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

Advisory Circular

Subject: GUIDANCE MATERIAL FOR FATIGUE LIMIT TESTS AND COMPOSITE BLADE FATIGUE SUBSTANTIATION

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Change:

1. **PURPOSE**. This advisory circular (AC) provides guidance and describes methods, but not the only methods, for demonstrating compliance with §35.37 of Title 14 of the Code of Federal Regulations, Fatigue limit tests. This AC also provides methods for the fatigue evaluation of propellers and the fatigue substantiation of composite propeller blades. Like all AC material, this AC is not, in itself, mandatory and does not constitute a regulation. While these guidelines are not mandatory, they are derived from extensive Federal Aviation Administration (FAA) and industry experience in determining compliance with the applicable regulations.

2. **CANCELLATION**. AC 35.37-1, Composite Propeller Blade Fatigue Substantiation, dated 5/11/93, and AC 35.37-1, Change 1, dated 9/7/93, are canceled.

3. **RELATED DOCUMENTS**.

a. Regulations. Sections 35.4, 35.37, 23.907, and 25.907.

b. Advisory Circulars.

(1) AC 20-66A, Vibration and Fatigue Evaluation of Airplane Propellers, dated 9/17/01.

(2) AC 20-107A, Composite Aircraft Structure, dated 4/25/84.

(3) AC 21-26, Quality Control for the Manufacture of Composite Structures, dated 6/26/89.

(4) AC 25.571-1C, Damage Tolerance and Fatigue Evaluation of Structure, dated 4/29/98.

c. Military Specifications. MIL-HDBK-17-1C, dated November 4, 1992.

4. **DEFINITIONS**. For the purposes of this AC, the following definitions apply.

a. Damage tolerance. Damage tolerance is 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 damage from fatigue, corrosion, accident, or a discrete source.

b. End of life condition. The end of life condition is the physical condition of the component when it has sustained the maximum extent of damage but maintains sufficient residual strength to meet all airworthiness loading requirements. This condition is defined during certification.

c. Fail-safe. Fail-safe is the attribute of the structure that permits it to retain its required residual strength for a period of use without repair after the failure or partial failure of a principal structural element.

d. Fixed pitch wood propellers of conventional design.

(1) A fixed pitch wood propeller of conventional design is a propeller that has the following physical properties:

(a) One piece laminated wood construction;

(b) Two or four blades;

(c) Surface coatings that do not contribute significantly to the propeller strength; and

(d) Surface coatings that only provide environmental protection.

(2) A fixed pitch propeller that has a composite shell over a wood core does not qualify as conventional design if the composite shell contributes significantly to the strength and frequency response of the propeller.

(3) A fixed pitch wood propeller with a fabric or composite covering for environmental protection that does not significantly alter the structure qualifies as conventional design.

e. Flaw. A flaw is a pre-existing anomaly in the structure that is either created during manufacture or has resulted from damage to the structure introduced after manufacture. Generally, flaws can be geometrically quantified by non-destructive inspection methods. The term flaw is widely used in fracture mechanics disciplines for assessing damage tolerance.

f. Hazardous propeller effects. The following conditions are considered hazardous propeller effects:

(1) A significant overspeed of the propeller.

(2) The development of excessive drag.

(3) Thrust in the opposite direction to that commanded by the pilot.

(4) A release of the propeller or any major portion of the propeller.

(5) A failure that results in excessive unbalance.

(6) The unintended movement of the propeller blades below the established minimum in-flight low-pitch position.

g. Life factor. This is a factor applied to increase or decrease the number of cycles on an S-N diagram as needed to properly account for test sample size and variations in material properties, environmental effects, service deterioration, repairs, and manufacturing anomalies. This is often used in combination with a load factor, as shown in Figure 1.

h. Limit load. This term refers to the maximum load expected in service.

i. Load factor. The load factor is a factor applied to increase or decrease load or stress level on a fatigue life diagram (S-N diagram) as needed to properly account for test sample size and variations in material properties, environmental effects, service deterioration, repairs, and manufacturing anomalies. This is often used in combination with a life factor, as shown in Figure 1.

j. Principal structural element. This term refers to the element that contributes significantly to the carrying of propeller loads and whose integrity is essential in maintaining the overall structural integrity of the propeller.

k. Safe-life. The safe-life of a structure is that number of events such as stress cycles, flights, landings, or flight hours, during which there is a low probability that the strength will degrade below its design value due to fatigue damage.

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

m. Stress ratio (R). For repeated stress cycles, this is the ratio of the minimum stress to the maximum stress, $R = \sigma_{min} / \sigma_{max}$.

5. **DISCUSSION**. Propellers are continuously subjected to steady and vibratory stresses under many different operating conditions on an airplane, both while in flight and on the ground. Therefore, §35.37 requires a fatigue evaluation and the determination of fatigue limits. This requirement also provides data that supports installation of the propeller on an airplane. Since the rate of accumulation of stress cycles for propeller blades, hubs, and other propeller components is very high, the design goal, whenever possible, should be to show that stresses are below the component or material endurance limit. However, not all materials have a well-defined endurance limit. In addition, the stresses developed during maneuvers, ground operation, ground air ground (GAG) cycles or in other areas of the airplane operating envelope may cause damage. The accumulation of this damage should be evaluated to determine if propeller components are life limited or require mandatory inspections, or to determine if the propeller is suitable for use on an airplane.

a. Fatigue Limits. Establishing propeller fatigue limits is one step in the structural evaluation for certification and approval for use on an airplane. Fatigue limits refer to the material and structural databases used to support the fatigue evaluation of the propeller on an airplane. Fatigue limits may take many forms, such as Goodman diagrams to assess whether or not stresses are below the endurance limit, S-N curves for safe-life evaluations, and crack growth curves (da/dn curves) and delamination growth curves for damage tolerance evaluations. The fatigue limits are developed with an understanding of the propeller component, material, failure mechanism, environment, and loading and may also include the effect of service damage. When determining the fatigue limits under §35.37, the possible airplane operating environment should also be considered. The development of fatigue limits can be supported by previous testing, past experience, and acceptable published data, when available.

b. Damage Accumulation. The fatigue limits are developed to support a damage accumulation algorithm, such as Miner's rule for safe-life calculations or a crack growth or damage growth algorithm for damage tolerance calculations. The damage accumulation algorithm is used for the fatigue evaluation required by §35.37 and may be also used as part of showing compliance with the airplane vibration requirements in §§23.907 and 25.907. The damage accumulation algorithm should be verified by previous testing, past experience, and acceptable published data, when available.

c. Applicable Components.

