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

AC 29.573. § 29.573 (Amendment 29- 54) DAMAGE TOLERANCE AND FATIGUE EVALUATION OF COMPOSITE ROTORCRAFT STRUCTURES

a. Purpose. This advisory material provides an acceptable means of compliance with the provisions of § 29.573, Amendment 29-54, Title 14 of the Code of Federal Regulations (CFR) dealing with the damage tolerance and fatigue evaluation of transport category composite rotorcraft structures. Paragraph f.(6) specifically addresses the advisory guidance applying to damage tolerance and fatigue evaluation as required by § 29.573, Amendment 29-54. Some information contained in AC 29-2C, MG 8 (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 29-54 . (Supplemental guidance can be found in AC 20-107, "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. 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 is significantly different from fixed wing fatigue substantiation. Since AC 20-107, 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 20-107 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 (see AC 29-2C, AC 29 MG 11, "Fatigue Tolerance Evaluation of Transport Category Rotorcraft Metallic Structure"). 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 MG 11, "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; Composites Materials Handbook-17 (CMH-17), formerly the 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 20-107 and provides compliance guidance for the requirements of § 29.573 Amendment 29-54 for rotorcraft composite structure.

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

(1) 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.

(2) 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.

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

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

(5) 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.

(6) Autoclave. A closed apparatus usually equipped with variable conditions of vacuum, pressure, and temperature. It is used for bonding, compressing or curing materials.

(7) 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.

(8) Balanced Laminate. A composite laminate in which all laminae at angles other than 0° occur only in ± pairs (not necessarily adjacent).

(9) <u>Bond</u>. The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent.

(10) <u>Catastrophic Failure.</u> An event that could prevent continued safe flight and landing.

(11) <u>Cocure</u>. 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.

(12) <u>Component</u>. 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.

(13) <u>Coupon.</u> 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).

(14) <u>Cure</u>. 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.

(15) <u>Damage.</u> 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.

(16) <u>Damage Tolerance</u>. 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.

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

(18) <u>Damage Tolerant Safe Life</u>. 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.

(19) <u>Delamination</u>. The separation of the layers of material in a laminate.

(20) <u>Design Limit Loads</u>. The maximum loads to be expected in service, as defined by § 29.301(a).

(21) 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).

(22) 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.

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

(24) 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).

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

(25) 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.

(26) 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.

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

(29) 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.

(30) Glass Transition Temperature. The approximate midpoint of the temperature range over which the glass transition takes place.

(31) Hybrid. Any mixture of fiber types (e.g., graphite and glass).

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

(33) 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.

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

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

(36) 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.

(37) Material System. The combination of single constituents chosen (e.g., fiber and resin).

(38) Material System Constituent. A single constituent (ingredient) chosen for a material system (e.g., a fiber, a resin).

(39) 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.

(40) 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.”

(41) 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.

(42) 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.

(43) 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.

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

(45) 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 that will involve complete curing. Pre-preg is usually drapable, tacky, and can be easily handled.

(46) 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.

(47) 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.

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

(49) Resin Content. The amount of matrix present in a composite by either percent weight or percent volume.

(50) 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 COCURE).

(51) 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.

(52) 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). Strain level is generally measured in thousandths of an inch per unit inch of part or microinches/inch (e.g., .003 in/in equals 3000 microinches/inch).

(53) 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).

(54) Symmetrical Laminate. A composite laminate in which the ply orientation is symmetrical about the laminate midplane.

(55) Tape. Hot melt impregnated fibers forming unidirectional pre-preg.

(56) 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.

(57) 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).

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

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

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 20-107	"Composite Aircraft Structure"
AC 21-26	"Quality Control for the Manufacture of Composite Materials"
CMH-17	"Composite Materials Handbook"
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: fabrication requirements; basic constituent, pre-preg and laminate material acceptance requirements, and material property determination requirements; protection of structure; lightning protection; static strength evaluation; damage tolerance and fatigue evaluation; dynamic loading and response evaluation; and special repair and continued airworthiness requirements.

Original as well as alternate or substitute material system constituents (e.g., fibers, resins), material systems (combinations of constituents and adhesives), and composite designs (e.g., laminates, cocured assemblies, bonded assemblies) 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 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 FAA/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 engineering. Engineering practices for many of the areas identified below are available in CMH-17.

(1) The first area is the fabrication requirements 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 (e.g., fiber waviness, warp defects, fill defects, porosity, hole edge effects, edge defects, resin content, large area disbonds, and delaminations) 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 created, validated, and approved for defect detection. 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 - see paragraph f.(6)) 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 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 certification team members (both the manufacturing inspection district office (MIDO) and FAA 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 CMH-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 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 witnessed.

