against hazards to the rotorcraft in the event of their malfunctioning or failure.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6-4, 24 F.R. 7072, Sept. 1, 1959, effective Oct. 1, 1959.)

Instruments; Installation

6.610 ***General.*** **The provisions of sections 6.611 through 6.613 shall apply to the installation of instruments in rotorcraft.**

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956.)

6.611 ***Arrangement and visibility of instrument installations.***

(a) Flight, navigation, and powerplant instruments for use by each pilot shall be easily visible to him.

(b) On multiengine rotorcraft, identical powerplant instruments for the several engines shall be so located as to prevent any confusion as to the engines to which they relate.

(c) The vibration characteristics of the instrument panel shall be such as not to impair seriously the readability or the accuracy of the instruments or to damage them.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956.)

6.612 ***Flight and navigational instruments.***

(a) ***Airspeed indicating system.*** **The airspeed indicating system shall be so installed that the airspeed indicator shall indicate true airspeed at sea level under standard conditions to within an allowable installational error of not more than plus or minus 3 percent of the calibrated airspeed or 5 mph, whichever is greater. The calibration shall be made in flight at all forward speeds of 10 mph or over. The allowable installation error shall not be exceeded at any forward speed above 80 percent of the climbout speed. (See sec. 6.732.)**

(b) ***Static air-vent system.*** **All instruments provided with static air case connections shall be so vented that the influence of rotorcraft speed, the opening and closing of windows, airflow variation, moisture, or other foreign matter will not seriously affect their accuracy.**

(c) ***Magnetic direction indicator.*** **The magnetic direction indicator shall be so installed that its accuracy shall not be excessively affected by the rotorcraft's vibration or magnetic fields. After the direction indicator has been compensated, the installation shall be such that the deviation in level flight does not exceed 10° on any heading. A suitable calibration placard shall be provided as specified in section 6.733.**

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6-4, 24 F.R. 7072, Sept. 1, 1959, effective May 3, 1962.)

6.613 ***Powerplant instruments.***

(a) ***Instrument lines.*** **Instrument lines shall comply with the provisions of section 6.425. In addition, instrument lines carrying flammable fluids or gases under pressure shall be provided with restricted orifices or equivalent safety devices at the source of the pressure to prevent the escape of excessive fluid or gas in case of line failure.**

(b) ***Fuel quantity indicator.*** **Fuel quantity indicators shall be calibrated to read zero during level flight when the quantity of fuel remaining in the tank is equal to the unusable fuel supply as defined by section 6.421. (See also sec. 6.736.)**

(c) ***Fuel flowmeter system.*** **When a flowmeter system is installed, the metering component shall include a means for by-passing the fuel supply in the event that malfunctioning of the metering component results in a severe restriction to fuel flow.**

(d) ***Oil quantity indicator.***

(1) Means shall be provided to indicate the quantity of oil in each tank when the rotorcraft is on the ground. (See sec. 6.735.)

(2) If an oil transfer system or a reserve oil supply system is installed, means shall be provided to indicate to the crew during flight the quantity of oil in each tank.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6-4, 24 F.R. 7072, Sept. 1, 1959, effective May 3, 1962.)

Electrical Systems and Equipment

[6.617 ***Installation.*** **Electrical systems in rotorcraft shall be free from hazards in themselves, in their method of operation, and in their effects on other parts of the rotorcraft. Electrical equipment shall be of a type and design adequate for the use intended. Electrical systems shall be installed in such a manner that they are protected from fuel, oil,**

water, other detrimental substances, and mechanical damage.]

(Added by Amdt. 6–5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.618 ***Electric power sources.***

[(a) Electric power sources, their transmission cables, and their associated control and protective devices shall have sufficient capacity to furnish the required power at the proper voltage to all load circuits which are essential to the safe operation of the rotorcraft.

[(b) Compliance with paragraph (a) of this section shall be shown by means of an electrical load analysis, or by electrical measurements, which take into account all electrical loads applied to the electrical system, in probable combinations and for probable durations.

[(c) At least one generator shall be installed if the electrical system supplies power to load circuits which are essential to the safe operation of the rotorcraft.

[(d) Electric power sources shall function properly when connected in combination or independently. The failure or malfunction of any electric power source shall not impair the ability of any remaining source to supply load circuits which are essential to the safe operation of the rotorcraft.

[(e) Electric power source controls shall be such as to permit independent operation of each source.]

(Added by Amdt. 6–5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.619] ***Storage battery design and installation.*** **Storage batteries shall be of such design and so installed that:**

(a) Safe cell temperatures and pressures are maintained during any probable charging or discharging condition. No uncontrolled increase in cell temperature shall result when the storage battery is recharged (after previous complete discharge) at maximum regulated voltage, during a flight of maximum duration, under the most adverse cooling condition likely to occur in service. Tests to demonstrate compliance with this regulation shall not be required if satisfactory operating experience with similar batteries and installations has shown that maintaining safe cell temperatures and pressures presents no problem.

