the Problem

The current regulations do not provide adequate certification standards for composite materials and structures. Certification has been based on a broad interpretation of the metallic fatigue substantiation and the design and construction airworthiness standards. The Type Certification solutions sought for composite and metal structures are different. As a result, special conditions were necessary for some certification projects using composite structures. The increased use of composites in rotorcraft structures, certification, and service experiences for composite rotorcraft structure over the last twenty-five years has suggested a need to reconsider the current regulations for damage tolerance and fatigue evaluation. To this end, an Aviation Rulemaking Advisory Committee (ARAC) working group (WG) was chartered to study the problems and offer recommendations for rotorcraft composite structure minimum safety standards.

History

The international ARAC WG that assembled in 2000 consisted of industry practitioners, regulatory officials, and technical specialists, each with unique experiences with composite applications. The two co-chairs, one from a U.S. manufacturer and the other from a European manufacturer, each had many years of rotorcraft industry experience with composite structures. The team included members familiar with the different rotorcraft structures, including airframe, main and tail rotor drive systems, main and tail rotor blades and hubs, rotor controls, fixed and movable control surfaces, and transmission mountings. Both Part 27 (normal) and 29 (transport) category rotorcraft manufacturers had representatives on the team. Such a balanced

team was needed to develop recommendations that consider the unique technical issues and design characteristics for composites used in the different rotorcraft structures.

The ARAC WG considered alternatives and determined that rulemaking action should be recommended. It felt that taking no action would result in continued confusion among and within manufacturers and authorities. Relying on advisory material only, as in the past, did not always remove the need for special conditions, and such material was not enforceable. The only existing rule for damage tolerance and fatigue evaluation of structure that has special considerations for composite materials is paragraph (a) of § 23.573. Much of this Part 23 rule has details for composite airframe structures subjected to relatively low numbers of fatigue cycles. Because some critical rotorcraft structures experience a high number of fatigue cycles, the ARAC WG felt the Part 23 airworthiness standards were not acceptable for the complete range of composite rotorcraft structures.

The ARAC WG's effort resulted in recommendations for new regulatory standards and standardized means of compliance that allows for innovation and changes in technology. To carry out the recommendations, the WG proposed regulations, Sections 27.573 and 29.573 of Title 14, Code of Federal Regulations (14 C.F.R.), specific to composite rotorcraft structure. The WG concurrently developed advisory material to support the proposed regulations.

The ARAC WG developed a 2-year plan ending in 2002. The plan included drafting a Notice of Proposed Rulemaking (NPRM) for each proposed rule and an Advisory Circular (AC) describing a means, but not the only means, of compliance for

each rule. The WG used the following approach. First, the WG discussed rotorcraft industry applications and service histories, as well as the related certification experiences in composite damage tolerance and fatigue evaluation. Next, the WG members developed and agreed to an approach for the new composite rule. At the same time, the WG updated the composite rotorcraft structures AC for the existing rule to represent relevant WG experiences. Finally, the WG drafted the new rule proposal and a new AC.

In reviewing experiences with composite rotorcraft structures, it became clear that different safety concerns existed depending on the specific rotorcraft application. Many of the differences related to fatigue and accidental damage threats, design detail, and critical load cases. For example, rotor blades, which are dynamically loaded by predominately centrifugal forces, are different from an airframe structure with significant compressive loads. Because of this variability, the ARAC WG developed a general rule applicable to all types of composite rotorcraft structures.

The WG began processing information by identifying the relevant technical issues and concerns, and narrowing the scope of the effort as required. Of special note, the WG recognized the safety concerns about the sensitivity of composite structure to defects and service damage. As a result, the WG recommended the new §§ 27.573 and 29.573 allow only a composite structure damage tolerance evaluation or a fatigue evaluation. Conventional (unflawed) safe-life evaluations as applied to metallic structure were not considered an option for composites.

Several other key technical issues were important points of discussion for the WG. A thorough damage threat assessment was believed crucial to defining the

damage tolerance and fatigue evaluation for each principal structural element (PSE) of the rotorcraft. This was particularly of concern for composite structure where the primary threats relate to accidental damage (for example, impact damage) and anomalous manufacturing flaws. The WG also believed the rule needed to be clearly written to describe the standards for inspection intervals and replacement times. In addition, the WG recognized the need to provide applicants some freedom in their approach to meeting the standards for performing a damage tolerance or fatigue evaluation on a given PSE. Therefore, based on the WG review of the referenced material and industry experiences in both the United States and Europe, the WG decided this rulemaking was needed to increase safety standards offered by advances in technology and to standardize composite rotorcraft structure certification.

Reference Material

The ARAC WG used the following material in developing their proposals. You may find copies in Rules Docket No. XXXXX.

1. 14 CFR 27.571, Amdt. 27-26, March 6, 1990.
2. 14 CFR 29.571, Amdt. 29-28, October 27, 1989.
3. AC 20-95, “Fatigue Evaluation of Rotorcraft Structure,” May 18, 1976, and AC 27-1B MG-11, “Fatigue Evaluation of Rotorcraft Structure”, February 12, 2003.
4. AC 20-107A, “Composite Aircraft Structure”, April 25, 1984.
5. AC 21-26, “Quality Control for the Manufacture of Composite Materials”, June 26, 1989.
6. AC 29-2C, “Certification of Transport Category Rotorcraft,” paragraphs AC 29

MG 8, “Substantiation of Composite Rotorcraft Structure,” and AC 29 MG 11, “Fatigue Evaluation of Transport Category Rotorcraft Structure (including Flaw Tolerance),” September 30, 1999.

7. Proceedings, 42nd Mil-HDBK-17 Coordination Group Meeting, Addendum to Polymer Matrix Composites Coordination Group, “Chapter 5, Damage Resistance and Damage Tolerance-new”, 20-23 February 2001, Clearwater, Florida.

8. Rouchon, J. “Effects of Low Velocity Impact Damage on Primary Composite Aircraft Structures: The Certification Issue”, Note 07/SP1/99, dated August 24, 1999.

9. Rauch, P. and Charreyre, A. “Damage-Tolerant Tail Rotor Blade for AS 332 L2 Super Puma Helicopter”.

10. Adams, D.O., Chairperson, The Fatigue Methodology Committee of the Aerospace Industries Association “Composites Qualification Criteria”, dated 1995.

11. Dickson, B., Roesch, J., Adams, D.O., and Krasnowski, B. “Rotorcraft Fatigue and Damage Tolerance”, prepared for the Technical Oversight Group on Aging Aircraft (TOGAA), January 1999.

12. Bansemir, H., Besson, J.-M., and Pfeifer, K. “Development and Substantiation of Composite Structures with Regard to Damage Tolerance”, presented at the 27th European Rotorcraft Forum, September 11-14, 2001, Moscow, Russia.