(1) Components. For propeller certification, fatigue limits are established for the propeller hub, blades, and blade retention. They should also be developed for those components whose failure due to fatigue may cause a hazardous propeller effect. Examples of components whose failure may cause a hazardous propeller effect are the pitch change piston pressure cylinder (dome), counterweights, and pitch control components. For items such as bearings, which are typically part of the hub assembly, the fatigue limits are generally established for the assembly, not for the individual components.

(2) Regions. Each applicable component should be assessed to determine if it has multiple critical regions or if it requires different limits for different regions. Blades are an example of a component that has different limits for different regions. The blade steady stresses vary substantially from the blade root to the blade tip; the blades should have fatigue limits established for each of the stress regions. Also, metallic blades may have different material limits to account for local working effects due to shot peening or cold rolling.

d. Propeller Loads. The loads applied during the fatigue limit tests are derived from the consideration of the steady and vibratory propeller loading conditions that occur on the intended airplane and engine installation throughout its life or on a typical airplane. The loads applied during fatigue limit tests reflect the fatigue data to be generated. For example, the loads establish the stress ratios (R) for coupon specimen tests and set the amplitude and direction for full-scale testing. Also, the load magnitude and direction should be established in a manner that represents the loading the propeller will experience in service.

(1) Operating Spectrum Conditions. The applied loads are derived from an airplane operating spectrum. The operating spectrum depends on the category and operation of the airplane and includes both flight and ground operation.

(a) Flight conditions include conditions that occur with each flight such as:

1. Take-off;

2. Climb;

3. Cruise;

4. Descent;

5. Approach;

6. Landing; and

7. Reverse thrust.

(b) Transient airplane flight conditions are associated with maneuvers such as:

1. Banked turns;

2. Side-slip;

3. Pull-ups;

4. Push-overs;

5. Rudder kicks; and

6. Gusts.

(c) Special flight conditions are specific to a mission such as:

1. Fire-fighting;

2. Aerobatic maneuvers;

3. Emergency conditions; and

4. Training maneuvers.

(d) Ground conditions include but are not limited to:

1. Taxi;

2. Operation in cross-winds; and

3. Maintenance checks.

(e) Engine load conditions include the loads generated by the engine and transmitted to the propeller, such as firing impulses from reciprocating engines.

(2) Operating Spectrum Source. The airplane operating spectrum, when available, may be obtained from the airplane company for the intended application. When the airplane operating spectrum is not available, the spectrum may developed from design assumptions and design experience on a typical or on the intended airplane.

e. Component Degradation. The fatigue limits should account for likely service deterioration, variations in material properties, manufacturing anomalies, and environmental effects. Many methods are available to account for component degradation, such as using a life or load factor to reduce the fatigue limits or apply to fatigue test loads. Reductions in fatigue limits may be applied as life or load factors based on the manufacturer's service experience and test database with the components, as shown in Figure 1. The reductions may also be developed by appropriate specimen tests or by testing intentionally degraded full-scale components.

(1) Manufacturing Anomalies. Propellers manufactured to a process specification have established parameters to control manufacturing variables. These parameters should be considered for their impact on structural integrity.

(a) For metal propellers these include, but are not limited to, surface finish and machining marks.

(b) For composite propellers these include, but are not limited to, fiber misalignment, fiber content, porosity, and delaminations.

(c) For adhesive bond joints these include, but are not limited to, adhesive thickness, voids, and disbonds.

(2) Service Damage. Propellers are exposed to and encounter a certain amount of service damage from corrosion, erosion, stone strikes, and handling damage. Composites, in addition, may be damaged by small bird strikes and hail impact. The damage to a composite should consider both visible surface damage and hidden internal damage, such as delaminations. The extent of tolerance to such damage and the method of demonstration should be considered.

(3) Environmental Degradation. Airplanes can spend their entire operational life in a severe climatic zone. Therefore, the strength degradation of the material system should be established. When assessing operational environments, the following should be considered:

(a) High temperature;

(b) Humidity;

(c) Low temperature;

(d) Thermal cycling;

(e) Ultraviolet light; and

(f) Aviation chemicals.

f. Repairs. Repairs change the condition of the propeller as manufactured. Therefore, the manufacturer should consider the impact of planned repairs on the structural integrity of the propeller.

g. Endurance limit. When conducting coupon tests, the coupons should be tested to failure under a combination of steady and vibratory loads. The steady loading should represent that anticipated in service for the propeller. The vibratory loading component should be selected to facilitate the generation of a fatigue or S-N curve, over a broad range of stress cycles, to define an endurance limit.

(1) The endurance limit for composites is normally projected to at least 500 million cycles.

(2) The endurance limit for aluminum is normally projected to at least 100 million cycles.

h. Accelerated Testing. The vibratory loading may be at any frequency, if internal heating does not produce harmful effects and representative failure modes are realized. Attempts to compress testing time by using high-frequency vibratory loading should be approached with caution, to prevent the introduction of non-representative temperatures, failure modes, and fatigue lives. Accelerated testing should include some form of temperature monitoring to prevent overheating the specimen. Cooling may be required to prevent overheating the component.

i. Coupon Tests. Establishing an S-N curve from coupon tests is recommended for determining fatigue strength and statistical strength distribution of propeller materials. The S-N curve should be established for both the low cycle fatigue (LCF) and high cycle fatigue (HCF) regimes. The S-N curve should be representative of the propeller's material system. S-N curves should be developed using acceptable published data, when available, or by testing a sufficient number of specimens that represent the propeller's material system manufacturing processes.

(1) Metals. For metals, this should include the following effects:

(a) Surface finish;

(b) Cold rolling;

(c) Shot peening; and

(d) Corrosion inhibitors.

(2) Composites. For composites, this should include the following effects:

(a) Fiber, resin and sizing;

(b) Ply stacking sequence;

(c) Ply orientation; and

(d) Manufacturing processes.

(3) Bond Joints. For bond joints, this should include the following effects:

(a) Adhesive materials;

(b) Surface preparation;

(c) Adhesive thickness; and

(d) Manufacturing processes.

(4) Propeller Stations. The propeller's station-to-station variance between metallic to composite bond regions, shank sections, mid-blade section, and tip regions may warrant the testing of coupons representative of each.

j. Full-scale Testing.

(1) Test Specimen. Full-scale components should be manufactured to represent the type design and should be fatigue tested at combinations of steady and vibratory loads as needed to support subsequent evaluations. Also, to address the issues of safe-life, damage tolerance, and continued airworthiness, the specimens should be manufactured to include specific manufacturing anomalies and likely service damage. The extent of such defects/damage should be consistent with inspection techniques employed in service.