(C) Material System Component Storage and Handling. Applicants should be able to demonstrate to FAA 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 (that 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) 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 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 specifications should be approved and must comply with § 29.605. Any alternate specifications should provide at least the same level of quality and safety as the original specification. Any changes should be presented for FAA approval well in advance of the effective date of the production change.

(2) The <u>second</u> area is the <u>basic raw constituent, pre-preg and laminate material acceptance requirements, and material property determination requirements</u> 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 CMH-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 CMH-17) has not completed development of "B" basis allowables for inclusion in CMH-17. 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 CMH-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) and constituents (e.g., fibers, resins) 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 CMH-17, other similar approved methods or per FAA 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, T_g, 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 element in a composite matrix and the T_g is critical to structural integrity, a T_g determination is necessary. The T_g margin methodology of CMH-17 should be implemented (i.e., the T_g should be 50° F higher than the maximum structural temperature (see definition)). For any type of resin or adhesive, an acceptable temperature margin using CMH-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) and structural discontinuities (e.g., joints, splices). 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 scale 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 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 applicants have not requested 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-approved.

(viii) Material allowable strength values for full-scale design and testing should be developed using the coupon procedures presented in CMH-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-approved early in certification (as part of the building block process), that reflect the material property determination considerations recommended in CMH-17 on an equal to or better than basis.

(3) The third area is the protection of structure as required by § 29.609. Protection against thermal, humidity, and other environmental effects (e.g., weathering, abrasion, fretting, hail, ultraviolet radiation, chemical effects, accidental damage) 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 20-53, "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 (see paragraph f.(2)) and protection of structure (see paragraph 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.