13. Bansemir, H. and Muller, R. “The EC135 – Applied Advanced Technology”, presented at the American Helicopter Society 53rd Annual Forum, April 29-May 1, 1997, Virginia Beach, Virginia.

14. Reddy, D.J. “Qualification Program of the Composite Main Rotor Blade for the Model 214B Helicopter”, presented at the 35th Annual National Forum of the

American Helicopter Society, May 1979, Washington, D.C.

15. Guzzetti, G., Mariani, U., and Oggioni, F. "Certification of the EH-101 Composite Components: A Comprehensive Approach", presented at the American Helicopter Society 51st Annual Forum, May 9-11,1995, Fort Worth, Texas.

This material provided state-of-the-art substantiation for the latest rotorcraft composite structures manufactured in both the United States and European industries. Some airworthiness authorities have issued special rules for composite structure substantiation such as the "Special Condition for Primary Structures Designed with Composite Material" issued by the Luftfarht-Bundesamt (LBA), the Federal Republic of Germany airworthiness authority. This special condition establishes increased safety standards required by the LBA.

Special features addressed in the LBA special condition are for:

- An investigation of the structural components to determine which are suitable or unsuitable for a damage-tolerance evaluation and the related inspection procedures for those components.
- An investigation of the growth rate of damage, under repeated loads expected in-service, that may occur from fatigue, corrosion, intrinsic flaws, manufacturing defects, and discrete source impact damage.
- Consideration of the effects of material variability.
- Consideration of the effects of environmental conditions such as the degradation of strength in composite structure because of heat and humidity over time.
- Residual strength requirements including consideration of manufacturing and impact damage.

- Substantiation of bonded joints.

The WG agreed that requirements similar to these LBA requirements are necessary to help in certification and substantiation of composite rotorcraft structures.

The Need for Regulatory Change

New composite rotorcraft structure standards are needed because:

- Existing regulations, §§ 27.571 and 29.571, do not adequately describe the fatigue certification requirements for composite rotorcraft structure. The lack of regulatory requirements has led to inconsistent interpretations from one rotorcraft certification project to another, resulting in different burdens on industry to substantiate their composite rotorcraft structure. As discussed previously, some special conditions have been required to establish adequate and appropriate safety standards.

- Some applicants state the lack of a separate composite rotorcraft structure rule has resulted in confusion because neither the current rules nor the various advisory materials provide clear and complete guidance for certification of composite rotorcraft structure. New rules and AC material should alleviate this problem.

- The new rules and advisory materials will integrate and reflect the rotorcraft industry historical experiences carrying out certification of composite rotorcraft structure.

- There have been significant changes in composite technology since the current rule and the original advisory material was written. In recent years, there have been changes in design, analysis, testing, manufacturing, maintenance techniques, and maintenance procedures. The new proposed regulations and advisory materials will account for these significant changes.

Rationale for Separate Metallic and Composite Rules and Related Activity

A separate rotorcraft ARAC WG developed a fatigue tolerance rulemaking proposal for metallic structure. To ensure the two rules were consistent and compatible, several of the same people served on both WG's.

Several reasons led to separate rules for the damage tolerance and fatigue evaluation of metal and composite structures. The primary reason was a need to emphasize different aspects of the evaluation for the most critical issues for each class of materials. In composites, there are some unique material behaviors and sensitivities to different damage and loading conditions. For example, this proposal for composites stresses damage tolerance because fatigue is only one of several conditions that can reduce strength. In composites, low cycle fatigue sensitivity often yields no damage growth, whereas accidental damage from impact can immediately reduce residual strength. In metals, all critical damage types have sensitivities to cyclic fatigue loads; therefore, you will notice more emphasis on fatigue tolerance in the proposal for metallic structures.

Several other issues are emphasized in the composite rule. These include material and process variability and environmental effects. A strength requirement for ultimate loads is also applied when maximum acceptable manufacturing defects and service damage are present to provide an option for some rotorcraft structures and damage scenarios that justify an approach based on retirement times instead of inspection intervals more commonly associated with damage tolerance standards. However, if it is adequately demonstrated that certain damage does not grow or does not grow beyond a certain threshold or size and that damaged structure can still carry

ultimate loads, inspection may not be necessary and the part can be assigned a retirement life in lieu of an inspection program. Finally, this proposal for composite structure provides more clarity on certain issues such as the threat assessment, which is closely associated with service history for composite structure.

General Discussion of the Proposals

Current evolution of composite technology used in rotorcraft structure is occurring at a fast pace. This evolution has prompted these proposals. Also these rapid changes suggest a need to review periodically the associated rules and AC material for composite damage tolerance and fatigue evaluation for composite rotorcraft structure substantiation and certification. We propose to revise and improve the regulatory requirements for substantiating and certifying composite rotorcraft structure that originally were written for metallic materials. Also, we propose to clarify the standards to achieve a more consistent level of safety among applicants.

We intend that the proposal account for the unique characteristics of composite materials and that the applicants evaluate these types of materials slightly different from traditional metallic materials. Specific details on damage, types of damage, loading conditions, threat assessments, manufacturing defects, and residual strength are all included as a part of the proposal. Recognizing that it may be impractical within the limits of geometry, inspectability, or good design practice to evaluate all the composite structures of a rotorcraft using a damage tolerance evaluation, this proposal allows for a fatigue evaluation of particular rotorcraft composite structures under §§ 27.573(h) and 29.573(h), in lieu of a damage tolerance evaluation, for particular structures if you establish that an impracticality exists. If a fatigue evaluation for a particular rotorcraft

composite structure is approved, you must identify the PSEs, identify the types of damages considered, establish supplemental procedures to minimize the risk of catastrophic failure associated with those damages, and include those procedures in the Airworthiness Limitation section of the Instructions for Continued Airworthiness. Minimizing the risk of catastrophic failure is not intended to require that all risk be eliminated, only that you minimize the risk to a practical and reasonable level for each particular fatigue evaluation.

Section-by-Section Discussion of the Proposals

Proposed §§ 27.573 and 29.573

There is no difference in the requirements between the proposed § 27.573 and § 29.573 for normal and transport category type certification of composite rotorcraft structure. Unless other procedures are approved under paragraph (h), each section requires composite rotorcraft structure to be substantiated by a damage tolerance evaluation. The evaluation must show, for each of the PSE's of the structure, that catastrophic failure of such a structure will be avoided throughout the operational life of the rotorcraft. You must accomplish this by establishing component replacement times or by establishing prescribed inspection intervals that will detect any damage growth before the required residual strength is exceeded for those components.

