CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 8. (Amendment 29-42) SUBSTANTIATION OF COMPOSITE ROTORCRAFT STRUCTURE.

a. Reference. 14 CFR §§ 29.305, .307, .571, .603, .605, .609, .610, .611, .613, .629, .923, .927, .931, .1529 and Appendix A.

b. Purpose. These substantiation procedures provide a more specialized supplement to the general procedures outlined by AC 20-107A, "Composite Aircraft Structure." These procedures address 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, some variations and deviations from the procedures described herein will be both necessary and acceptable. Significant deviations from this material should be coordinated in advance with the FAA/AUTHORITY.

c. 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 is 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.

d. 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 composites 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, Chg 1, MG 11, "Fatigue Tolerance Evaluation of Transport Category Rotorcraft Metallic 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 specimens (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, Chg 1, 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., MIL-HDBK-5 for metals, MIL-HDBK-17 for composites, or 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 aeroplane.

(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 (reference paragraph a above) to be considered in substantiating composite rotorcraft structure, but this advisory material is needed to supplement the general guidance of AC 20-107A by providing specific rotorcraft guidance for obtaining consistent compliance with 14 CFR sections applicable to rotorcraft.

e. 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 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 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 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 local 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. The resins used in most aircraft structure are thermoset polymers.

(24) <u>MAXIMUM STRUCTURAL TEMPERATURE</u>. 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) <u>POROSITY</u>. 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.

(26) <u>PRE-PREG, PREIMPREGNATED</u>. 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.

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

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

(29) <u>SECONDARY BONDING</u>. 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) <u>SHELF LIFE</u>. 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/or remain suitable for its intended function.

(31) <u>STRAIN LEVEL</u>. 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 micro inches/per inch (e.g., .003 in/in equals 3000 micro inches/inch).

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

(33) <u>TAPE</u>. Hot melt impregnated fibers forming unidirectional pre-preg.

(34) <u>THERMOPLASTIC</u>. 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) CATASTROPHIC FAILURE. Any structural failure, which results in death, severe injury, or loss of the aircraft.

(39) FATIGUE TOLERANCE. The capability of structure to continue functioning without catastrophic failure after being subjected to fatigue (repeated) loads expected during operation of the rotorcraft. Fatigue tolerance should be achieved by flaw tolerance design, or if impractical, safe-life design, or a combination.

(40) SAFE-LIFE. The capability of as-manufactured structure as shown by tests, or analysis based on tests, not to initiate fatigue cracks during the service life of the rotorcraft or before an established replacement time.

(41) FLAW TOLERANCE. The capability of rotorcraft structure to achieve fatigue tolerance accounting for the presence of flaws and damage that may occur in manufacturing and service use. Flaw tolerance can be achieved by either flaw tolerance safe-life or fail-safe designs. The term "Damage Tolerance" is frequently used to describe the ability of a structure to tolerate the effects of flaws and damage; however, the terminology of § 29.571, Amendment 28, is used in this AC to maintain consistency.

(42) FLAW TOLERANT SAFE-LIFE. The capability of as-manufactured structure, with expected flaws, as shown by tests or analysis based on tests, not to initiate fatigue cracks or flaw/damage growth during the service life of the rotorcraft or before an established replacement time.

(43) FAIL-SAFE. The capability of structure remaining after a partial failure to withstand design limit loads without catastrophic failure within an inspection period.

(44) 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.

(45) ACTIVE MULTIPLE LOAD PATH. Structure providing two or more load paths that are all loaded during operation to a similar load spectrum.

(46) 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.

(47) ACCIDENTAL DAMAGE FLAWS. Discrete damage that may occur in service use or in manufacturing due to impacts or collisions, such as dents, scratches, gouges, abrasions, disbonds, splintering, and delaminations.

(48) MANUFACTURING-RELATED FLAWS. Intrinsic imperfections related to manufacturing operations, processing, or assembly such as voids, gaps, porosity, inclusions, fiber dislocation, disbonds, and delaminations.

(49) FATIGUE/ENVIRONMENTAL FLAWS. Structural damage related to fatigue or environmental effects such as delaminations, disbonds, splintering, or cracking.

(50) DESIGN LIMIT LOADS. The maximum loads to be expected in service, as defined by § 29.301(a).

(51) 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.