(b) Explosive or toxic gases emitted by the storage battery in normal operation, or as the result of any probable malfunction in the charging system or battery installation, shall not accumulate in hazardous quantities within the rotorcraft.

(c) Corrosive fluids or gases which may be emitted or spilled from the storage battery shall not damage surrounding rotorcraft structure or adjacent essential equipment.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6–4, 24 F.R. 7072, Sept. 1, 1959, effective Oct. 1, 1959; redesignated by Amdt. 6–5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.620 ***Generator.*** **Generators shall be capable of delivering their continuous rated power.]**

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6–5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.621 ***Generator controls.***

[(a) Generator voltage control equipment shall be capable of dependably regulating the generator output within rated limits.

[(b) A generator reverse current cutout shall be incorporated and designed to disconnect the generator from the battery and other generators when the generator is developing a voltage of such value that current sufficient to cause malfunctioning can flow into the generator.]

(Added by Amdt. 6–5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.622 ***Electric power system instruments.*** **Means shall be provided to indicate to appropriate crewmembers those electric power system quantities which are essential for the safe operation of the system. For direct current systems, an ammeter which can be switched into each generator feeder shall be acceptable. When only one generator is installed, it shall be acceptable to locate the ammeter in the battery feeder.]**

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6–5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.623 ***Master switch arrangement.*** **A ~~master switch arrangement shall be provided to permit expeditious disconnection of all electric power sources from all load circuits.~~**

~~The point of disconnection shall be adjacent to the power sources.~~] See Amend 6-6

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6-5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

6.624 *Master switch installation.* The master switch or its controls shall be so installed that it is easily discernible and accessible to a member of the crew in flight.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956.)

[6.625 *Fuses or circuit breakers.* Protective devices (fuses or circuit breakers) shall be installed in the circuits to all electrical equipment, except that such items need not be installed in the main circuits of starter motors or in other circuits where no hazard is presented by their omission. Not more than one circuit, which is essential to safety in flight, shall be protected by a single protective device. All resettable type circuit protective devices shall be designed so that a manual operation is required to restore service after tripping and so that, when an overload or circuit fault exists, they will open the circuit irrespective of the position of the operating control.

[*Note:* The aforementioned resettable type circuit protective devices are known commercially as "trip-free"; i.e., the tripping mechanism cannot be overridden by the operating control. Such circuit protective devices can be reset on an overload or circuit fault, but will trip subsequently in accordance with their current-time trip characteristic.]

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6-5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.626 *Protective devices installation.* If the ability to reset a circuit breaker or to replace a fuse is essential to safety in flight, such circuit breaker or fuse shall be so located and identified that it can be readily reset or replaced in flight. If fuses are used, one spare of each rating or 50 percent spare fuses of each rating, whichever is the greater, shall be provided.]

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6-5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

[6.627 *Electric cables.* Electric connecting cables shall be of adequate capacity. Cables which would overheat in the event of circuit overload or fault shall be flame-resistant and shall not emit dangerous quantities of toxic fumes.]

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6-5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

6.628 *Switches.* Switches shall be capable of carrying their rated current. They shall be accessible to the crew and shall be labeled as to operation and the circuit controlled.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956.)

Lights

6.630 *Instrument lights.*

(a) Instrument lights shall provide sufficient illumination to make all instruments, switches, etc., easily readable.

(b) Instrument lights shall be so installed that their direct rays are shielded from the pilot's eyes and so that no objectionable reflections are visible to him.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956.)

6.631 *Landing lights.*

(a) When landing or hovering lights are required, they shall be of an approved type.

(b) Landing lights shall be installed so that there is no objectionable glare visible to the pilot and so that the pilot is not adversely affected by halation.

(c) Landing lights shall be installed in a location where they provide the necessary illumination for night operation including hovering and landing.

(d) A switch for each light shall be provided, except that where multiple lights are installed at one location a single switch for the multiple lights shall be acceptable.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956.)

6.632 *Position light system installation.*

(a) *General.* The provisions of sections 6.632 through 6.635 shall be applicable to the position light system as a whole. The position light system shall include the items specified in paragraphs (b) through (e) of this section.

(b) *Forward position lights.* Forward position lights shall consist of a red and a green light spaced laterally as far apart as practicable

and installed forward on the rotorcraft in such a location that, with the rotorcraft in normal flying position, the red light is displayed on the left side and the green light is displayed on the right side. The individual lights shall be of an approved type.

(c) *Rear position light.* The rear position light shall be a white light mounted as far aft as practicable. The light shall be of an approved type.

(d) *Circuit.* The two forward position lights and the rear position light shall constitute a single circuit.

(e) *Light covers and color filters.* Light covers or color filters used shall be of flame-resistant material and shall be constructed so that they will not change color or shape or suffer any appreciable loss of light transmission during normal use.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6–1, 22 F.R. 1274, Mar. 1, 1957, effective Apr. 1, 1957; Amdt. 6–4, 24 F.R. 7072, Sept. 1, 1959, effective Oct. 1, 1959.)