Key Language in the New Rule

The existing § 29.571, provided a starting point for drafting a separate composite rule. The WG kept some aspects of that rule in this proposal. These included several steps in the evaluation, such as identification of the PSE's, the in-flight measurements of loads, and the loading spectra as severe as those expected in-service. The proposal

explains other steps in greater detail. The removed details related to a specific means of compliance. For example, we remove the reference to flaw tolerant safe-life and fail-safe evaluations because there are more suitable ways of describing each approach under damage tolerance. As mentioned earlier, we remove the traditional safe-life method because composites have sensitivities to defects and damage that must be considered in design and certification testing.

We wish to improve the standards for determining inspection intervals and retirement times. Inspection intervals and retirement times are important determinations from a damage tolerance and fatigue evaluation. This proposal reflects that emphasis.

Currently, the minimum residual strength requirement for any damage or defect that can be found by inspection is tied to limit loads. We propose wording to link the required residual strength to the probability of a given damage type, inspection interval, and damage detectability. This link is necessary for at least two reasons. First, one of the more critical threats--impact damage--could immediately lower residual strength well below ultimate loads if it occurs. The intent of the standard is to clarify, as the residual strength is lowered, the earlier damage needs to be detected and repaired. Operators must perform inspections on the rotorcraft to reveal any damage or defect growth to minimize the time operating the rotorcraft at less than an ultimate loads capability. Second, this proposal addresses rare damage (such as high-energy, blunt impacts) that is not detectable with the prescribed inspection schemes issued for aircraft in operational service. Although such damage may have a low probability of occurring, this proposal would require that sufficient residual strength exists to account for such damage.

This proposal would require the residual strength requirement for any damage not subjected to maintenance inspection be linked to ultimate loads. A need to establish a retirement time may occur either because the damage may not be found by inspection or the applicant may not want to burden the rotorcraft owners with repeated inspections. This proposal is worded so some damage scenarios for a given PSE may be covered by a retirement time (an ultimate load requirement). Other damage scenarios may be detected by inspection (a limit load requirement) and repaired to restore ultimate strength capability.

In this proposal, all PSE assessments for the damage threats and the residual strength and fatigue characteristics have moved to the list of requirements for inspection intervals or replacement times. In paragraph (c), we propose that you evaluate certain parts including PSE's of the airframe, main and tail rotor drive systems, main and tail rotor blades and hubs, rotor controls, fixed and movable control surfaces, engine and transmission mountings (provided by the airframe manufacturer), landing gear and other parts, detail design points, and fabrication techniques deemed critical by the FAA to avoid catastrophic failure due to static or fatigue loads. We also propose a paragraph on the need to consider the effects of damage on stiffness, dynamic behavior, loads, and functional performance. In the existing rule, such requirements are limited in scope and only appear under fail-safe evaluations. Some mechanisms of fatigue damage growth in rotorcraft composite structures can affect stiffness, dynamic behavior, and functional performance, without being a serious threat to residual strength.

We propose paragraph (h) for those special cases in which it may be impractical to do a damage tolerance evaluation. This proposal would require that any applicant

justify any deviation from the damage tolerance evaluations standards described in paragraphs (c) through (g). Because not all damage scenarios for each PSE are likely to fall within the exception stated in paragraph (h), the applicant must identify and include the justification for establishing impracticability for the specific types of damage associated with the particular rotorcraft composite structure PSE that may qualify for a fatigue evaluation. Finally, we include procedures in the proposal to add mandatory replacement times, structural inspection intervals, and related structural inspection procedures. You must include these replacement times, inspection intervals, and structural inspections in the Instructions for Continued Airworthiness as required by §§ 27.1529 and 29.1529 of Parts 27 and 29 respectively, to minimize the risk of catastrophic failure because of such damage.

Sometimes, operators may not find rare damages, such as high-energy, blunt impacts during a base field inspection at scheduled maintenance inspection intervals. Under this proposal, you must substantiate sufficient residual strength to maintain a sufficient level of safety after such damage has occurred. In addition, supplemental procedures may require maintenance organizations to use more advanced field inspections or contact with the manufacturer when a rare impact occurs, regardless of the damage revealed when using the base inspection techniques.

Appendix A to Parts 27 and 29 – Airworthiness Limitations Section

We propose to revise the second sentence in paragraphs A27.4 and A29.4, “Airworthiness Limitations Section,” as follows:

This section must set forth each mandatory replacement time, structural inspection interval, and related structural inspection procedure required for type certification.

This proposal will require the mandatory replacement times, structural inspection intervals, and related structural inspection procedures produced under the requirements of §§ 27.571 and 29.571, the new §§ 27.573 and 29.573, and any other similar requirement for type certification be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness.

Paperwork Reduction Act

This proposal contains the following new information collection requirements. As required by 44 U.S.C. § 3507(d) of the Paperwork Reduction Act of 1995, as implemented by 5 CFR Part 1320, the FAA has submitted the information requirements associated with this proposal to the Office of Management and Budget (OMB) for review.

Title: Damage Tolerance and Fatigue Evaluation of Composite Rotorcraft Structure.

Summary: This proposal would add new certification standards for normal and transport category rotorcraft to address advances in structural damage tolerance and fatigue substantiation technology for composite rotorcraft structures. These proposals would increase the current minimum safety standards to require compliance with certain current industry practices and FAA policies that would result in higher safety standards, and would result in harmonized international standards. These proposals would help ensure that if damage occurs to composite structures during manufacturing or within the operational life of the rotorcraft, the remaining structure could withstand fatigue loads that are likely to occur, without failure, until the damage is detected. The damaged structure must then be repaired to restore ultimate load capability or the part must be

replaced. Proposed §§ 27.573 and 29.573 would require that applicants get FAA approval of their proposed methods for complying with the certification requirements for damage tolerance and fatigue evaluation of composite structures.

Use of: The required damage tolerance and fatigue evaluation information would be determined for principal composite structural elements or components, detail design points, and fabrication techniques and would be collected from rotorcraft certification applicants. The FAA would use the approval process for the Applicant's submitted compliance methodology to determine whether the proposed methods were sufficient to comply with the certification requirements for damage tolerance and fatigue evaluation of composite structures. The FAA also would use the approval process for the Applicant's submitted compliance methodology to determine if the rotorcraft has any unsafe features in the composite structures.

Respondents (including number of): The likely respondents to this proposed damage tolerance and fatigue evaluation information are applicants requesting type certification of composite structures. We anticipate about 10 normal and transport category rotorcraft certification applicants (including supplemental type certificate applicants) each year.

Frequency: The frequency of determining the damage tolerance and fatigue evaluation methodologies would depend on how often an applicant seeks certification of a composite structure. This compliance methodology would be provided during each certification. We anticipate 10 certifications each year.

Annual Burden Estimate: The compliance methodology would be required to be submitted and approved during each certification of a composite rotorcraft structure.