(52) RESIDUAL STRENGTH. The strength retained for some period of unrepaired use after a failure or partial failure due to fatigue or accidental or discrete source of damage.

(53) 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.

(54) 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).

(55) POINT DESIGN. An element or detail of a specific design that 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.

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

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

(58) 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).

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

(60) ENVIRONMENT. External, nonaccidental 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).

f. 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, Chg 1, 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, Chg 1, MG 11	Fatigue Tolerance Evaluation of Transport Category Rotorcraft Metallic Structure (Including Flaw Tolerance)
DOT/FAA/CT-86/39	Whitehead, R.S., Kan, H.P., Cordero, R., and Seather, R., “Certification Testing Methodology for Composite Structures”, October 1986.

g. 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) fatigue tolerance evaluation (including tolerance to flaws); (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. This approach will minimize the risk associated with substantiation of the full-scale article. It is strongly recommended that a certification team approach, involving fabrication, quality, and engineering specialists from both the applicant and FAA/AUTHORITY, be used for composite structural substantiation.

The team should assure that permanent documentation of the building block approach 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). FAA/AUTHORITY engineering approves the Airworthiness Limitations Section of the maintenance manual. Engineering practices for many of the areas identified below are available in MIL-HDBK-17.

(1) The first area is the fabrication requirements of § 29.605:

(i) The quality system should be developed considering the critical engineering, manufacturing, and quality requirements along with 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 disbonds , 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 worst-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) of this AC paragraph) 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 and 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 systems provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, such 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 under their approved production inspection, fabrication inspection, and quality systems .

(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, such 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 under their approved production inspection, fabrication inspection, and quality systems.

(D) Statistical Validation Level. It is necessary to maintain the minimum required statistical validation level of the quality 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 and FAA/AUTHORITY-approved 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) 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. Although MIL-HDBK-17 does have some "B" basis allowables available in Volume 2, company testing is required for "A" basis and other "B" basis material systems not listed. 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 approved and 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 (g)(1) above).

(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 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. In actuality, 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 (matrix) is the critical structural constituent in a composite 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 in paragraph e(24) of this AC paragraph). 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 nucleation 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 such 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 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 seen to be extremely important.

(vii) Composite and adhesive properties should be determined such 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 a 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 certification 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 20-53A and FAA Report DOT/FAA/CT-86/8. For composite structure projects involving rotorcraft certified 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 certification 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 not 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 paragraph g(2) of this AC paragraph) and protection of structure (reference paragraph g(3) of this AC parapraph). Static strength tests should be conducted for substantiation of new structure. For the critical loading conditions, two approaches to account for prior repeated loading and 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 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 MG 8-1 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 MG 8-1, 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).