6.633 *Position light system dihedral angles.* The forward and rear position lights as installed on the rotorcraft shall show unbroken light within dihedral angles specified in paragraphs (a) through (c) of this section.

(a) Dihedral angle *L* (left) shall be considered formed by two intersecting vertical planes, one parallel to the longitudinal axis of the rotorcraft and the other at 110° to the left of the first, when looking forward along the longitudinal axis.

(b) Dihedral angle *R* (right) shall be considered formed by two intersecting vertical planes, one parallel to the longitudinal axis of the rotorcraft and the other at 110° to the right of the first, when looking forward along the longitudinal axis.

(c) Dihedral angle *A* (aft) shall be considered formed by two intersecting vertical planes making angles of 70° to the right and 70° to the left, respectively, looking aft along the longitudinal axis, to a vertical plane passing through the longitudinal axis.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956.)

6.634 *Position light distribution and intensities.*

(a) *General.* The intensities prescribed in this section are those to be provided by new equipment with all light covers and color filters in place. Intensities shall be determined with the light source operating at a steady value equal to the average luminous output of the light source at the normal operating voltage of the rotorcraft. The light distribution and intensities of position lights shall comply with the provisions of paragraph (b) of this section.

(b) *Forward and rear position lights.* The light distribution and intensities of forward and rear position lights shall be expressed in terms of minimum intensities in the horizontal plane, minimum intensities in any vertical plane, and maximum intensities in overlapping beams, within dihedral angles L, R, and A, and shall comply with the provisions of subparagraphs (1) through (3) of this paragraph.

(1) *Intensities in horizontal plane.* The intensities in the horizontal plane shall not be less than the values given in figure 6–1. (The horizontal plane is the plane containing the longitudinal axis of the rotorcraft and is perpendicular to the plane of symmetry of the rotorcraft.)

(2) *Intensities above and below horizontal.* The intensities in any vertical plane shall not be less than the appropriate value given in figure 6–2, where I is the minimum intensity prescribed in figure 6–1 for the corresponding angles in the horizontal plane. (Vertical planes are planes perpendicular to the horizontal plane.)

(3) *Overlaps between adjacent signals.* The intensities in overlaps between adjacent signals shall not exceed the values given in figure 6–3, except that higher intensities in the overlaps shall be acceptable with the use of main beam intensities substantially greater than the minima specified in figures 6–1 and 6–2 if the overlap intensities in relation to the main beam intensities are such as not to affect adversely signal clarity.

(Part 6, 21 F.R. 10291, Dec. 22, 1956, effective Dec. 20, 1956; as amended by Amdt. 6–5, 27 F.R. 2996, Mar. 30, 1962, effective May 3, 1962.)

Appendix A

Methods of Rotor Service Life Determination

Introduction

Service experience in the helicopter field indicates that fatigue considerations are of extreme importance in the design of the rotating major load-carrying members of the helicopter. In view of the importance of this problem, designers are urged to give great care to the detail design of rotor blades, hub retention systems and controls in order that stresses associated with oscillatory loading be kept well below the allowable material endurance limit. As far as practicable, the design should be clean, care being taken to reduce stress concentrations to a minimum. Since lack of quality control may easily result in large variations in fatigue life, great care should be taken to insure that production parts and assemblies are made with the same care as the components used in any fatigue test.

Although a uniform approach to rotor fatigue problems is desirable, it is recognized that in such a relatively new field, new design features, methods of fabrication or configurations may require variations and deviations from the methods described herein. Engineering judgment should therefore be exercised in each case.

Although there is some question as to whether a completely rational method exists for the prediction of the fatigue life of a built-up structure subject to random loading, nevertheless it is believed that an engineering approach to the subject can be attained through the application of the Cumulative Damage Hypothesis. This hypothesis asserts that every cycle of stress above an "endurance limit" produces damage proportional to the ratio of cycles run at that stress to the fatigue life at that stress level. Laboratory tests of this hypothesis indicate that it is reasonably valid when the stress cycles are of random magnitude. That is, stress spectra in which all high-stress magnitudes are applied consecutively and then all low-stress magnitudes applied, do not obey the hypothesis. Despite the approximations involved in the hypothesis and the lack of an adequate theory connecting the hypothesis with more basic properties of materials, it attempts to take more factors into account than any other method developed so far.

In any rational determination of the fatigue life of a structure, three basic factors must be known. These factors are:

Knowledge of the stresses and associated flight maneuvers to be expected in normal operation;

Knowledge of the frequency of occurrence of specific loadings;

Knowledge of the fatigue strength characteristics of the structure.

Flight Stress Measurements

It is generally agreed that because of the approximations employed in rotor load and stress distribution analyses, it is not possible at present to determine analytically a reasonable approach to rotor fatigue stress levels.