(2) Failure Criteria. Rational failure criteria should be established. This will be different for various types of construction and materials. Composites tend to fail between layers first, while metals fail through the thickness of the part and have much smaller permissible defect sizes.

(a) For composites, as shown in Figure 2, fatigue failure is markedly progressive, starting at initiation (region I) through a damage growth or delamination growth phase (region II) to an advanced damage state in which large delamination and secondary failure modes and locations form throughout the component (region III). Failure criteria such as a specific loss of stiffness in the structure, visible damage, and delamination area should be applied.

(b) For metallic components, the failure criteria may be the initiation of a fatigue crack or a specific crack length. The selected failure criteria will be a factor in establishing the component life, since the component should maintain sufficient residual strength throughout its life.

(c) When using stiffness loss as failure criteria, the effect on component resonant frequency placement should be addressed, as this may affect the continued airworthiness of the component. Vibratory loads may be magnified as component natural frequencies approach the frequency of the load.

(3) Component Monitoring. Components should be strain gaged and have load cells, as appropriate, for testing to monitor stresses and loads. When a test rig applies load with an amplitude actuator, the input amplitude may require adjustment to account for test rig and/or component wear in or degradation, to assure that the test load is maintained. The specimen should be examined regularly for cracks, delamination, or other degradation. When using stiffness as the failure criteria, it should be measured for each specimen at the outset of each test and periodically monitored throughout the test. The monitoring should be more frequent during the failure process. When appropriate, testing should be continued after the component has failed, according to the failure criteria, to demonstrate damage growth characteristics and residual strength capacity. This may include testing to the limit load conditions after LCF or HCF tests are completed to demonstrate residual strength at the end of life condition.

k. Data Reliability.

(1) Mean S-N Diagram. The mean trend line for the data for each critical location should be defined from the mean steady and vibratory stress for all coupons/components tested. This mean trend line should be extrapolated out to an asymptote representing the HCF endurance limit and extrapolated back to a low number of cycles. The coupon-derived S-N curve shape may be used to supplement extrapolation through the failure points when the coupon tests duplicate the failure mechanism.

(2) Reliability. As dynamically loaded components, the structural integrity of propellers is generally governed by their fatigue, rather than their static strength. Accordingly, a reliability at least as good as the "A" basis of a normal distribution should be demonstrated. That is a reliability of 99 percent, with a 95 percent confidence level.

(3) Number of Test Specimens. There is no set number of full-scale test specimens that should be fatigue tested for each critical propeller section, if the required reliability is satisfied.

(a) For "A" basis reliability, the mean endurance limit (E_{50}) should be reduced by a factor (k), governed by the selected sample size (n) and the standard deviation (σ) data. Specifically, the 99 percentile endurance limit (E_{99}) may be expressed: $E_{99} = E_{50} - k\,\sigma$; in which (k) is a function of the sample size (number of specimens tested). The "A" basis values may also be defined by other distributions such as Weibull or Log Normal, depending upon the fit. Refer to MIL-HDBK-17-1C for details.

(b) For the normal strength distribution, 0.95 confidence, (k) varies with sample size as follows:

Sample size	Normal Distribution Reduction Factor
n	k
2	37.094
3	10.553
4	7.042
5	5.741
6	5.062
7	4.642
8	4.354
9	4.143
10	3.981
REF. Table 8.812, MIL-HDBK-17-1C	

(c) The manufacturer's in-house methodology, development data, and analytic expertise, along with limited full-scale testing, may be used as an alternative in substantiating the required reliability.

(4) Working Curve. The working curve is the design curve. It is developed from the "A" basis curve by dividing by appropriately justified load or life factors to account for the manufacturing variations, environment, and service conditions. Factors for composite materials should be developed for each new composite material. These materials include, but are not limited to, fiber, resin, coating, and adhesives.

6. **COMPOSITE COMPONENT FATIGUE LIMITS**.

a. Discussion. An increasing number of propeller blades are constructed of composite materials. Due to the anisotropic characteristics of composite materials, component design and verification of design differ significantly from that of metal structures; propeller blade fatigue loading differs significantly from most other structures. Past practices for metal structures or other composite structures may not be adequate when dealing with composite propeller structure and fatigue loading. Although this discussion focuses on propeller blades, the principles may be applied to any composite structure. With the added flexibility of composite component design, damage tolerance and fail safe principles should be included whenever possible. This AC assumes that when damage tolerance methods are applied the component has been designed using damage tolerance and fail-safe principles.

(1) Failure Progression. Figure 2 illustrates the typical progression of fatigue degradation and reduction in residual strength of a typical composite component. Region I is the damage initiation phase. Region II is the damage growth or delamination growth phase; this is the region in which delamination growth propagates steadily and predictably. Region III is an advanced damage state in which large delamination and secondary failure modes and locations form throughout the laminate. In region III, component stiffness and residual strength degrade rapidly. This AC provides guidance for fatigue limits to be used in a fatigue evaluation, to assure that region III is not reached in service.

(2) Fatigue Limits. Fatigue limits should be developed to support a fatigue evaluation based on a safe-life approach or a damage tolerance approach. This AC provides guidance for both of these approaches. When appropriate, the fatigue limits should address permissible repairs that alter the as manufactured condition of the propeller.

b. Safe-Life Fatigue Limits. Developing the fatigue strength characteristics for composite components to be used in a safe-life-fatigue evaluation should be done with a two phase approach. The first phase is coupon specimen testing for both static properties and fatigue properties, to supplement the fatigue (S-N) curve shape, standard deviation, and statistical distribution. The second phase is full-scale specimen testing to establish the strength level, statistical distribution, failure location, and failure mode and mechanism for the material system and geometry of the propeller. For blades this may involve testing of metallic to composite bonds, shank, mid-blade, and tip specimens. Figure 3 illustrates some types of test rigs that have been used to assess various areas of a blade. The development of the fatigue properties may be supported by previous testing, past experience, and acceptable published data, when available.

(1) Loading.

(a) Two types of loading may be needed for testing full-scale composite structures. Constant loading defines the S-N curve. Spectrum loading addresses cumulative loading effects, since damage is dependent on both LCF and HCF loads. These loads include GAG cycles, low occurrence high amplitude maneuver limit loads, and high occurrence take-off, climb, and cruise loads. The fatigue characteristics may also depend on the order in which the loads are applied; this should be considered when establishing the test load spectrum. This effect should be addressed by performing many sets of load blocks.