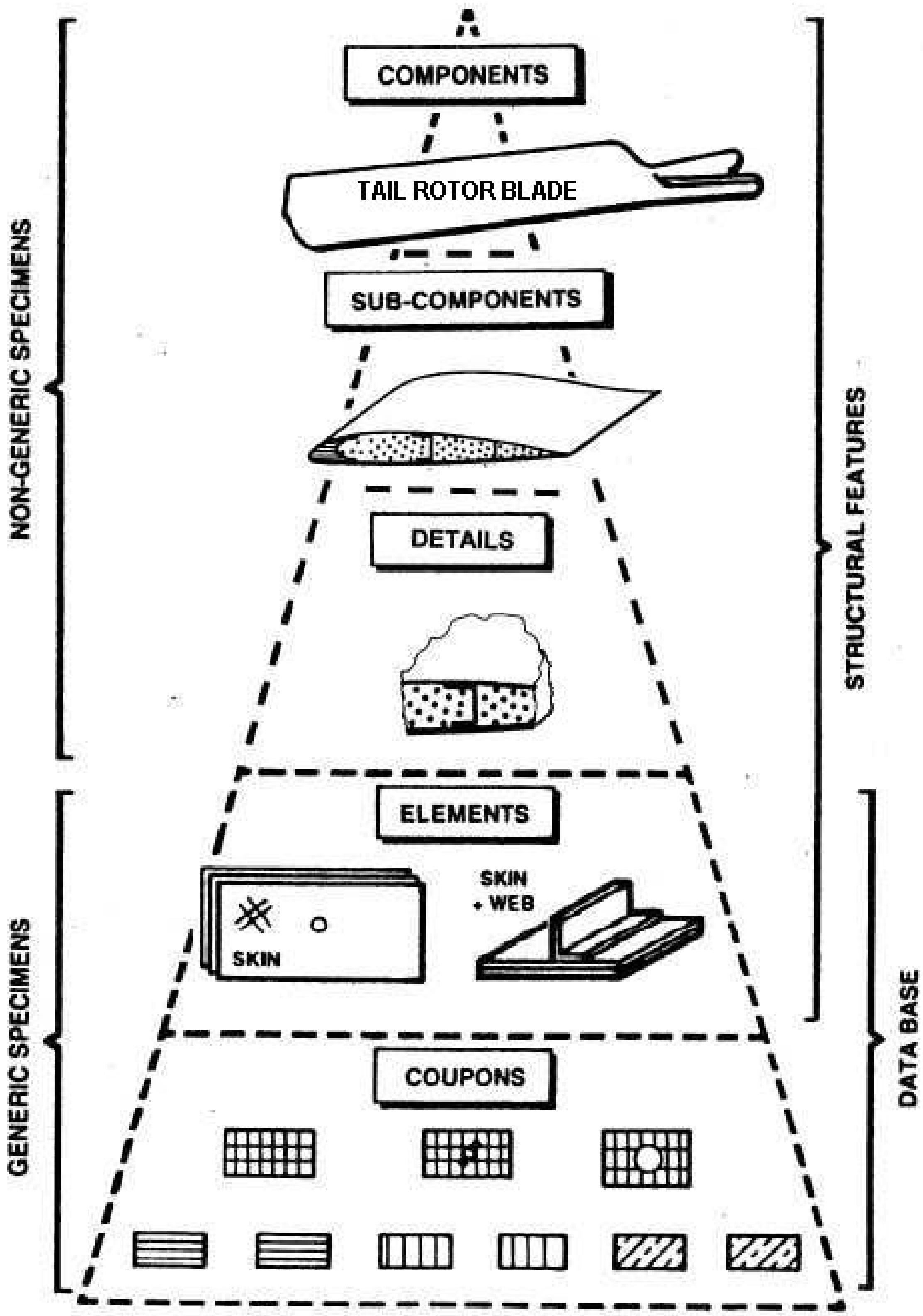

Figure AC 29 MG 8-1. Schematic diagram of building block tests.

(iii) Allowables should be 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 paragraph address the issues associated with damage in excess of that considered in g(5) of this AC paragraph and drops in residual strength below ultimate load capability (reference paragraph g(6)) below.

(6) The <u>sixth</u> area is the <u>fatigue evaluation</u> of structure requirements of § 29.571.

(i) FATIGUE EVALUATION - 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 fatigue tolerance evaluation required by § 29.571 establishes 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 procedures established by the fatigue tolerance evaluation include component retirement times and/or inspection intervals. The fatigue tolerance evaluation requires a flaw tolerance assessment that assumes that the baseline ultimate strength capability might be compromised by damage caused by fatigue, environmental effects, intrinsic discrete flaws, or accidental damage. The flaw tolerance assessment establishes procedures that do not allow the static strength capability to degrade below the ultimate strength capability for extended periods, assuming such damage occurs within the operational life of the structure. When this damage occurs, the remaining structure will withstand reasonable loads without failure or excessive structural deformations until the damage is detected and the component is either repaired to restore ultimate load capability or retired.

(ii) FLAW TOLERANCE EVALUATION - GENERAL. The nature and extent of the required analysis or tests on complete structures or 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 or 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.

(C) 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 numbers 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 of the following or a combination thereof.

(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 acceptable, provided there is satisfactory service history of like or similar components.

(D) 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.

(E) Environmental effects (temperature and humidity representative of the expected service usage) on the static or 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. For example, typical hot-wet environmental test criteria are 180º F +/- 5º F for temperature and 85% +/- 5% for relative humidity.

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

(G) The following Figure AC 29 MG 8-2 illustrates the extent of the impact damage that needs to be considered in the flaw tolerance evaluation.