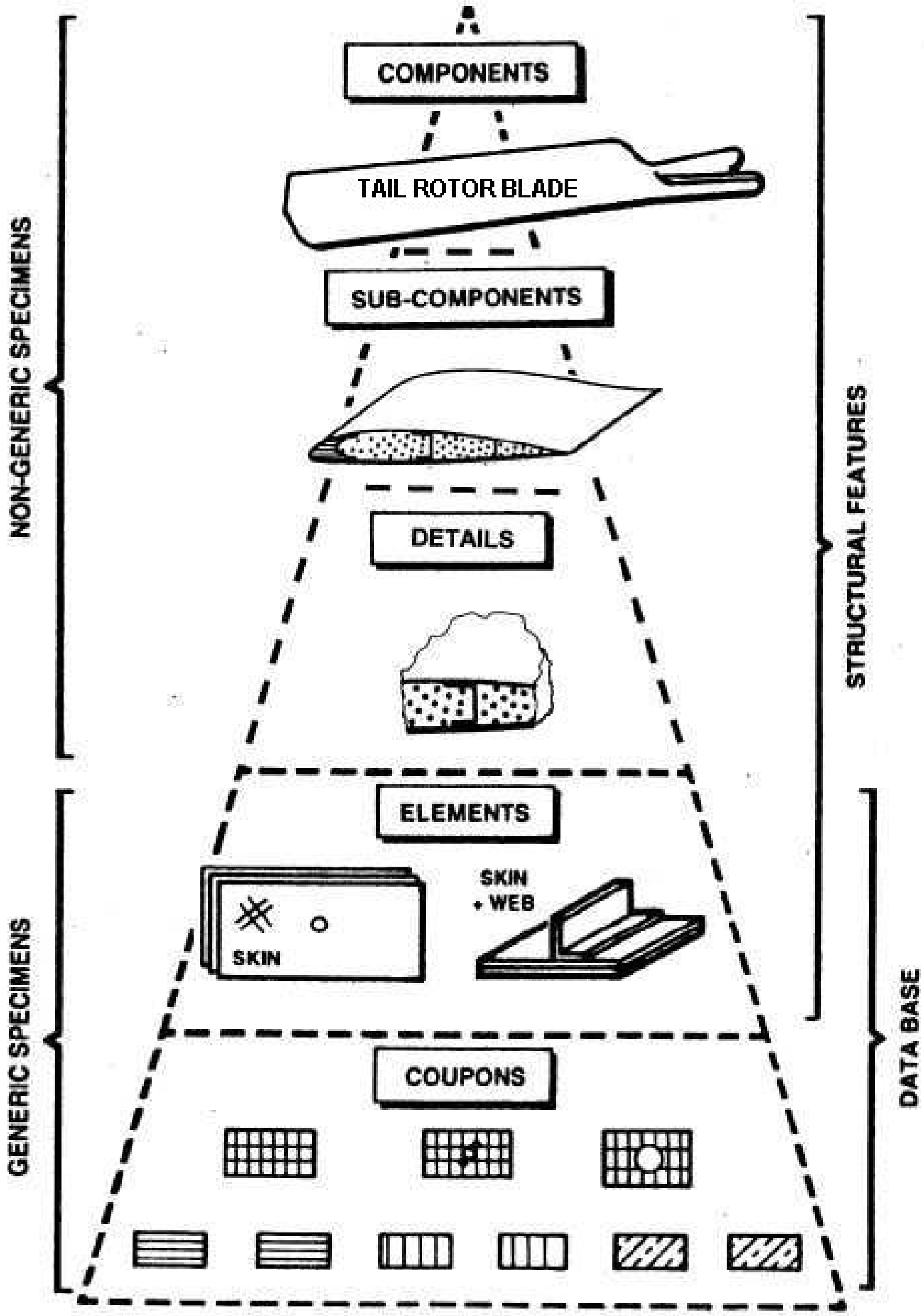

Figure AC 29.573-1. Schematic diagram of building block tests.

(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), 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 (see 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, FAA/AUTHORITY 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 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, 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) 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 FAA/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, 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 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 (see 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.

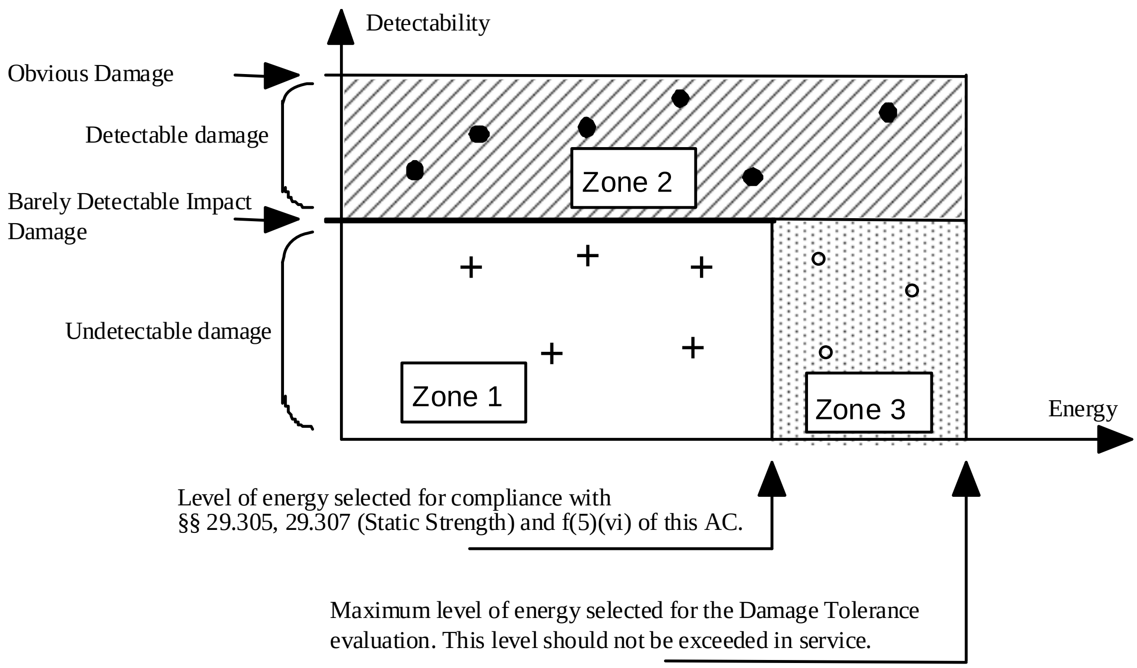

Figure AC 29.573-2. Characterization of Impact Damage.

(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 (BDID) 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(d) and (e) 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 it is established 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 for approval. In any case, the FAA 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, 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.