(b) LCF and HCF are closely related; both of these should be addressed. Full-scale testing of composite components usually emphasizes LCF initially, since composite S-N diagrams have a shallow slope. However, unless the coupons duplicate the mode of damage, coupon tests have limited value in areas with complex geometry or metallic to composite bonds.

(c) Constant amplitude HCF failures are generated in the 10^6 to 10^8 cycle range. To induce failures in this cycle range, the alternating component of the load(s) is generally increased by a load factor from design or operating values. The steady component of load is generally not increased to induce failures. If no failure occurs, a no failure point is generated. No failure points are known as run-out points. Specimens that have completed a run-out should not be re-tested to generate failure points at higher load levels. The run-out specimen may have accumulated damage during previous testing that could bias the test results. A new specimen should be used for testing at a higher load level to generate a failure point.

(d) Constant amplitude LCF failures are generated in less than 10^6 cycle range. Failures may be induced by increasing the number of cycles by a life factor on the applied load or by increasing the vibratory component of the applied load by a load factor, or a combination of both. The steady component of load is generally not increased to induce failures. Subsequent data evaluation should account for the cycle or load increase required to generate the failures. Use load factors with caution, as the failure mode may change while testing at increased load.

(e) Spectrum load testing is often used to verify the damage accumulation algorithm for subsequent fatigue evaluations. Miner's rule, with appropriate consideration for scatter factor, has proven to be suitable for calculating component life. A load spectrum includes operating conditions representative of the intended operation conditions for the propeller. Within this spectrum a high amplitude low cycle maneuver load should be applied periodically. Life factors to account for scatter should be applied instead of load factors, which may alter the failure mode. For blades, the load spectrum may include elements of a typical flight consisting of the following:

1. Start-up;

2. Taxi out;

3. Run-up to take-off thrust;

4. Maximum once-per-revolution (1P) vibratory load;

5. Climb;

6. Cruise;

7. Descent;

8. Landing;

9. Reverse thrust;

10. Taxi back; and

11. Shut-down.

(2) Data Reliability. The data should be developed as in paragraph 5.k. of this AC.

c. Damage Tolerance Data.

(1) General. Developing the fatigue strength characteristics of composite propeller components for a damage tolerance fatigue evaluation should be done using full-scale test specimens. This establishes the damage growth, strength, failure location, and failure mode and mechanism of the components. The damage tolerance discussion focuses on propeller blades, but may be applied to any composite propeller component. The damage tolerance approach is based on the following factors/assumptions:

(a) Damage is inherent in the structure or inflicted in service and may grow with the repeated application of loads;

(b) The propeller or propeller components will be inspected at intervals to assess the extent of damage; and

(c) When damage reaches the maximum permissible flaw size, the propeller or propeller component will be retired.

(2) Tests to Determine Failure Mechanism. A test should be conducted to determine the failure mechanism, using knowledge of the propeller blade loading spectrum and blade design. This is the first step in the fatigue verification cycle, to verify the failure mechanism determined in the blade design phase. This also confirms the location of critical defects that will be used in subsequent fatigue tests.

(a) Test Specimen. The specimen should represent actual type design. This test verifies the critical stress locations and failure mechanism determined during the blade design process. No artificial defects or repairs should be incorporated. The predicted high stress (failure initiation) point should be far from any naturally occurring flaws.

(b) Loading. The blade should be loaded to accurately simulate the predicted critical loading environment. This may not be the highest steady load condition. For example, a low rotational speed condition with a high vibratory bending load may be more damaging due to increased transverse stress and increased compressive stress. Most composite designs are more sensitive to the LCF part of the spectrum. Therefore, particular attention should be paid to conditions that produce high fatigue loads but not high cycles. Loads induced during these and other conditions can be substantially higher than normally occurring design load conditions such as take-off rotation. To accelerate testing, alternating bending loads may be increased above the actually occurring loads to initiate and propagate damage. Some examples of these conditions are:

1. High power;

2. High yaw;

3. High g loading due to maneuver or gust loading;

4. Ground air ground;

5. Resonant vibration loads occurring during initial run-up or run-down in a cross-wind environment; and

6. Thermally induced stresses.

(c) Monitoring Failure Mechanism. The initiation and growth of damage should be monitored using non-destructive inspection (NDI) techniques. The NDI techniques to be used on the type certificated product and defined in the Instructions for Continued Airworthiness should be evaluated during this test. Since the test determines failure mechanism, the test should be run to structural failure. The definition of this failure should be the point at which the stiffness of the specimen has begun to rapidly degrade and, if possible, the component has been brought to total separation.

(2) Test to Determine Flaw Growth. This portion of the testing process determines the rate of flaw growth as a function of loading. The purpose is to determine the load that will propagate a flaw and the rate of flaw growth vs. loading.

(a) Test Specimens. Test specimens should be manufactured with the allowable manufacturing anomalies and/or permissible repairs, as appropriate, located in the most critical location. The most critical location should be determined in the design analysis and confirmed in the failure mechanism test in paragraph 6.c.(2) of this AC. If more than one critical defect has been identified, then all should be evaluated. If all defects result in the propagation of the same type of flaw, then only the propagating flaw should be analyzed. For example, if voids, resin pockets, and delaminations are possible manufacturing anomalies, and all defects result in the initiation and growth of delamination, then only delamination growth should be analyzed.

(b) Loading. As indicated in paragraph 6.c.(2)(b) of this AC, steady loads should represent normally occurring loads, and alternating loads should be applied to produce flaw growth. If flaw growth cannot be produced within the projected loading conditions, elevated loads should be used to produce flaw growth. Constant amplitude testing should be conducted at several different alternating load conditions.

(c) Monitor Failure Mechanism. Flaw growth should be monitored using NDI techniques. During the test the NDI techniques to be used on the type certificated product and defined in the Instructions for Continued Airworthiness should be evaluated. Since the purpose of this test is to determine flaw growth data, the test should be run until sufficient data has been collected to accurately model flaw growth. During flaw growth testing, the failure mode should be the same as in the failure mechanism test. If there is an inconsistency, the test and the design should be evaluated. If there is a second failure mode present, it should either be addressed through design or be independently tested.

(d) End of Life Condition. The end of life condition is established in conjunction with the service life. The component at the end of life condition meets all airworthiness loading requirements. Therefore, the component in its end of life condition is still safe. The loading requirements should account for resonant conditions when applicable. Resonant conditions may be applicable when the stiffness of the component changes sufficiently to change the natural frequency.