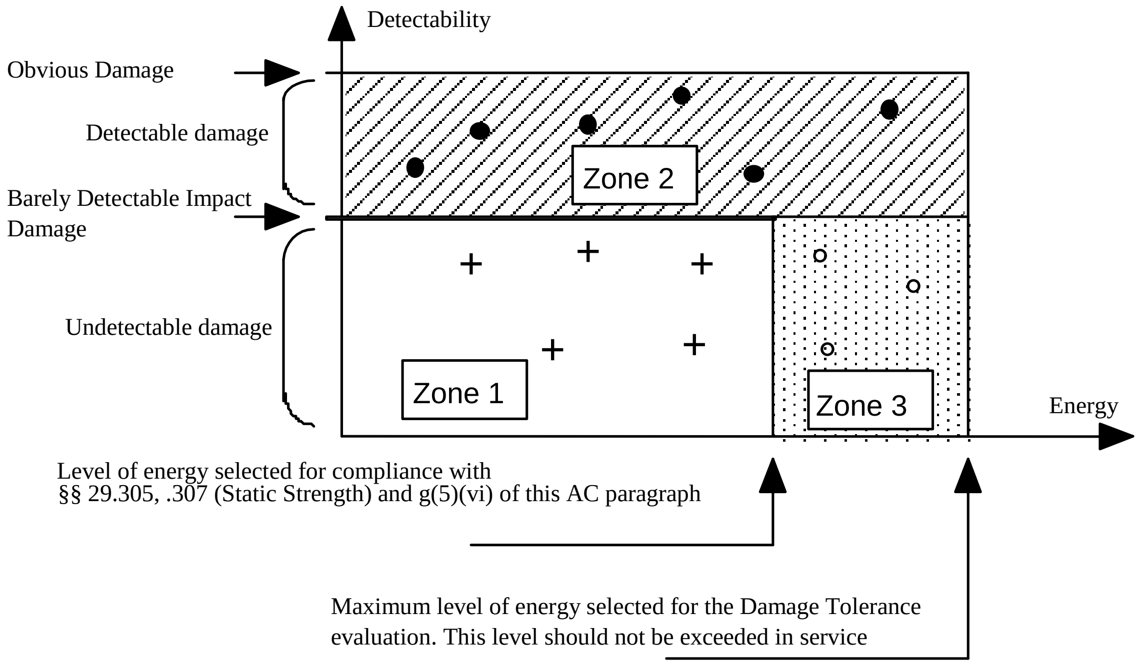

Figure AC 29 MG 8-2 Characterization of Impact Damage

(1) Both the energy level associated with the static strength demonstration and the maximum energy level associated with the damage tolerance evaluation (defined in the figure above) are dependent on the part of the structure under evaluation and a threat assessment.

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

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

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

(5) 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.

(6) Three Zones are defined by Figure AC 29 MG 8-2:

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

- Zone 2: Since the damage can be detected at scheduled inspection, Limit Load (considered as Ultimate) 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 an improved inspection procedure.

(iii) FATIGUE TOLERANCE EVALUATION – MEANS OF COMPLIANCE. One, or a combination of, the methods below should show compliance with the requirements of this section. The Flaw Tolerant Safe-Life Evaluation or the Fail-Safe Evaluation are to be used unless it can be shown that neither can be achieved within the limitations of geometry, inspectability, or good design practice. In that case, the Safe-Life Evaluation should be used. From current state-of-the-art with rotorcraft applications, it is widely admitted that composite materials have good flaw or damage tolerance capabilities and therefore the safe-life option is rarely necessary. Flaw Tolerance evaluations are best suited for most 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 Flaw Tolerant Safe-Life and Fail-Safe evaluations.

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 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. Flaws and damage as determined in paragraph g(6)(ii) above for the specific structure being substantiated should be imposed at each critical area of the structure.

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

No specific inspection requirements are generated from the test program in this method. However, routine inspections for cracking, delaminations, and service damage as outlined in § 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 flaws present at critical locations in the structure. AC 27-1B, Chg 1, MG-11, provides guidance that can be appropriate for this method in composites. 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 flaws and defects, or structurally significant cracking, disbonding, splintering or delamination 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 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 flaw 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. Please reference "Certification Testing Methodology for Composite Structure", Report No. DOT/FAA/CT-86/39 for a discussion of the S-N approach. Any significant growth of the imposed flaws and defects, 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 flaw 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) Fail-Safe (Residual Strength after Flaw Growth) Evaluation. This method demonstrates that the structure following a partial failure still has a sufficient residual strength capability within a specified inspection interval or the established retirement life of the component. If a retirement life is established, an ultimate design load capability is generally required while, if an inspection interval is determined, a limit load capability is the minimum acceptable residual strength capability that needs to be demonstrated. 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 of the three methods described below or a combination thereof.