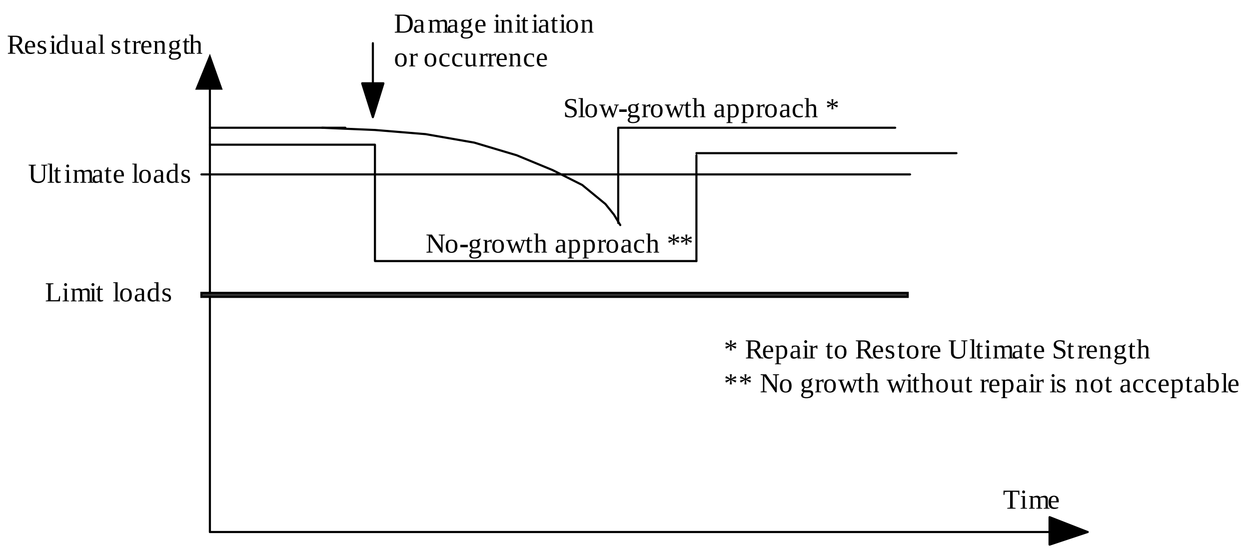

Figure AC 29.573-3. Residual Strength vs. Time.

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.

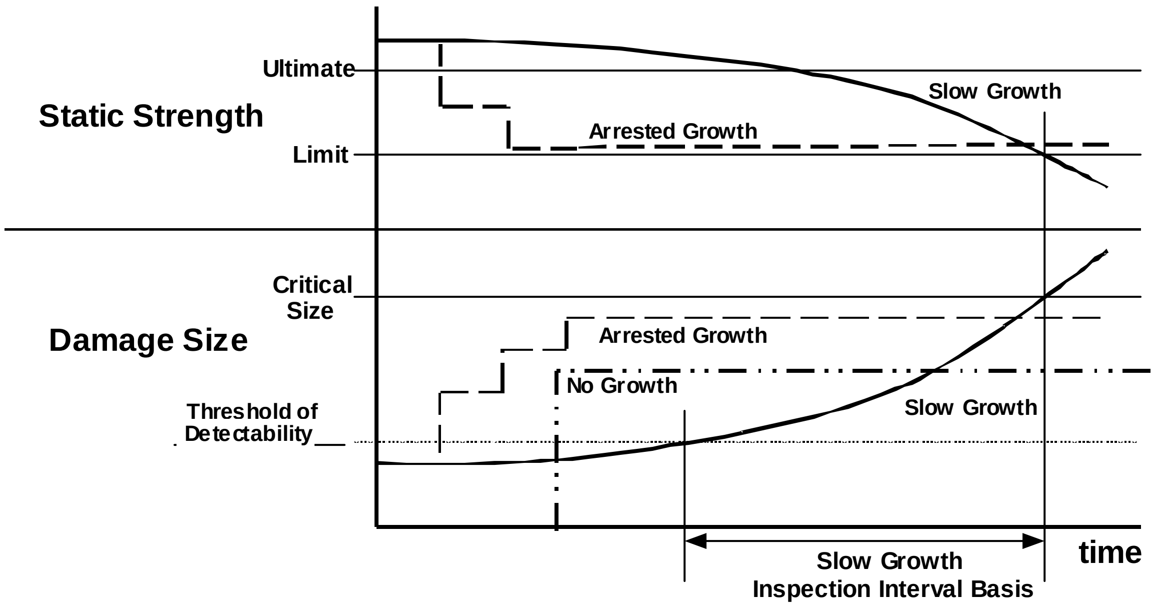

Figure AC 29.573-4. Illustration of Residual Strength and Damage Size Relationships for Fail-Safe Substantiation.

(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 § 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 FAA/AUTHORITY per § 29.573(e). The applicant must establish that the damage tolerance criteria are impracticable and 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.

(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; and

(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 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(d)(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 (e.g., 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 outside the original structural substantiation, 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 specific rotorcraft model involved in the added mission provided:

(i) changes are adopted through the airworthiness directives process and proper part re-identification is established; or

(ii) a Rotorcraft Flight Manual (RFM) supplement outlining the limitations is approved; or

(iii) an airworthiness limitations section (ALS) supplement is approved; or

(iv) 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, including 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.

(i) All vibration and resonance critical composite structures must be identified and properly evaluated.

(ii) All flutter-critical composite structures 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.

(iii) 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.