Rotor stress levels are therefore determined by means of carefully controlled, instrumented flight strain gage testing. These tests are aimed at the determination of the magnitude of steady and oscillatory stresses associated with normal helicopter operation and the correlation of the occurrence of critical stresses with specific maneuvers or operating conditions. In some cases the information so obtained can be used to limit or placard against specific maneuvers. In other cases where prohibition of specific maneuvers or operations is not feasible the information so obtained can be of use in setting up a test program which would determine the fatigue life of the part.

Prior to conducting a flight strain gage testing program, some rational evaluation of the critical stress areas must be made in order to determine the proper distribution of gages. A qualitative study is usually made by means of brittle coatings (such as Stresscoat), by photo-elastic methods or by analytic means. In conducting flight strain measurements, besides the proper distribution of strain gages on hubs, blades, blade attachments and control members, provision is usually made for recording the collective pitch setting of the rotor blades and the center of gravity acceleration during maneuvers. This is done so that it can be ascertained that for maneuvers in which a rapid control movement is utilized the severity of application of control is representative of that which can be encountered during actual service operation.

Table 1 contains a suggested list of maneuvers for investigation in a flight strain survey. These maneuvers are usually investigated over the complete r. p. m. range (from minimum design r. p. m. to maximum design r. p. m.) as well as the complete speed, altitude, center of gravity and weight ranges.

Frequency of Loading

The second item of great importance in the determination of service life, is the matter of determining the percentage of total operating time associated with each flight maneuver. At best, this evaluation can only be a statistical one and will of necessity be a function of the purpose for which the particular helicopter is intended to be used. Obviously a helicopter used solely for crop dusting would have a different time distribution for various maneuvers than one which is to be used for mail or passenger ferry service between a local airport and the center of a nearby city. At present, because of the limited number of helicopters in use this problem can be handled by means of reasonable, conservative approximations. As the types of operation increase, with the rapidly expanding field of helicopter operation, this problem will undoubtedly require re-evaluation.

Table 1 represents the considered opinion of a number of helicopter specialists regarding the maneuvers to be investigated (over the complete r. p. m., speed, c. g., weight and altitude ranges) as well as an appropriate percentage distribution of the occurrence of these maneuvers.

TABLE 1

Percent Occurrence

I GROUND CONDITIONS	
(a) Rapid increase of r. p. m. on ground to quickly engage clutch	0. 5
(b) Taxiing with full cyclic control	. 5
(c) Jump takeoff	. 5
II HOVERING	
(a) Steady hovering	. 5
(b) Lateral reversal	1. 0
(c) Longitudinal reversal	1. 5
(d) Rudder reversal	1. 0
III FORWARD FLIGHT POWER ON	
(a) Level Flight—20% V_{NE}	5. 0
(b) Level Flight—40% V_{NE}	10. 0
(c) Level Flight—60% V_{NE}	18. 0
(d) Level Flight—80% V_{NE}	18. 0
(e) Maximum Level Flight (but not greater than V_{NE})	10. 0
(f) V_{NE}	3. 0
(g) 111% V_{NE}	. 5
(h) Right Turns	3. 0
(i) Left Turns	3. 0
(j) Climb (Max. Continuous Power)	4. 0
(k) Cyclic and collective pull-ups from level flight	. 5
(l) Change to autorotation from power-on flight	. 5
(m) Partial power descent (including condition of zero flow through rotor)	2. 0
(n) Landing approach	3. 0
(o) Lateral reversals at V_H	. 5
(p) Longitudinal reversals at V_H	. 5
(q) Rudder reversals at V_H	. 5
(r) Climb (Takeoff Power)	2. 0
IV AUTOROTATION—POWER OFF	
(a) Steady forward flight	2. 5
(b) Right turns	1. 0
(c) Left turns	1. 0
(d) Lateral reversals	. 5
(e) Longitudinal reversals	. 5
(f) Rudder reversals	. 5
(g) Cyclic and collective pull-ups	2. 0
(h) Landings (including flares)	2. 5
	100. 0

Fatigue Strength

The third phase of the fatigue evaluation program involves the determination of the fatigue strength of the actual structure. Although the fatigue characteristics of simple material specimens are often available, the direct application of this information to built-up structures is questionable. The available material data modified by appropriate stress concentration factors can undoubtedly be used as an important tool in design. However, propeller and helicopter rotor experience indicates that various factors may reduce the fatigue strength of a built-up structure below that of material specimens with severe notched stress concentrations. It therefore is necessary that endurance tests of the critical parts be conducted by applying steady and oscillatory loads in a manner simulating the loading actually encountered in service.

Although the foregoing indicates the difficulty in correlating material fatigue data with that of a built-up structure, nevertheless it is recognized that minimum acceptable stress levels can be established, such that, if the maximum measured stresses in a component be lower than the established levels, no fatigue testing need be required. The following technique which is based on the use of a Goodman Diagram for the material modified by suitable factors to account for stress concentration factors plus a factor of safety is considered acceptable for the establishment of this minimum stress level.