(e) Growth Rate. Figure 4 shows the progression of a composite blade flaw growth. The flaw in this case is the delamination area as a percentage of the total area defined by the end of life condition for the blade. The flaw growth test involves the application of a steady and alternating load while measuring the flaw growth at an interval sufficient to establish the flaw growth rate. The measured flaw growth rate is generally non-linear, as the internal stresses in the component redistribute themselves as the flaw grows. For the case shown, the growth rate lessens with increased flaw size. The conservative approach to a non-linear flaw growth rate in which the growth rate lessens with increasing flaw size is to assign a linear value to the growth data, as long as the linear value predicts flaw growth to the end of life condition before the actual failure. All components should be treated in a similar manner when assigning flaw rates. For designs that do not show a decreasing growth rate with increasing flaw size, a much more conservative approach should be used when predicting blade life and inspection intervals. The test began with a known defect size and ended prior to the end of life condition defined as 100% of the flaw size. This practice is acceptable as long as the failure mode and its progression are known. When the failure mode is unknown or unpredictable this method should not be used. For this practice to be acceptable, the initial failure test should show ample margin between the point when the assigned flaw rate predicts the end of life condition and when the actual end of life condition occurs. Multiple flaw growth tests are needed to establish the flaw growth rate as a function of load.

(f) Flaw Growth Curve. The flaw growth rate results from multiple flaw growth tests are used to develop the flaw growth curve shown in Figure 5. The flaw growth rate curve is established for data collected at uniform steady load test conditions. The flaw growth rate curve is then used to compute the flaw growth for each propeller load cycle using a flaw growth accumulation algorithm. The scales shown in Figure 5 are above 100% because flaw growth testing was conducted at loads above the limit load.

(g) Spectrum Load Testing. Spectrum load testing verifies the flaw growth accumulation algorithm used for subsequent flaw growth evaluations. A load spectrum includes operating conditions representative of the intended operation for the propeller. The load spectrum may include a typical flight consisting of no load, increase to take-off thrust and maximum 1P vibratory load, reduce to a climb conditions, reduce to cruise conditions, increase to maximum reverse conditions and return to no load. Within this spectrum a high amplitude low cycle limit maneuver load should be periodically applied.

(h) Inspection Intervals. Paragraph 9.b.(3) of this AC discusses the development of inspection intervals.

7. **METALLIC COMPONENT FATIGUE LIMITS**.

a. Discussion. The fatigue limits for metallic components are determined by developing fatigue data through coupon and full-scale testing, previous testing, past experience and acceptable published data. This data should be used to develop S-N diagrams and Goodman diagrams suitable for a safe-life propeller fatigue evaluation and da/dn diagrams for damage tolerance evaluation of the applicable propeller components.

b. Safe-Life Fatigue Limits. Developing fatigue strength characteristics for metallic components follows the two phase approach presented in paragraph 6.b. of this AC. The first phase is coupon testing for both static properties and fatigue properties to supplement the fatigue (S-N) curve shape, standard deviation, and statistical distribution. The second phase is full-scale specimen testing to establish the strength level, statistical distribution, failure location, and failure mode and mechanism for the material system and geometry of the propeller. The development of the fatigue properties may be supported by previous testing, past experience and acceptable published data, when available.

(1) Loading. Steady loads should represent normally occurring loads; alternating loads should be applied to produce the desired failure results. Testing may be conducted in steps with constant amplitude alternating loads. If the alternating load level is below the endurance limit, failure will not occur within the number of cycles considered acceptable to verify the endurance limit. This is considered a run-out condition. The alternating load should be increased and the test should be resumed at ever increasing load levels until failure occurs. LCF should also be addressed. Due to the shape of S-N diagrams for metallics, the initial emphasis of full-scale testing of metallic components is usually HCF. Coupon tests have limited value in areas with complex geometry.

(2) Data Reliability. The data should be developed as in paragraph 5.k. of this AC.

c. Damage Tolerance Data. A damage tolerance evaluation of metallic components usually uses classical fracture mechanics techniques. Alternate methodologies include a safe-life approach coupled with experimental assessment of the effect of damage, using coupons or full-scale components. Equating damage to a crack may be extremely conservative for some materials; an alternate method for quantifying the effect of the damage on life should be used in those cases. This AC assumes that when damage tolerance methods are applied the component has been designed using damage tolerance and fail-safe principles.

(1) Tests to Determine Failure Mechanism. The full-scale tests should be used to define the fracture location for an undamaged component. When including the effects of damage, it should be assumed that the damage occurs at the fracture site.

(2) Tests to Determine Crack Growth. Crack growth data for the material of interest may be obtained from acceptable published data. If no data is available, the data should be obtained by experiment, using appropriate coupon configurations and taking into account the effects of stress ratio (R).

(a) Crack Growth Curve. The experimental data should be used to produce a crack growth curve in the form of da/dn (amount of crack extension per cycle) versus ΔK (stress intensity range) as a function of stress ratio (R). Figure 6 illustrates such a curve. The curve shape is sigmoidal when presented on a log-log plot. In the mid-region the growth is linear with a slope of "m." In the initial region, called "near threshold," the plot exhibits curvature and becomes asymptotic to the threshold stress intensity range (ΔKth). In the final region, the plot exhibits opposite curvature and becomes asymptotic to critical stress intensity range (Kic).

(b) Spectrum Load Considerations. Commercial and government computer programs, such as NASA/FLAGROW, NASGRO, and FASTRAN, contain analytical crack growth algorithms that sufficiently account for growth under load spectra anticipated in service for metals. These techniques may be used for performing crack growth calculations from the initial detectable crack to the critical crack length. The loading should include all of the segments of the spectrum unless an analytical damage assessment shows that specific segments produce negligible damage. Consideration should be given to material statistical scatter. Generally this is accomplished using typical growth data and applying scatter factors on inspection intervals.

(c) Inspection Intervals. Paragraph 9.b.(3) of this AC discusses the development of inspection intervals.