(1) Fail-Safe, No Growth Evaluation. This approach is appropriate for inspectable in-service accidental damage. Structural details, elements, and sub-components of critical structural areas, components, or full-scale structures, should be tested under repeated loads for validating a no-growth approach to the flaw 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. Residual strength testing or evaluation should be performed after repeated load cycling and demonstrate 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 in the Figure AC 29 MG 8-3. Once the damage is detected, the component is either repaired to restore ultimate load capability or replaced.

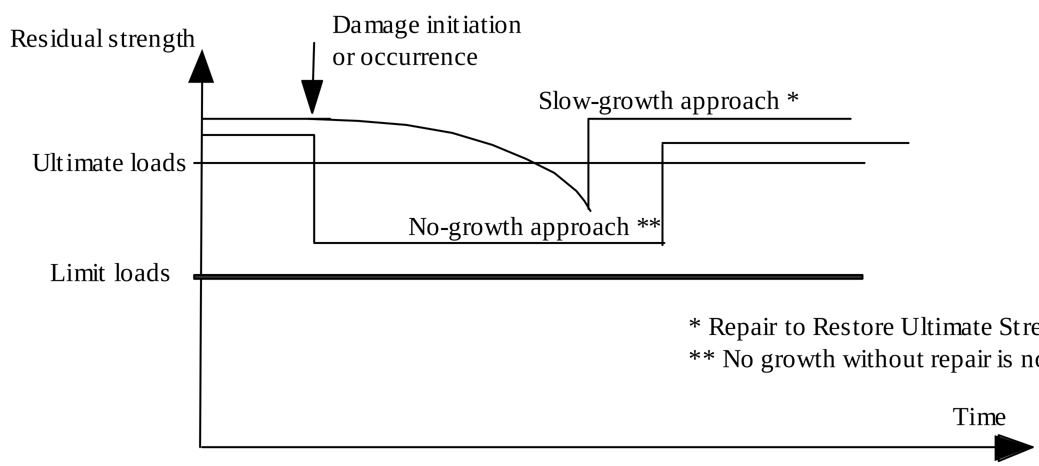

Figure AC 29 MG 8-3. Residual Strength vs. Time

The lower the residual strength caused by 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 flaw grows in the test and the growth rate is shown to be slow, stable, and predictable, as illustrated in Figure AC 29 MG 8-4. 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 such 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.