We anticipate there would be 10 certifications each year and it would take 267 hours to submit and get the compliance methodology approved for each certification for a total annual hourly burden of 2670 hours. We anticipate that submitting and getting approval of the compliance methodology for each certification methodology would cost $150.00 per hour. Therefore, the estimated total annual cost burden would be $40,000.00.

The agency is soliciting comments to--

(1) evaluate whether the proposed information requirement is necessary for the proper performance of the functions of the agency, including whether the information will have practical utility;

(2) evaluate the accuracy of the agency's estimate of the burden;

(3) enhance the quality, utility, and clarity of the information to be collected; and

(4) minimize the burden of collecting information on those who are to respond, by using appropriate automated, electronic, mechanical, or other technological collection techniques or other forms of information technology.

Individuals and organizations may submit comments on the information collection requirement by *[Insert date 60 days after publication in the Federal Register],* and should direct them to the address listed in the **ADDRESSES** section of this document. Comments also should be submitted to the Office of Information and Regulatory Affairs, OMB, New Executive Building, Room 10202, 725 17th Street, N.W., Washington, DC 20053, Attention: Desk Officer for FAA.

According to 5 CFR § 1320.8(b)(3)(vi) and the 1995 amendments to the Paperwork Reduction Act, at 44 U.S.C. § 3506(c)(1)(B)(iii)(v), we may not conduct or sponsor the collection of information, nor may we penalize any person for failing to

comply with these information collection requirements unless we display a currently valid OMB control number. The OMB control number for this information collection will be published in the Federal Register, after the Office of Management and Budget approves it.

International Compatibility

In keeping with U.S. obligations under the Convention on International Civil Aviation, it is FAA policy to comply with International Civil Aviation Organization (ICAO) Standards and Recommended Practices to the maximum extent practicable. The FAA determined that ICAO Annex 8, part IV, Chapter 3, paragraph 3.8, corresponds to these proposed regulations. The proposed regulations are consistent with these general ICAO standards and recommended practices.

Executive Order 12866, DOT Regulatory Policies and Procedures, Economic Assessment, Regulatory Flexibility Determination, International Trade Impact Assessment, and Unfunded Mandates Assessment

Proposed changes to Federal regulations must undergo several economic analyses. First, Executive Order 12866 directs that each Federal agency shall propose or adopt a regulation only upon a determination that the benefits of the intended regulation justify its costs. Second, the Regulatory Flexibility Act of 1980 requires agencies to analyze the economic impact of regulatory changes on small entities. Third, the Trade Agreements Act (19 U.S.C. §§ 2531-2533) prohibits agencies from setting standards that create unnecessary obstacles to the foreign commerce of the United States. In developing U.S. standards, this Trade Act also requires agencies to consider international standards and, where appropriate, use them as the basis of U.S.

standards. Fourth, the Unfunded Mandates Reform Act of 1995 (Public Law 104-4) requires agencies to prepare a written assessment of the costs, benefits, and other effects of proposed or final rules that include a Federal mandate likely to result in the expenditure by State, local, or tribal governments, in the aggregate, or by the private sector, of $100 million or more annually (adjusted for inflation).

In conducting these analyses, FAA has determined this rule: (1) has benefits that do justify its costs, (2) does not impose costs sufficient to be considered "significant" under the economic standards for significance under Executive Order 12866 or under DOT's Regulatory Policies and Procedures, (3) would not have a significant economic impact on a substantial number of small entities, (4) would not constitute a barrier to international trade, and (5) would not constitute an unfunded mandate. The FAA has placed these analyses in the docket and summarized them below.

The proposed rule would amend Parts 27 and 29 of Title 14, Code of Federal Regulations (14 CFR) to modify the regulations applicable to normal and transport category rotorcraft structures. This proposed rule would add airworthiness standards to evaluate the fatigue strength of composite rotorcraft structure using a damage tolerance evaluation. The current regulations were written for metallic structure and do not provide adequate certification standards for composite materials and structures. This proposal would add safety requirements for composite rotorcraft structure to address advances in composite structures technology. As well as improving the level of safety for all principal structural elements, the proposed rule would lead to a harmonized international standard with the European Authorities.

Summary of the Cost-Benefit Analysis

Overview of Costs and Benefits

The proposed rule would require rotorcraft manufacturers and operators to take additional actions that include: 1) perform a more thorough threat assessment, and submit the threat assessment methodology report to the FAA for approval, 2) perform a more rigorous residual strength assessment, and 3) establish inspection intervals and conduct inspections. It is current practice for rotorcraft manufacturers to submit voluntarily a threat assessment methodology report to the FAA for approval. However, there may be some additional costs associated with this methodology report for current manufacturers. The rotorcraft manufacturers currently perform a threat assessment and a residual strength assessment, but those would become more robust under the proposed rule. The current rule mandates that manufacturers establish inspection intervals or retirement times. The proposed rule mandates that both retirement times and inspection intervals are to be included in the Airworthiness Limitation Section of the Instructions for Continued Airworthiness. It is current practice for the rotorcraft manufacturers to establish retirement times for composite structures. Therefore, there should be no additional costs associated with establishing retirement times, but the manufacturers must establish inspection intervals.

Costs

Based on information from industry representatives on the ARAC Working Group, the FAA estimates the average additional cost for Part 29 aircraft manufacturers to perform a more thorough threat assessment would be $100,000 for each certification. Also, the average additional cost to perform the more rigorous residual strength

assessment proposed by this rule would be an additional $50,000 for each certification. Including retirement times, inspection intervals, and inspection procedures in the airworthiness limitation section of the Instructions for Continued Airworthiness would cost on average an additional $54,000 for each certification. Based on information received from industry representatives, the FAA also estimates that over the next 20 years, Part 29 rotorcraft structures will be comprised of approximately 50% metallic parts and 50% composite parts, and Part 27 rotorcraft structures will be comprised of approximately 80% metallic parts and 20% composite parts. Therefore, the additional Part 29 certification cost under this proposed rule would be $50,000 for a threat assessment ($100,000 * 0.5 = $50,000), $25,000 for a residual strength assessment ($50,000 * 0.5 = $25,000), and $27,000 for putting both inspection intervals and retirement times in the Airworthiness Limitation Section ($54,000 * 0.5 = $27,000). Therefore, the FAA estimates the average total certification cost for each new Part 29 type certification would be $102,000 ($50,000 + $25,000 + $27,000 = $102,000).

Based on information from industry representatives, the FAA estimates there should be minimal additional costs to Part 27 manufacturers for the threat assessment and the residual strength assessment. Including retirement times, inspection intervals, and inspection procedures in the Airworthiness Limitation Section would cost on average an additional $48,000 for each Part 27 certification. Therefore, the average total certification cost for each new Part 27 type certification under this proposed rule would be $9,600 ($48,000 * 0.2 = $9,600).