1. Establish the Goodman Diagram from material data for the perfect specimen. This line will mark the endurance limit for various vibratory and steady stress levels.

2. The allowable full reversal stress for the material should then be reduced to account for the stress concentration factor present in the actual rotor part. The stress concentration factor chosen should adequately account for surface finish, fabrication methods, probability of galling as well as the stress concentrations around notches, threads, holes, fillets, etc. The resulting line on the Goodman Diagram will then be the failure boundary line for the part.

3. A margin of safety of two should be applied to the failure boundary curve in order to establish an operating boundary line. Thus the operating boundary line would have a slope of ⅓ the failure boundary curve.

4. If the flight strain measurements indicate that all nominal operating stresses [1] fall below the operating boundary line, no fatigue testing is required.

When the measured stresses are above the operating boundary line (see figure 1) fatigue tests of the actual component are required.

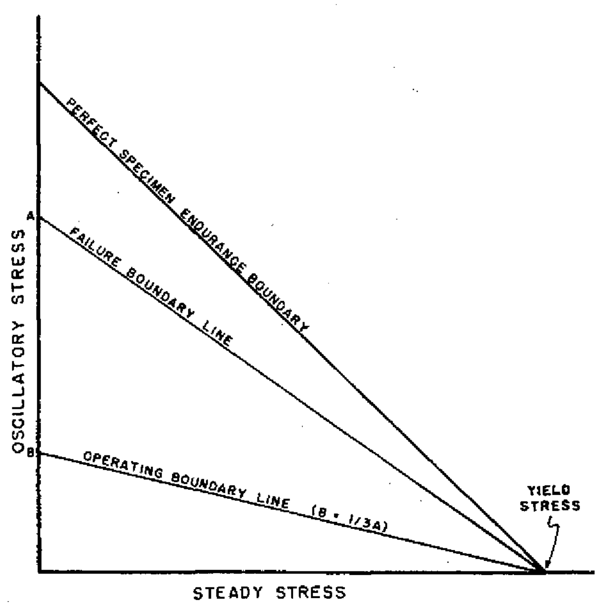

Figure 1.

Several methods of fatigue testing are currently available. The various methods such as laboratory, flight endurance or whirl stand testing methods are of course applicable only to the extent that the range of steady and vibratory flight stresses can be duplicated in the fatigue test procedure. Because of the greater degree of control which can be maintained in the laboratory, this method is recommended. However, flight or whirl stand testing is acceptable in lieu of laboratory testing if they are conducted under controlled conditions.

[1] Nominal operating stress: It is usually not possible to place the strain gage so that the stress at the critical section is measured. Instead, the gage is located at a reference point close to the critical section. The measured stress data can be reduced to equivalent loads. Subsequent application of conventional methods of stress analysis would convert these loads to stresses at the critical section (neglecting stress concentration factor).

Methods of Loading

(A) Laboratory

The laboratory method of determining the fatigue strength involves testing in a fatigue machine the critical section or sections of a rotor component. In this procedure small sections can be tested under carefully controlled conditions.

(B) Flight

The flight method of fatigue testing involves the use of the entire helicopter itself as a fatigue machine. This method, if employed, should be conducted under such controlled conditions that the level of stresses and number of fatigue cycles are known accurately enough during the test to determine the fatigue limit and service life of the critical components of the rotor system.

(C) Whirl Stand

The whirl stand procedure can be considered to be a variation of the flight test method. This involves testing complete rotor components on a test stand. The validity of this method is predicated on the ability to duplicate flight stress conditions in the test set-up.

Test Procedures

Several procedures are available for the determination of the fatigue strength of the critical component. Fatigue strength evaluation through (A) the establishment of *S–N* curves, (B) by testing in cyclical units or a suitable combination of these two procedures is considered to be acceptable.

(A) Establishment of *S–N* Curves

An *S–N* curve for a particular section can be established by testing samples of the critical section at a fixed steady stress and varying the oscillatory component of the stress. Thus, if at a steady tensile stress of level A and oscillatory stress of level B, the sample is fatigue tested to failure, failure occurring after N_1 cycles, a point on the *S–N* diagram for steady stress level A is determined. Additional points can be determined by maintaining the same steady stress A and choosing a different oscillatory stress for each sample. One such curve is needed for each critical steady stress level. Because of scatter usually associated with fatigue testing, a large number of specimens are tested in order to establish these curves. This procedure of establishing *S–N* curves can theoretically be achieved either by laboratory or whirl stand testing, however, for obvious practical reasons this procedure is usually reserved for laboratory testing.

Since it may be impossible to handle the complete blade and retension system with one setup due to practical limitations of applying required loads to the structure for establishing a representative *S–N* diagram for the rotor, it may be desirable to establish a set of criteria for hub and retention portions of the rotor separately from the blade. Also, since the critical loads entering the hub retention can be along different axes, it may be necessary to determine an *S–N* curve for each axis individually, i. e., one for the major axis and another for the minor axis since one may be critical for certain r. p. m. or maneuvers and the other at a different set of conditions.