8. **METALLIC HUB AND RETENTION FATIGUE TESTS**.

a. General. The complex nature of the hub and retention structure with bolted joints and bearings makes it difficult, if not impossible, to measure critical stress locations. Therefore, the fatigue test should be conducted on full-scale components, in a test rig that simulates the orientation of operating loads, so that applied test loads result in the stresses that match or exceed operating stresses in critical areas. Hub and retention testing develops S-N curve fatigue data or shows that the hub has a sufficient life through run-out testing at an appropriate load or spectrum loading. Run-out testing should be supported by previous testing, past experience, and acceptable published data, when available.

b. Loading. The centrifugal load, thrust, torque, cyclic aerodynamic loads and engine cyclic loads for reciprocating engines should be applied simultaneously during fatigue testing. The test rig in Figure 7 is one type has been shown to be suitable for testing with simultaneously applied loads for high power installations (1500 shp or greater) with high 1P loading. This type of test rig assesses the interface between the hub and propeller shaft flange. The hub is mounted to the actual propeller flange and shaft unit, which has been adapted to the test rig. The blade loads are input by means of actuators that are aligned in the direction of the applicable steady and vibratory blade loads. Centrifugal loads are also applied to load the blade retention. Since the hub loading is complex, sound judgement is required to approximate the complete loading pattern for the hub. The test rig in Figure 7 approximates that the blade vibratory and steady loads are aligned in the same direction. The suitability of any test rig should be determined before conducting the fatigue testing.

9. **FATIGUE EVALUATION**. Section 35.37 requires a fatigue evaluation. The fatigue evaluation should be conducted on a typical or the intended airplane. The intended airplane may be the airplane used during airplane certification to conduct the vibration tests and evaluation used to show compliance with §§23.907 or 25.907. The typical airplane may be the airplane used to develop design criteria for the propeller or another appropriate airplane.

a. Airworthiness Limitations. The propeller fatigue evaluation establishes the mandatory replacement times (life limits) for the propeller and, in some cases, mandatory inspections. Compliance results for a typical airplane are preliminary, as propeller vibratory stress on the intended airplane has not been measured. The evaluation is a design evaluation on a typical airplane. The evaluation provides an acceptable level of assurance, before installation on the intended airplane, that the propeller will be structurally acceptable for use on that airplane. When the fatigue evaluation is conducted on the intended airplane, the evaluation may be directly applicable to §§23.907 or 25.907. In either case the airworthiness limitations must be identified and appropriately documented in the Airworthiness Limitations Section of the propeller Instructions for Continued Airworthiness. The Instructions for Continued Airworthiness should note that the applicability of the Airworthiness Limitations should be reevaluated for each new airplane installation.

b. Approaches. There are a number of different approaches to fatigue evaluation. This AC presents fatigue limits that support safe-life and damage tolerance evaluations. AC 20-66A provides guidance on conducting the fatigue evaluation. The following concepts should be considered when conducting the fatigue evaluation:

(1) Unlimited Life. When it is shown that all stresses are below the endurance limits established for the component, the component is said to have unlimited life. Components with unlimited life will be removed from service for reasons other than fatigue. In addition, when the safe-life of a component is shown to be greater than 70,000 hours, and it is shown that the component will be safely retired from service for reasons other than fatigue before its safe-life, the component may be said to have unlimited life.

(2) Safe-Life. Safe-life is the component fatigue life reduced by an appropriate scatter factor that accounts for the variability of the fatigue evaluation process. The fatigue life is determined by combining the airplane loading spectrum with the fatigue data using a damage summation algorithm (safe-life evaluation). Mandatory replacement times are established for parts with safe-lives. Unless substantially justified the following scatter factors should be applied:

(a) Metallic scatter. A scatter factor of three or greater should be used for metallic components.

(b) Composite scatter. A scatter factor of 10 or greater should be used for composite components.

(3) Damage Tolerance Inspection Interval. For damage tolerance methods the inspection interval is determined by the relationship between the time the damage reaches maximum permissible flaw size as defined during certification (detectable damage) and the end of life condition. The maximum permissible flaw size is established during certification by considering the inspection method, the inspection interval, and the end of life condition. The inspection interval permits multiple opportunities, usually three, to find the damage before the component reaches the end of life condition. The inspection method should also be evaluated to determine the probability of detection (POD). Inspection methods should have a POD of 90% probability with 90% confidence. When the POD is less than 90% probability with 90% confidence, the inspection frequency should be increased. The component should be removed from service when damage is detected at the maximum permissible flaw size. The Airworthiness Limitations Section of the Instructions for Continued Airworthiness establishes these inspections as mandatory.

(4) Limit Load Fatigue Test. The propeller blade should be able to withstand limit loads without detrimental permanent deformation or deformation that would result in a hazardous propeller effect. This should be demonstrated by a fatigue test, since the propeller is a rotating device. Appropriate life factors should be applied during the test. The test should be performed on a blade in end of life condition.

Original signed by JJP on 9/17/01

Jay J. Pardee
Manager, Engine and Propeller Directorate,
Aircraft Certification Service