The inspections of composite structures would include visual, x-ray, and ultrasonic inspections. Industry representatives on the ARAC Working Group also

estimated that it would take a mechanic approximately 55 hours to perform a composite inspection on a Part 29 rotorcraft, and approximately 24 hours to perform a composite inspection on a Part 27 rotorcraft. For the Part 29 composite rotorcraft structure inspections, the inspection time is based on industry's estimate of 8 hours for each rotor blade, 1 extra hour for each rotor blade pitch change mechanism, which is part of the main rotor assembly, and 2-4 hours for the composite stabilizer. Based on industry information, a Part 29 rotorcraft is assumed to have 6 rotor blades (4 main rotor blades and 2 tail rotor blades), 4-rotor blade pitch change mechanism which is part of the main rotor assembly, and a single stabilizer. Therefore, the total time to inspect a Part 29 rotorcraft would be 54-56 hours (8*6 + 1*4 = 52 hours plus 2 to 4 hours = 54 to 56 hours). The FAA agrees with industry's estimate, and is using an average of 55 hours for a Part 29 inspection. For the Part 27 composite rotorcraft structure inspections, the inspection time is based on industry's estimate of 6 hours for each blade. Based on industry information, a Part 27 rotorcraft is assumed to have 4 blades on each aircraft. Therefore, the total is 24 hours (6*4 = 24). The FAA agrees with industry's estimate of 24 hours for a Part 27 inspection.

At the mechanic wage rate of $60 per hour, each Part 29 composite inspection would cost $3,300 (55 work-hours * $60 per hour = $3,300), and each Part 27 composite inspection would cost $1,440 (24 work-hours * $60 per hour = $1,440). Based on information received from industry representatives, the FAA estimates that inspections would occur on average approximately every 1,250 flight hours for Part 29 rotorcraft, and would occur on average approximately every 3,750 flight hours for Part 27 rotorcraft. Based on FAA data and information from industry representatives, the

FAA estimates that inspections would occur on average about once every 3 years.

According to the “2003 Aerospace Source Book” by Aviation Week & Space Technology (January 13, 2003), the growth of the civil helicopter market is expected to be flat for the next several years, with perhaps a few percent growth each year. According to the “FAA Aerospace Forecasts: Fiscal Years 2003-2014” (March 2003), the number of turbine powered rotorcraft is expected to total 4,590 by 2014—an increase of less than 100 rotorcraft over the 2001 level. Therefore, the rate of new rotorcraft production is assumed to approximate the rate of rotorcraft attrition.

Based on information from industry representatives, the FAA estimates there would be 2 new Part 29 type certifications every 10 years, and 3.5 new Part 27 type certifications every 10 years. For cost estimation purposes, the FAA assumes the new models would be certificated in years 1 and 11 during the 20-year analysis period, and that each future rotorcraft certification would have a production run of 10 years. The forecast production rates for new Part 29 models are based on information in the “World Rotorcraft Overview” (July 2002) by the Teal Group. The forecast production rates for new Part 27 models are based on information from industry representatives. The FAA estimates the total inspections costs over the 20-year analysis period would be about $18,640,000 in undiscounted costs or about $7,345,000 in discounted costs (by applying a 7% discount rate). Therefore, the total costs of this proposed rule over 20 years is estimated to be about $19,116,000 in undiscounted costs ($18,640,500 + $475,200 = $19,115,700) or about $7,680,000 in discounted costs ($7,344,916 + $334,938 = $7,679,854).

<u>Benefits</u>

Discounted at 7 percent annually, total potential benefits for significantly reducing the likelihood of fatigue-related accidents for rotorcraft composite structures amount to an estimated $15.6 million over the 20-year analysis period. Without a new rule, it is likely that future fatigue-related accidents will occur. A key benefit of the proposed rule would be avoiding the fatigue-related accidents.

Using composite materials in aircraft structures is much newer than the use of metallic materials. Therefore, the FAA has used advisory material and a broad interpretation of the fatigue substantiation and the design and construction airworthiness standards to maintain a certain level of safety. For purposes of this benefits analysis, the FAA assumes that if rotorcraft involved in accidents because of metallic fatigue had been equipped with composite materials instead, several rotorcraft accidents could have been avoided. Based on estimates from industry experts, the reduction in failure rates of composite materials compared to metallic materials is 75 percent. The FAA assumes that in the future, composite materials will be used more often in producing rotorcraft structures.

In the review of the accident and incident history, the FAA only considered accidents that were relevant to metallic rotorcraft structure fatigue problems that could have been avoided if composite replacements had been used. For example, 13 of the Part 29 metallic fatigue accidents could have been avoided if a similar rotorcraft metallic structures rule had been in place, but only 9 out of the 13 accidents were due to failures of metallic materials that had composite replacements. Similarly, 24 of the Part 27 metallic fatigue accidents could have been avoided if a similar rotorcraft metallic structures rule had been in place, but only 14 out of the 24 accidents were due to

failures of metallic materials that had composite replacements. In addition, the FAA did not consider events in which externally aggravating circumstances existed, such as when repairs weren't done according to the manufacturer's specifications. Databases that the FAA examined include the NTSB Aviation Accident Database & Synopses, the FAA Aviation Safety Accident Prevention (ASAP) database, and the National Aviation Safety Data Analysis Center (NASDAC) database.

Since 1982, a total of 23 accidents (9 + 14 = 23) were identified that may have been prevented if existing composite replacement materials had been used and if this rule had been in effect. The 23 accidents resulted in 28 fatalities, 8 serious injuries, and 12 minor injuries. In addition, all the aircraft involved in the accidents were either destroyed or received substantial damage. Assuming a 75 percent reduced failure rate for composite materials, there would have been only about 6 accidents (23 x 0.25 = 5.75) if composite structures had been used instead of the metallic structures. If this rule had been in effect, these 6 accidents could have been prevented.

To quantify future benefits, the FAA calculated the costs of a future averted accident as a direct result of this proposed rule. The minimum value of a statistical aviation fatality avoided is set at $3.0 million, that of a serious injury (assumed to be the average of a severe, serious, and moderate injury) at $260,500, and that of a minor injury at $6,000. The associated medical and legal costs for a fatality is $132,700, a serious injury (assumed to be the average of a severe, serious, and moderate injury) $46,633, and that of a minor injury, $2,500. In addition, the weighted average replacement cost of a destroyed rotorcraft is represented by a value of $1,463,000, the weighted average restoration cost of a substantially damaged rotorcraft is represented

by a value of $244,000, and an NTSB accident investigation costs about $26,000. The number of fatalities, serious injuries and minor injuries represents the average number of such casualties in the accidents. Based on the above information, the FAA estimates the average value of avoiding a fatigue-related composite rotorcraft accident is about $4.9 million.