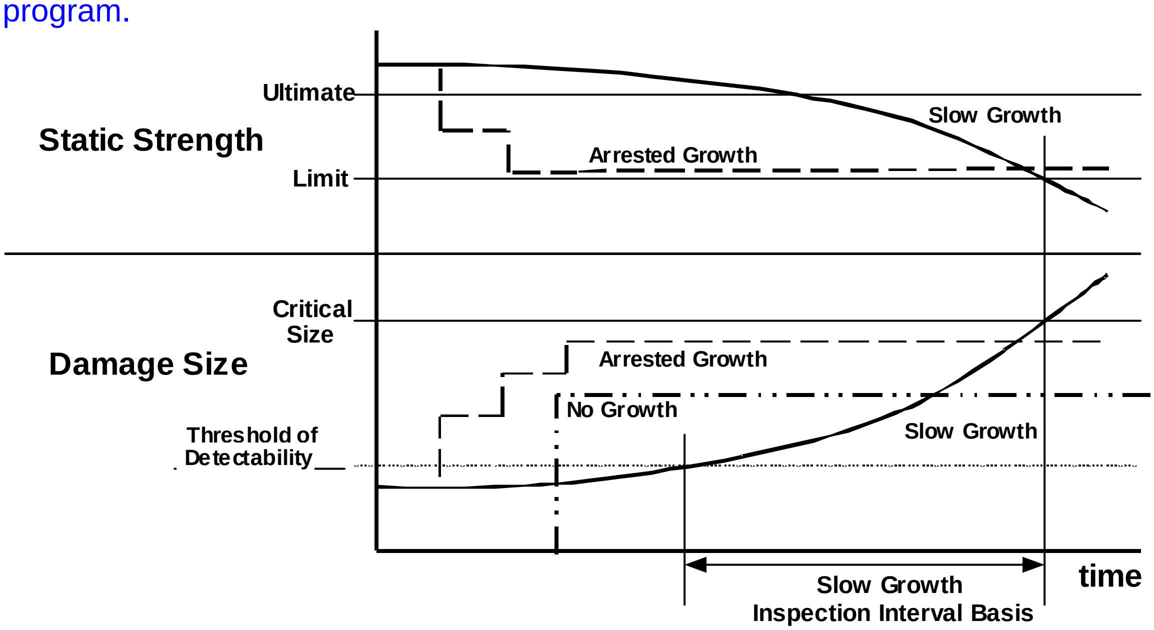

Figure AC 29 MG 8-4.
Illustration of Residual Strength and Damage Size Relationships for Fail-Safe Substantiation.

(3) Arrested Growth Evaluation. This method is applicable when the flaw 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 MG 8-4. 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, components 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 and/or life considerations. Residual strength testing or evaluation should be performed after repeated load cycling and demonstrate 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 ensure 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 MG 8-3. For any damage size that reduces load capability below ultimate, the component is either repaired to restore ultimate load capability or replaced.

The lower the residual strength caused by 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) Safe-Life Evaluation. This method demonstrates that the structure, in an as-manufactured condition, is able to withstand repeated loads of variable magnitude without detectable cracks, disbonds, or delaminations for the life of the rotorcraft or within a specified retirement time. It is available for use only when both the Fail-Safe and Flaw Tolerant Safe-Life methods have been shown to be impractical due to considerations of geometry, inspectability, or good design practice. Further guidance for Safe-Life substantiation is provided in AC 27-1B, Chg 1, MG-11, "Fatigue Evaluation of Rotorcraft Structure". The fatigue 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.

(D) Combination of Safe Life and Fail Safe Evaluations. Generally it may be appropriate to establish both a retirement time and an inspection program for a given structure.

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

(A) Experience with the application of methods of fatigue and flaw 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 flaw 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).

(<u>1</u>) <u>Identification of Principal Structural Elements.</u> 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:

(<u>i</u>) Rotor blades and attachment fittings.

(<u>ii</u>) Rotor heads, including hubs, hinges, and some main rotor dampers.

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

(<u>iv</u>) Rotor supporting structure (lift path from airframe to rotor head).

(<u>v</u>) Fuselage, including stabilizers and auxiliary lifting surfaces.

(<u>vi</u>) Main fixed or retractable landing gear and fuselage attachment structure.

(<u>2</u>) <u>Identification of Locations Within Principal Structural Elements to be Evaluated</u>. 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:

(<u>i</u>) Strain gauge data on undamaged structure to establish points of high stress concentration as well as the magnitude of the concentration.

(<u>ii</u>) Locations where analysis shows high stress or low margins of safety.

(<u>iii</u>) 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 failure locations in an element where high stress concentrations are present in the remaining structure.

(vii) Locations where detection would be difficult.

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

(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 g(6)(iv)(C) above, the actual flight condition occurrences of paragraph g(6)(iv)(D) above, and the fatigue strength material properties of paragraph g(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 and 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, Chg 1.)

(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 define the load spectrum adequately for each part essential to the safe operation of the rotorcraft as identified in § 29.571(a)(1)(i). 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 Attachment 1 to paragraph AC 29 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 maximum likely 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 fatigue 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, rotorcraft 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 manufacturer's technical support is provided and documented; i.e., the manufacturer either provides the factorization analyses or checks them on a continuing basis for each rotorcraft.

(4) Where one or more of the above 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/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 § 29.1529.

(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) Proper part re-identification is established;

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

(iii) an airworthiness limitations section 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 such 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. 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 that govern V_c or cruise speed. Where possible, it is desirable to conduct the flight strain-gauge program by simulating the usage as determined above, with continuous recording of stresses and loads, thus obtaining directly the stress and 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 are to 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. 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 structure are identified and properly substantiated.

(ii) All flutter-critical composite structures are identified and properly substantiated. 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. Please reference §§ 29.562 and 29.952.

(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 must be shown to provide structure that continually meets the guidance of paragraphs (1) through (7) of this AC paragraph. All certification based repair and continued airworthiness standards, limits, and inspections must be clearly stated and their provisions and limitations defined and 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 representative (DER or staff engineer).