FIGURE 1

LOAD AND LIFE FACTORS

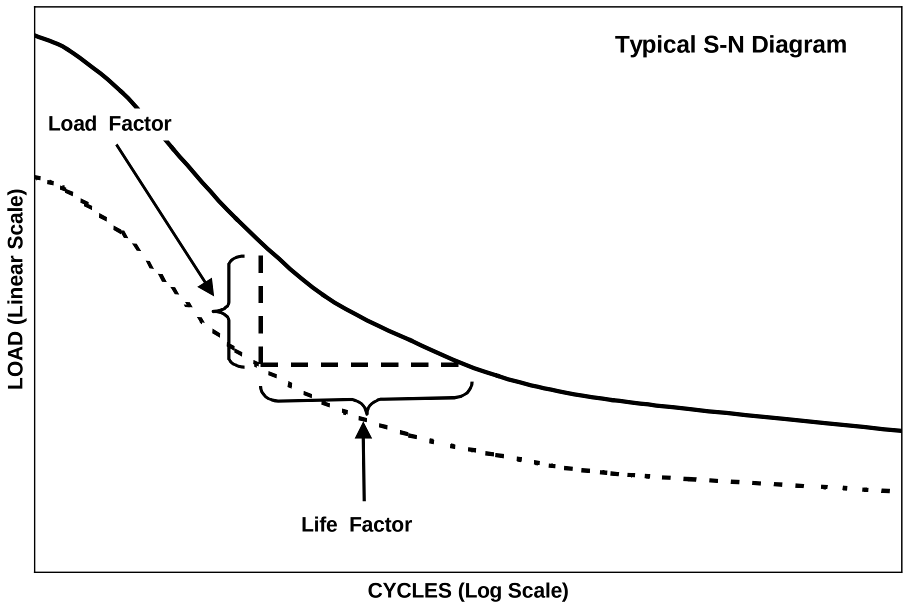


FIGURE 2

COMPOSITE BLADE DAMAGE DEVELOPMENT

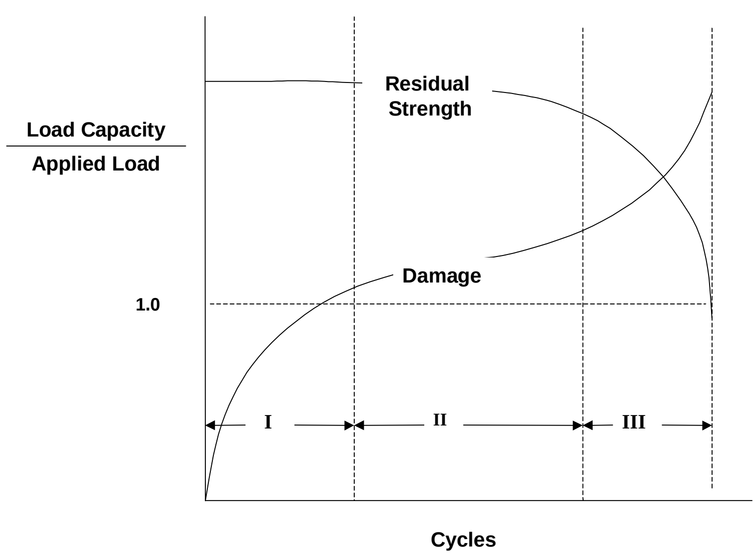
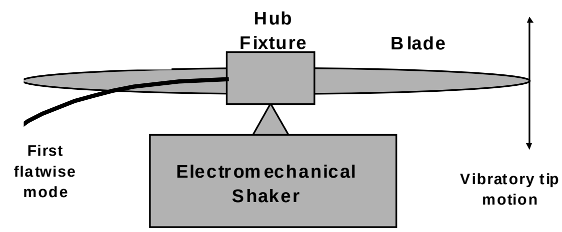


FIGURE 3

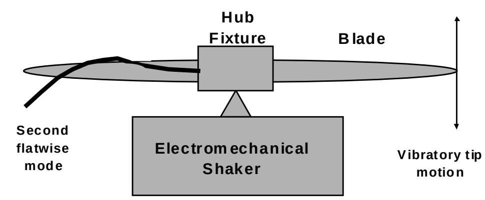
EXAMPLES OF BLADE TEST RIGS

ROOT ZERO MEAN STRESS TEST



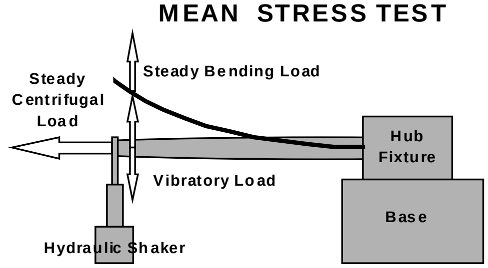
- Peak stress near the blade root
- R ratio equal zero, does not match flight loads for blade root
- Resonant test

TIP ZERO MEAN STRESS TEST



- Peak stress near the blade tip
- R ratio equal zero
- Resonant test

MEAN STRESS TEST



- Peak stress near the blade root
- R ratio adjusted for flight condition simulated
- Forced response test