Stress raisers have little effect on the static failing load owing to plastic deformation relieving the high stress. Similarly, for oscillatory or repeated high-loads, the effect of stress raisers on the fatigue strength is diminished by the above form of stress relief. At low loads, however, the stress raisers are fully effective in reducing the fatigue strength, which then approaches that appropriate to the nominal stress concentration.

In general, tests have shown that the fatigue stress concentration factor although lower than the theoretical factor (determined by photoelastic or other rational methods) varies with the loading, decreasing sharply in the region of the yield stress. An arbitrarily chosen high-stress level might therefore result in the critical section being beyond the yield point with resultant stress relief and reduced stress concentration factor whereas a neighboring section might be operating close to the yield stress and fail first, even though for the actual operating stress range the first section would be critical. Therefore, as a general rule it is not advisable to conduct fatigue tests at arbitrarily chosen levels appreciably higher than actual operating stress levels.

From the flight stress measurements, the frequency of occurrence of the flight maneuvers and the *S–N*. curves, the fatigue life of the part can be calculated.

While all maneuvers are to be conducted over the complete speed, center of gravity, altitude, rotor r. p. m. and weight ranges, only the combination of those conditions which produce the most critical stress for any one maneuver should be used in calculating the fatigue life. The percentage of occurrence value given in table 1 should be used with this critical condition for the maneuver. Thus the stress associated with the most critical r. p. m. center of gravity, altitude and weight for level flight, power on at 20% V_{NE} would be considered to occur for 5% of the life in the fatigue evaluation. An example of a fatigue life determination is given at the end of appendix A.

(B) Testing in Cyclical Units

This procedure involves the testing of each specimen at a series of stress levels, the number of cycles to be run at each level being proportional to the expected percentage of time associated with the particular sought condition giving rise to the specific stress level. Since the life of the part is unknown beforehand, the stress levels must be covered in arbitrarily chosen cyclical units. Thus, if units of 100 hours are chosen, then reference to table 1 would indicate 0.5 hour of rapid increase of r. p. m. on the ground to quickly engage clutch, 0.5 hour jump takeoff, 10 hours at 20% V_{NE} for level flight, etc. Then if a failure occurred at some time during such a unit, the fatigue life would be determined by the number of completed units. Thus, if the unit was 100 hours and failure occurred during the 14th unit, the fatigue life would be based on 13 completed units (i. e., 1,300 hours). It should be noted that the Cumulative Damage Hypothesis which is being used herein for fatigue life evaluation has been found to be valid only when the stress cycles are of random magnitude. Therefore, if the cyclical unit procedure is adopted, care should be taken to avoid the application of all high stress levels consecutively and then all low stresses. It is therefore likewise desirable to keep the units of time at reasonably low levels.

(C) Acceptable Modified Procedures

As mentioned previously, rational modifications or combinations of the above procedures may be made. Thus, if it is desired to limit the scope of fatigue testing, a single *S-N* curve based on the highest measured mean stress could be utilized in the fatigue life calculations. Another acceptable approach would be to demonstrate that the most critical stress level was below the endurance limit. This could be demonstrated by testing at the highest stress level to 10^7 cycles for ferrous materials and 5 x 10^7 cycles for nonferrous materials. An acceptable combination of *S-N* and cyclic unit approach would involve the establishment of the knee of the *S-N* curve (endurance limit) and the flight conditions which resulted in stresses falling below the endurance limit. The method of cyclic testing could then be employed only for those flight conditions which would cause fatigue damage. Thus, if it is established that all level flight conditions result in stresses below the endurance limit, the length of the fatigue test by cyclic units could be appreciably reduced.

Fatigue Life vs. Service Life

Since actual operating conditions involve factors the quantitative effects of which cannot readily be ascertained, it becomes necessary to distinguish between fatigue life as determined by laboratory or other accelerated fatigue tests and service life which is interpreted as the required retirement life of the part. Furthermore, because of material and fabrication variations, even under idealized laboratory conditions it has been estimated that approximately thirty test specimens are required to establish each *S–N* curve. In view of the required time and high costs involved, it must be recognized that only a limited amount of testing can be economically tolerated by most manufacturers. It is therefore important that a minimum fatigue test program be determined and that a service life which is less than the calculated fatigue life, but consistent with the degree of fatigue testing, be established.

Service Life

For some designs, it may be possible to demonstrate that all flight and ground load stresses are below the endurance limit for the critical parts of the rotor. For such cases, no limit need be imposed on the service life.

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Compliance with either of the following conditions may be considered to be a minimum acceptable level of demonstrating that all stresses are below the endurance limit.