Given that six composite structure accidents could have been prevented in the past, without preventative action a number of accidents could occur in the future. The Poisson probability distribution provides a good model for estimating the number of "rare events" observed in a given unit of time. Using the Poisson probability distribution, the FAA estimated probabilities associated with the projected number of future accidents (rare events) for the proposed rulemaking. Based on the Cumulative Poisson probability distribution with mean equal to 6, over the next 20 years, there is a probability of approximately 93% that there would be 3 or more accidents.

The present value benefit estimate assumes that the probability of an accident is equally likely in any year of the 20-year study period. The undiscounted benefit in any year would be $1,476,189 (6 accidents x $4,920,630 per accident / 20 years = $1,476,189 per year). If 6 accidents were avoided over the next 20 years, the present value benefit would be approximately $15.6 million.

The benefits of the proposed regulation include harmonization with the European standard. Such harmonization would result in improved acceptability of FAA certificated products in European countries. The FAA has not attempted to quantify the cost savings that may accrue due to harmonization of this rule, beyond noting that it would be a large potential cost savings.

Comparison

In the absence of this proposed rule, it is highly likely that future fatigue-related failures of composite rotorcraft structure contributing to accidents will occur. The FAA finds that on average, six accidents within the fleet could be prevented by the enactment of this proposed rule. The benefit of the proposed rule would be the avoidance of these accidents. The FAA estimates that the discounted present value (2003$) benefits of the proposed rule would be about $15.6 million if six accidents were avoided. As previously discussed, the probability of three or more accidents occurring in the absence of this rule is 93%. The benefit of avoiding three accidents is about $7.8 million. Accordingly, based on this analysis, there is a 93% probability that even if only three accidents are avoided, the benefits of this proposal will exceed its costs. These benefits are derived from preventing accidents due to fatigue-related failures.

It is estimated that the discounted present value (2003$) cost of the proposed rule would be $7.68 million. The cost figure above includes the cost of systems design, qualification, certification, equipment purchase and installation, testing, and inspections.

The estimated $15.6 million benefits of this proposed rule far exceeds the estimated $7.68 million costs. The $15.6 million in benefits assumes that all future accidents due to fatigue-related failures are prevented within the aircraft produced under the 5.5 new certifications. Therefore, if this rule is more than 49.1% effective ($7.68/$15.6 = 0.4911), then benefits would exceed costs. Thus, the FAA concludes that the benefits of the proposed rule do justify its costs. However, we request comments with supporting justification on both the cost and benefit estimates.

Regulatory Flexibility Determination

The Regulatory Flexibility Act (RFA) of 1980 as amended by the Small Business Regulatory Enforcement Fairness Act of 1996 (the Act) (5 U.S.C. 601-612), establishes, as a principle of regulatory issuance that agencies shall endeavor, consistent with the objective of the rule and of applicable statutes, to fit regulatory and informational requirements to the scale of the business, organizations, and governmental jurisdictions subject to regulation." Congress has further found that "small businesses bear a disproportionate share of regulatory costs and burdens." To address these concerns, the Act requires agencies to solicit and consider flexible regulatory proposals and to explain the rationale for their actions. The Act covers a wide-range of small entities, including small businesses, not-for-profit organizations and small governmental jurisdictions.

Agencies must perform a review to determine whether a proposed or final rule would have a significant economic impact on a substantial number of small entities. If the determination is that it would, the agency must prepare a regulatory flexibility analysis as described in the Act.

However, if an agency determines that a proposed or final rule is not expected to have a significant economic impact on a substantial number of small entities, section 605(b) of the Act provides that the head of the agency may so certify and a regulatory flexibility analysis is not required. The certification must include a statement providing the factual basis for this determination, and the reasoning should be clear.

The FAA believes that this proposed rule would not have a significant economic impact on a substantial number of small entities. Four out of six rotorcraft manufacturers potentially impacted by this proposed rule exceed the Small Business

Administration small-entity criteria of 1,500 employees for aircraft manufacturers. Sikorsky Aircraft (a subsidiary of United Technologies), Bell Helicopters (a subsidiary of Textron Inc.), Kaman Aerospace Corporation, and Bell/Augusta Aerospace Corporation (a joint venture of Textron and Augusta Westland) are not considered small entities because they have over 1,500 employees. On the other hand, Enstrom Helicopter Corporation and Schweizer Aircraft Corporation (now owned by Sikorsky) are considered small entities because they have fewer than 1,500 employees.

Robinson Helicopter Company is not included in this list because Robinson does not use, and does not plan to use, composites in any primary structures. Therefore, Robinson probably would not be impacted by this proposed rule. Enstrom Helicopter Corporation and Schweizer Aircraft Corporation (now owned by Sikorsky) do use composites in the production of primary rotorcraft structures. However, the FAA believes that two companies do not constitute a substantial number of small entities. Given that only two small entity helicopter manufacturers would be impacted by this proposal, the FAA certifies that this proposed rule would not have a significant economic impact on a substantial number of small entities.

Trade Impact Assessment

The Trade Agreement Act of 1979 prohibits Federal agencies from establishing any standards or engaging in related activities that create unnecessary obstacles to the foreign commerce of the United States. Legitimate domestic objectives, such as safety, are not considered unnecessary obstacles. The statute also requires consideration of international standards and, where appropriate, that they be the basis for U.S. standards. The FAA has assessed the potential effect of this proposed rule and

determined that it would harmonize the U.S. standards with the European international standards thereby lowering the costs of international trade.

Unfunded Mandates Assessment

The Unfunded Mandates Reform Act of 1995 (2 U.S.C. § 1501-1571), requires each Federal agency, to the extent permitted by law, to prepare a written assessment of the effects of any Federal mandate in a proposed or final agency rule that may result in the expenditure by State, local, and tribal governments, in the aggregate, or by the private sector, of $100 million or more (adjusted annually for inflation) in any one year. This proposed rule does not contain a Federal intergovernmental or private sector mandate that exceeds $100 million in any year; therefore, the requirements of the act do not apply.

Executive Order 13132, Federalism

The FAA has analyzed this proposed rule under the principles and criteria of Executive Order 13132, Federalism. We determined that this action would not have a substantial direct effect on the States, on the relationship between the national Government and the States, or on the distribution of power and responsibilities among the various levels of government, and therefore, would not have federalism implications.

Regulations Affecting Intrastate Aviation in Alaska

The Federal Aviation Reauthorization Act of 1996, Public Law No. 104-264, § 1205, 110 Stat. 3213, 3286, requires the Administrator, when modifying regulations in title 14 of the Code of Federal Regulation in any manner affecting intrastate aviation in Alaska, to consider the extent to which Alaska is not served by transportation modes other than aviation, and to establish such regulatory distinctions as he or she considers

appropriate. Because this proposed rule would apply to the certification of future designs of normal and transport category rotorcraft and their subsequent operation, it could, if adopted, affect intrastate aviation in Alaska. The FAA therefore specifically requests comments on whether there is justification for applying the proposed rule differently in intrastate operations in Alaska.