FIGURE 4

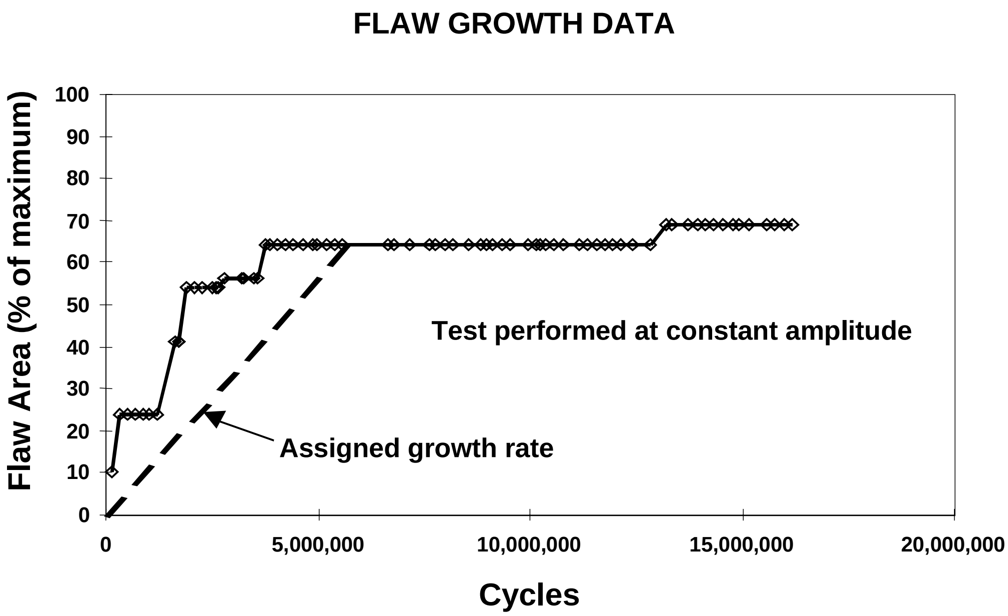


FIGURE 5

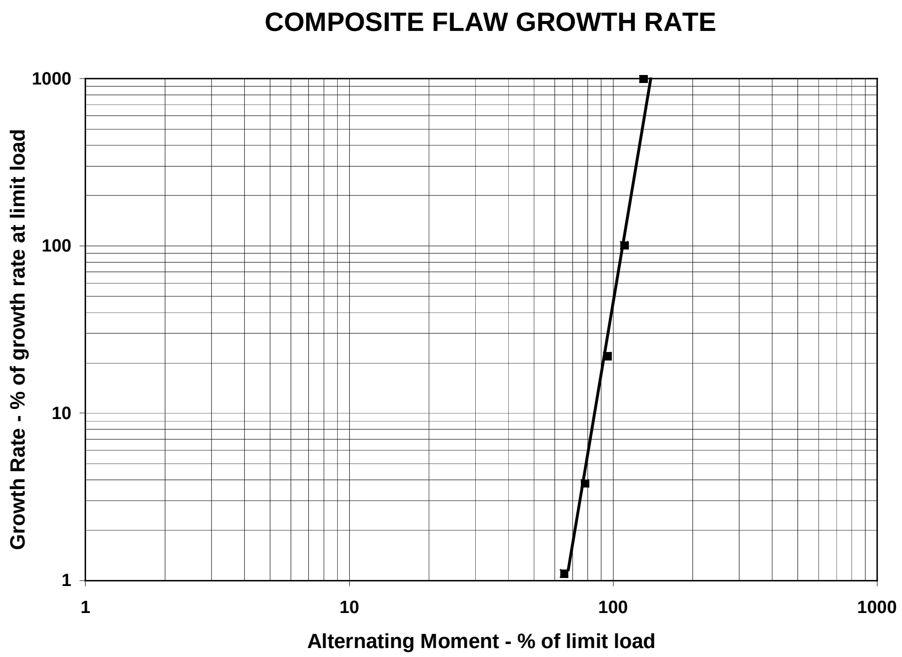


FIGURE 6

Typical Crack Growth Curve Shape

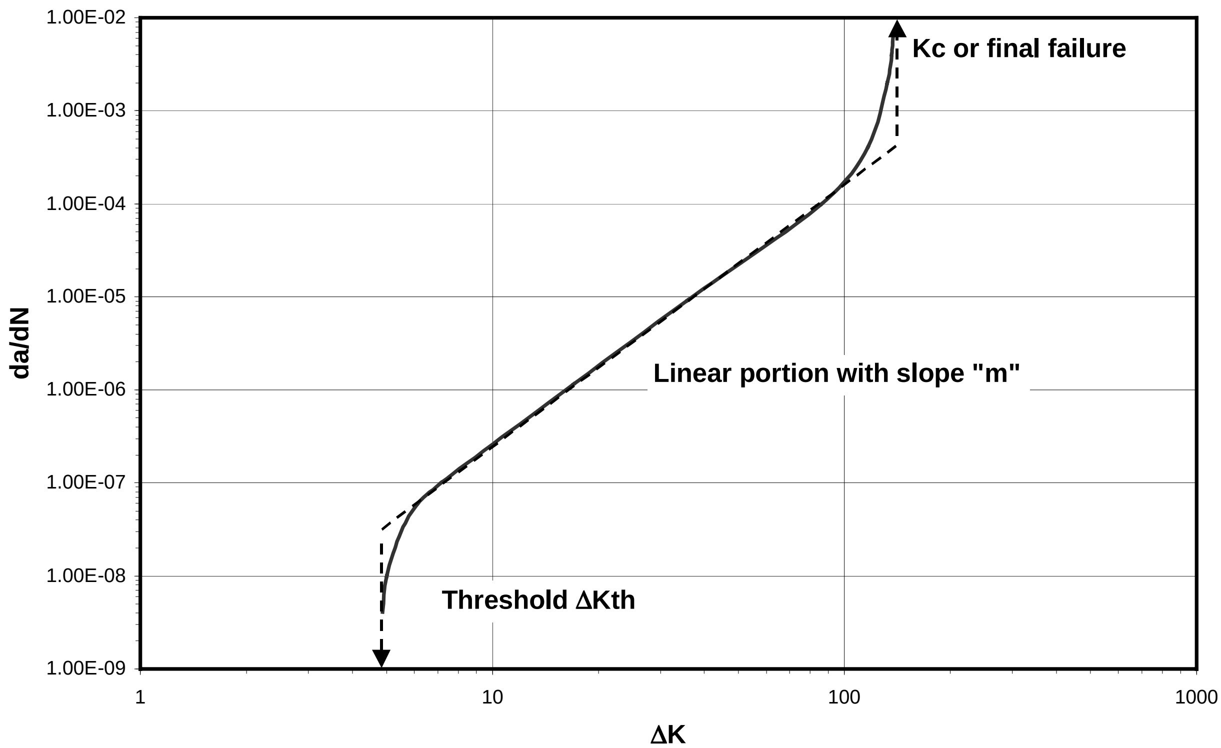
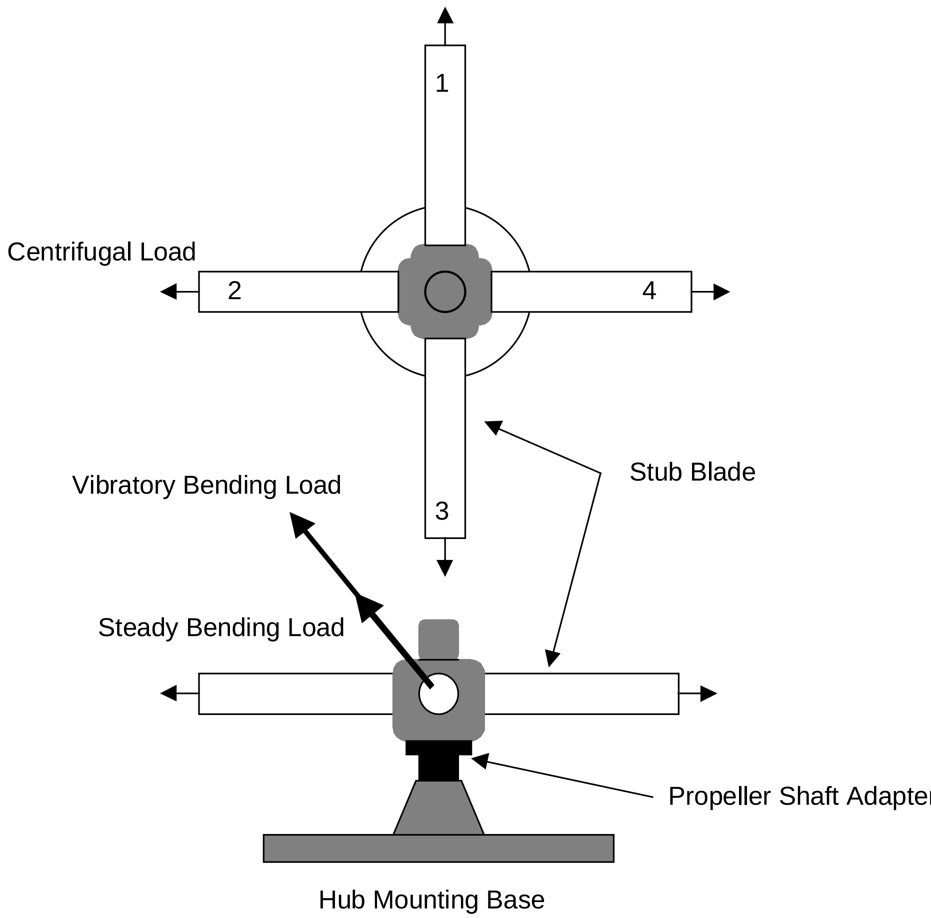


FIGURE 7

HUB AND BLADE RETENTION TEST RIG



Notes:

Loads are applied to each blade stub

Vibratory bending is simple harmonic loading and 90 degrees out of phase from blade stub to blade stub

Steady bending is maintained at all times

Centrifugal loading is maintained at all times