1. If *all measured stresses fall below the* operating boundary line (figure 1), no fatigue testing is required.

2. Fatigue testing at the mean stress associated with the most critical mean-oscillatory stress level measured in flight. No failure should occur before 10^7 cycles for ferrous materials nor before 5×10^7 cycles for nonferrous materials. The minimum number of test specimens required is dependent on the oscillatory test level, in the following manner:

(*a*) A minimum of 4 test specimens if the oscillatory level is chosen at 1.1 times the critical oscillatory stress level.

(*b*) A minimum of 3 test specimens if the oscillatory level is chosen at 1.25 times the critical oscillatory stress level.

(*c*) A minimum of 2 test specimens if the oscillatory level is chosen at 1.5 times the critical oscillatory stress level.

(*d*) One specimen if the oscillatory level is chosen at twice the critical oscillatory stress level.

It is to be noted at this point that the previous reference recommending against the use of arbitrary stress levels appreciably higher than actual operating stress levels is considered to be inapplicable in this case. This is due to the fact that the stresses involved here are low since the test involved is aimed at demonstrating that the arbitrarily raised stresses are still below the endurance limit.

Where finite fatigue life is indicated and *S–N* curves are employed in determining this life, a minimum of 4 points on each *S–N* curve should be established. If it is desired to limit the fatigue tests, a single *S–N* curve based on the highest measured mean stress could be utilized in the fatigue life calculations. However, if this approach tends to unduly limit the fatigue life, a family of curves can be developed from two established *S–N* curves by means of Goodman or similar diagrams. Service life should then be established at 75% of the calculated fatigue life but should be no greater than 2,500 hours. Where the fatigue life is established by *cyclic variation of load*, a minimum of 4 specimens should be tested. The fatigue life should be based on the specimen in which the smallest number of such cycles is completed. The service life should be established at 75% of this fatigue life but should be no greater than 2,500 hours. At the expiration of the established service life, the critical part should be retired from service. Where the service life is limited by the arbitrary 2,500-hour figure, the service life can be extended beyond this figure after thorough inspection of several specimens which successfully reach the 2,500-hour limit. However, the upper limit to this extension is limited to 75% of the demonstrated fatigue life.

Example of Fatigue Life Determination From *S–N* Data

If the normal life of a specimen at a certain stress level is N, and if n be the number of cycles actually run at that level, then as a consequence of cumulative damage theory, a fatigue specimen stressed at several different stress levels in random order will fail when

$$\Sigma \frac{n_i}{N_i} = 1$$

Where the summation is taken over all values of i corresponding to the repeated stresses imposed on the specimen.

Using the above expression, it is possible to determine the fatigue life in hours of a part subject to random application of stresses above the endurance limit, if the fraction or percentage *of total life expectancy at each stress level* is known.

Thus, if:

L=total life of part in hours
x=life in hours at stress level (1)
y=life in hours at stress level (2)
a=fraction of total life at level (1)
b=fraction of total life at level (2)

then:

$$\frac{n_1}{N_1}=\frac{aL}{x}$$

$$\frac{n_2}{N_2}=\frac{bL}{y}$$

and

$$\Sigma\frac{n}{N}=\frac{aL}{x}+\frac{bL}{y}=1$$

or

$$L=\frac{100}{\frac{a}{x}+\frac{b}{y}}=\frac{100}{\Sigma\frac{\text{Percent of life for particular maneuver}}{\text{Endurance life in hours at that maneuver}}}$$

In the life determination, the highest measured stress associated with a particular maneuver should be used. Thus, in Table 1, the most critical steady hovering condition should be investigated (from minimum design r. p. m. to maximum design r. p. m.) at the most critical weight and center of gravity condition, and the 0.5% occurrence of hovering should be based on this critical condition.

This method can be illustrated further by referring to a specific example. Suppose that for only two maneuvers, lateral reversal and autorotation landing, the measured stresses are above the endurance limits. The life of the structure can be determined as follows:

TABLE 2

Flight Condition:	*Lateral Reversal Hovering, 300 r. p. m.*	*Auto-rotation Landing 320 r. p. m.*
1. Vibratory Stress (from flight test) psi	4, 900	2, 500
2. Steady Stress (from flight test) psi	8, 600	7, 690
3. Endurance in cycles (from S–N curve)	1.1×10^5	5.5×10^6
4. Cycles of critical stress per minute	300	320
5. Endurance in hours $\frac{(3)}{\text{cpm} \times 60}$	6. 11	286. 46
6. Percent of life at flight condition	1. 0%	2. 5%

$$L=\frac{100}{\frac{1}{6.11}+\frac{2.5}{286.46}}=\frac{100}{.1637+.0087}=580\text{ hrs.}$$

Service Life=75% of calculated life=435 hrs.

It should be noted that in the above example it is conservatively assumed that the peak stresses associated with each maneuver have been taken for the duration of the maneuver. Since in some cases this may be unduly conservative, the actual measured distribution of stress levels associated with each maneuver can be employed in the fatigue life determination than as a new section 6.343 because it is concerned with buoyancy. This change necessitates the inclusion of the word "hulls" in sestion 6.340.