Plain English

Executive Order 12866 (58 FR 51735, Oct. 4, 1993) requires each agency to write regulations that are simple and easy to understand. We invite your comments on how to make these proposed regulations easier to understand, including answers to questions such as the following:

- Are the requirements in the proposed regulations clearly stated?
- Do the proposed regulations contain unnecessary technical language or jargon that interferes with their clarity?
- Would the regulations be easier to understand if they were divided into more (but shorter) sections?
- Is the description in the preamble helpful in understanding the proposed regulations?

Please send your comments to the address specified in the ADDRESSES section.

Environmental Analysis

FAA Order 1050.1E identifies FAA actions that are categorically excluded from preparation of an environmental assessment or environmental impact statement under the National Environmental Policy Act (NEPA) in the absence of extraordinary circumstances. The FAA has determined this proposed rulemaking action qualifies for

the categorical exclusion identified in paragraph 312f and involves no extraordinary circumstances.

Regulations That Significantly Affect Energy Supply, Distribution, or Use

The FAA has assessed the energy impact of the proposed rule in accordance with the Energy Policy and Conservation Act (EPCA) Public Law 94-163, as amended (42 U.S.C. 6362) and the Department of Transportation implementing regulations, specifically 14 C.F.R. § 313.4, that defines a "major regulatory action." We have determined that this notice is not a major regulatory action under the provision of EPCA. Additionally, we have analyzed this proposal under Executive Order 13211, Actions Concerning Regulations that Significantly Affect Energy Supply, Distribution, or Use (May 18, 2001). We have determined that it is not a "significant energy action" under executive order because it is not a "significant regulatory action" under Executive Order 12866, and it is not likely to have a significant adverse affect on the supply, distribution, or use of energy.

List of Subjects

14 CFR Part 27

Air transportation, Aircraft, Aviation safety, Rotorcraft, Safety.

14 CFR Part 29

Air transportation, Aircraft, Aviation safety, Rotorcraft, Safety.

The Proposed Amendment

In consideration of the foregoing, the Federal Aviation Administration proposes to amend Parts 27 and 29 of Title 14, Code of Federal Regulations, as follows:

PART 27-AIRWORTHINESS STANDARDS: NORMAL CATEGORY ROTORCRAFT

1. The authority citation for part 27 continues to read as follows:

Authority: 49 U.S.C. 106(g), 40113, 44701–44702, 44704

2. Add a new § 27.573 to read as follows:

§ 27.573 Damage Tolerance and Fatigue Evaluation of Composite Rotorcraft Structure.

(a) You, the applicant, must evaluate the composite rotorcraft structure under the damage tolerance standards of paragraphs (b) through (g) of this section unless you show that it is impractical within the limits of geometry, inspectability, and good design practice. If you establish that the damage tolerance evaluation described in paragraphs (c) through (g) of this section is impractical for a particular composite rotorcraft structure, then you must do a fatigue evaluation in accordance with requirements of paragraph (h) of this section.

(b) You must submit the compliance methodology to the FAA for approval.

(c) You must show that catastrophic failure due to static and fatigue loads considering the intrinsic or discrete manufacturing defects or accidental damage are avoided throughout the operational life or prescribed inspection intervals of the rotorcraft, by performing damage tolerance evaluations of the strength of composite principal structural elements (PSEs) or components, detail design points, and fabrication techniques. Catastrophic failure is an event that could prevent continued safe flight and landing. You must account for the effects of material and process variability along with environmental conditions in the strength and fatigue evaluations. You must evaluate parts that include PSEs of the airframe, main and tail rotor drive systems, main and tail rotor blades and hubs, rotor controls, fixed and movable control

surfaces, engine and transmission mountings, landing gear, and other parts, detail design points, and fabrication techniques deemed critical by the FAA. Each evaluation required by this section must include:

(1) The identification of all PSEs. PSEs are structural elements that contribute significantly to the carrying of flight or ground loads and the failure of which could result in catastrophic failure of the rotorcraft.

(2) In-flight and ground measurements for determining the loads or stresses for all PSEs, for all critical conditions throughout the range of limits in § 27.309 (including altitude effects), except that maneuvering load factors need not exceed the maximum values expected in service.

(3) The loading spectra as severe as those expected in service based on loads or stresses determined under paragraph (c)(2) of this section, including external load operations, if applicable, and other operations including high-torque events.

(4) A threat assessment for all PSEs that specifies the locations, types, and sizes of damage, considering fatigue, environmental effects, intrinsic and discrete flaws, and impact or other accidental damage (including the discrete source of the accidental damage) that may occur during manufacture or operation.

(5) An assessment of the residual strength and fatigue characteristics of all PSEs that supports the replacement times and inspection intervals established under paragraphs (e) and (f) of this section.

(d) For all PSEs, you must establish replacement times, inspections, or other procedures to require replacing or revealing compromised parts before a catastrophic failure. You must include these replacement times, inspections, or other procedures in

the Airworthiness Limitations Section of the Instructions for Continued Airworthiness required by § 27.1529.

(e) You must determine replacement times for PSEs by tests, or by analysis supported by tests, and must show that the structure is able to withstand the repeated loads of variable magnitude expected in-service. In establishing these replacement times, you must consider the following items:

(1) Damage identified by the threats determined in paragraph (c)(4) of this section.

(2) Maximum acceptable manufacturing defects and in-service damage (i.e., those that do not lower the residual strength below ultimate design loads and those that can be repaired to restore ultimate strength).

(3) Ultimate load strength capability after applying repeated loads.

(f) You must establish inspection intervals for PSEs that will reveal any damage identified in paragraph (c)(4) of this section that may occur from fatigue or other in-service causes before such damage has grown to the extent that the component cannot sustain the required residual strength capability. In establishing these inspection intervals, you must consider the following items:

(1) The growth rate or no-growth of the damage under the repeated loads expected in-service determined by tests or analysis supported by tests.

(2) The required residual strength for the assumed damage established after considering the damage type, inspection interval, detectability of damage, and the techniques adopted for damage detection. The minimum required residual strength is limit load.

(3) Whether the inspection will detect the damage growth before the minimum residual strength is reached and restored to ultimate load capability, or whether the component will require replacement.

(g) You must consider the effect of damage on stiffness, dynamic behavior, loads, and functional performance on all PSEs in establishing the allowable damage size and inspection interval.

(h) If you establish that the damage tolerance evaluation described in paragraphs (c) through (g) of this section is impractical within the limits of geometry, inspectability, or good design practice for a particular composite rotorcraft structure, then you must do a fatigue evaluation for the particular composite rotorcraft structure and:

(1) identify all PSE's considered in the fatigue evaluation;

(2) identify the types of damage for all PSE's considered in the fatigue evaluation;

(3) establish supplemental procedures to minimize the risk of catastrophic failure associated with the damages identified in paragraph (h)(2) of this section; and

(4) include these supplemental procedures in the Airworthiness Limitations section of the Instructions for Continued Airworthiness required by § 27.1529.

3. Amend APPENDIX A TO PART 27-INSTRUCTIONS FOR CONTINUED AIRWORTHINESS in section A.27.4, Airworthiness Limitations Section, by replacing "approved under § 27.571" with "required for type certification" at the end of the second sentence.

PART 29-AIRWORTHINESS STANDARDS: TRANSPORT CATEGORY ROTORCRAFT

4. The authority citation for part 29 continues to read as follows:

Authority: 49 U.S.C. 106(g), 40113, 44701–44702, 44704

5. Add a new § 29.573 to read as follows:

§ 29.573 Damage Tolerance and Fatigue Evaluation of Composite Rotorcraft Structure

(a) You, the applicant, must evaluate the composite rotorcraft structure under the damage tolerance paragraphs (b) through (g) of this section unless you show that it is impractical within the limits of geometry, inspectability, and good design practice. If you establish that the damage tolerance evaluation described in paragraphs (c) through (g) of this section is impractical for a particular composite rotorcraft structure, then you must do a fatigue evaluation in accordance with requirements of paragraph (h) of this section.

(b) You must submit the compliance methodology to the FAA for approval.

(c) You must show that catastrophic failure due to static and fatigue loads considering the intrinsic or discrete manufacturing defects or accidental damage are avoided throughout the operational life or prescribed inspection intervals of the rotorcraft, by performing damage tolerance evaluations of the strength of composite principal structural elements (PSEs) or components, detail design points, and fabrication techniques. Catastrophic failure is an event that could prevent continued safe flight and landing. You must account for the effects of material and process variability along with environmental conditions in the strength and fatigue evaluations.

You must evaluate parts that include PSEs of the airframe, main and tail rotor drive systems, main and tail rotor blades and hubs, rotor controls, fixed and movable control surfaces, engine and transmission mountings, landing gear and other parts, detail design points, and fabrication techniques deemed critical by the FAA. Each evaluation required by this section must include:

(1) The identification of all PSEs. PSE's are structural elements that contribute significantly to the carrying of flight or ground loads and the failure of which could result in catastrophic failure of the rotorcraft.

(2) In-flight and ground measurements for determining the loads or stresses for all PSEs for all critical conditions throughout the range of limits in § 29.309 including altitude effects, except that maneuvering load factors need not exceed the maximum values expected in service.

(3) The loading spectra as severe as those expected in service based on loads or stresses determined under paragraph (c)(2) of this section, including external load operations, if applicable, and other operations including high-torque events.

(4) A threat assessment for all PSEs that specifies the probable locations, types, and sizes of damage, considering fatigue, environmental effects, intrinsic and discrete flaws, and impact or other accidental damage (including the discrete source of the accidental damage) that may occur during manufacture or operation.

(5) An assessment of the residual strength and fatigue characteristics of all PSEs that supports the replacement times and inspection intervals in paragraphs (e) and (f) of this section.

(d) For all PSEs, you must establish replacement times, inspections, or other

procedures to require replacing or revealing compromised parts before a catastrophic failure. You must include these replacement times, inspections, or other procedures in the Airworthiness Limitations section of the Instructions for Continued Airworthiness required by § 29.1529.

(e) You must determine replacement times for PSEs by tests or by analysis supported by tests, and must show that the structure is able to withstand the repeated loads of variable magnitude expected in-service. In establishing these replacement times, you must consider the following items:

(1) Damage identified by the threats determined in paragraph (c)(4) of this section.

(2) Maximum acceptable manufacturing defects and in-service damage (i.e., those that do not lower the residual strength below ultimate design loads and those that can be repaired to restore ultimate strength).

(3) Ultimate load strength capability after applying repeated loads.

(f) You must establish inspection intervals for PSEs that will reveal any damage identified in paragraph (c)(4) of this section that may occur from fatigue or other in-service causes before such damage has grown to the extent that the component cannot sustain the required residual strength capability. In establishing these inspection intervals, you must consider the following items:

(1) The growth rate or no-growth of the damage under the repeated loads expected in-service determined by tests or analysis supported by tests.

(2) The required residual strength for the assumed damage established after considering the damage type, inspection interval, detectability of damage, and the

techniques adopted for damage detection. The minimum required residual strength is limit load.

(3) Whether the inspection will detect the damage growth before the minimum residual strength is reached and restored to ultimate load capability, or whether the component will require replacement.

(g) You must consider the effect of damage on stiffness, dynamic behavior, loads, and functional performance on all PSEs in establishing the allowable damage size and inspection interval.

(h) If you establish that the damage tolerance evaluation described in paragraphs (c) through (g) of this section is impractical within the limits of geometry, inspectability, or good design practice for a particular composite rotorcraft structure, then you must do a fatigue evaluation for that particular composite rotorcraft structure and:

(1) identify all PSE's considered in the fatigue evaluation;

(2) identify the types of damage for all PSE's considered in the fatigue evaluation;

(3) establish supplemental procedures to minimize the risk of catastrophic failure associated with the damages identified in paragraph (h)(2) of this section; and

(4) include these supplemental procedures in the Airworthiness Limitations section of the Instructions for Continued Airworthiness required by § 29.1529.

6. Amend APPENDIX A TO PART 29-INSTRUCTIONS FOR CONTINUED AIRWORTHINESS, in section A.29.4, Airworthiness Limitations section, by replacing “approved under § 29.571” with “required for type certification” at the end of the second sentence.

Issued in Washington, DC, on [the date of issuance is the signature date.]

[Name of Office Director]

[Title of Office Director]

[Name and title of the individual signing the NPRM, generally, the OPI director. If the individual signing the NPRM is “acting” for another individual, this must be noted in the signature block.]

G:Rule:DraftRule:Composite folder: file: 07-26-05_newestcompV_cch_comments.doc

O:Rule:DraftRule:Composite folder:1-5-06_newestcompV_kj_comments.doc

O:Rule:DraftRule:Composite folder:8-16-06-with-stylewriter.doc

O:Rule:DraftRule:Composite folder:12/12/06-with-addition from Sikorsky

O:Rule:DraftRule:Composite folder:1/19/07-with-Legal-cmts.doc

O:Rule:DraftRule:Composite folder:1/29/07-with-addtl-Legal-cmts.doc