The regulations covering Part 6 fuel systems are not in the same form and do not use terminology similar to that used in other airworthiness parts. To eliminate this inconsistency, new sections 6.418 and 6.419 are being adopted, and changes are being made to sections 6.420, 6.421, and 6.424.

Section 6.420 presently requires that, insofar as practicable, the entire fuel supply can be utilized under certain conditions. Such a requirement is unnecessary, even when practicable, because a rotorcraft will continue to be airworthy so long as usable fuel can be used regardless of the quantity of unusable fuel. Therefore, this provision is being deleted in favor of the objective requirement being added in section 6.418, which covers fuel system construction and arrangement to insure a satisfactory fuel flow.

Currently effective section 6.421 defines unusable fuel supply as that quantity at which the first evidence of malfunctioning occurs. This definition is unnecessarily restrictive and is not essential to safety since a rotorcraft is no less airworthy if an unusable fuel supply is selected as a quantity which is in excess of that which would produce a malfunction. Accordingly, the definition of unusable fuel supply is being revised to make it not less than the quantity at which the first evidence of malfunction occurs, the same as in other airworthiness parts.

As a result of comment received on Draft Release 61–12, specific requirements for demonstrations or tests are being deleted from sections 6.420 and 6.421. Adequate authority for any ground or flight tests which might be required continues to rest in presently effective sections 6.15 and 6.16. The provisions of paragraph (b) of section 6.421 as proposed are being transferred to a new paragraph (c) under section 6.420 as an editorial change, since the provision for fuel feed belongs more appropriately in the fuel flow section than in the unusable fuel supply section. In addition, the requirements for a low fuel quantity warning indicator presently in section 6.420(a), and a means to indicate when the emergency fuel system is in operation presently in section 6.424, are being transferred to section 6.604 which lists required items of equipment. In addition the powerplant operating limitation dealing with fuel is being brought up to date by including reference to turbine engine fuel in section 6.714.

Presently effective Part 6 contains no requirement pertaining to the bypass of engine oil around a filter element when the element becomes clogged. Although installation of a filter is not required, it is necessary to provide for the bypass of a clogged filter, if a filter is installed, to insure continued normal functioning of the rest of the oil system. Accordingly, a new section 6.447 is being adopted to provide for bypass capability, consistent with the same requirement now appearing in all the other airworthiness parts.

Revisions to the regulations concerning electrical systems and equipment are being made involving sections 6.617 through 6.627. These changes are being made in recognition of the substantial growth in capacity, complexity, and significance to safety of electrical systems on rotorcraft. In particular, new section 6.618 dealing with electric power sources is being added. Revisions are being made to sections 6.623, 6.626, and 6.627 concerned with master switch arrangement, protective devices, and electric cables, respectively. In conjunction with these changes, sections 6.623–1, 6.625–1, 6.625–2, and 6.627–1 are being deleted because the material in these sections is being included or is already contained, in other sections.

Two changes are being made to the lighting requirements. Figure 6–2 now specifies that position light intensity for angles 40° to 90° above or below the horizontal be at least 2 candles. Because this results in an irrational discontinuity when related to the other data in figure 6–2, figure 6–2 is being amended to require an intensity of 0.05 I for these angles.

The current anticollision light requirements in section 6.637(a) permit 0.03 steradians blockage. In view of recent qualitative studies, it has been determined that such a limitation might be unduly restrictive. Therefore, section 6.637(a) is being amended to permit 0.5 steradians of obstruction.

Part 6 currently does not require the tail rotor to be marked. Because there have been a number of accidents attributable to persons walking into tail rotors, section 6.738(f) is being added to require that tail rotors be marked conspicuously.

Miscellaneous changes of an editorial or clarifying nature are being made to sections 6.11, 6.203, 6.237, 6.251, 6.306, 6.605, 6.642, and 6.738. Among the miscellaneous amendments there is one to expressly exclude from the provisions of section 6.11(b) consideration of provisional type certificates. While it was proposed that this be accomplished by a note, it now appears that it is more appropriate to include such a provision within section 6.11(b) rather than as a note thereto.

Interested persons have been afforded an opportunity to participate in the making of this amendment, and due consideration has been given to all relevant matter presented.

Amendment made the following changes;

(1) Amended sections 6.11(b), 6.203(d), 6.225, 6.237(a), 6.251, 6.306(c), 6.307(b), 6.340, 6.341, 6.420, 6.421, 6.424, 6.605(d) 6.620, 6.622, 6.623, 6.625, 6.626, 6.627, 6.637(a), 6.642(a), 6.714(c), and 6.738(b)(1), and Figure 6–2;

(2) Deleted sections 6.623–1, 6.625–1, 6.625–2, and 6.627–1;

(3) Redesignated section 6.621 as section 6.619 and added a new section 6.621; and

(4) Added sections 6.226, 6.418, 6.447, 6.604 (l) and (m), 6.617, 6.618, and 6.738(f).

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