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Rotorcraft Airworthiness

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§ 6.18–1 *Approval of aircraft components (CAA rules which apply to § 6.18).* Aircraft components made the subject of technical standards orders shall be approved upon the basis and in the manner provided in Part 514 of this title (Regulation of the Administrator).

§ 6.203–1 *Fixed or ground adjustable stabilizing surfaces (CAA policies which apply to § 6.10 and § 6.203 (b)).* The purpose of § 6.203 is to require the testing of certain components which in the details of their construction, operational characteristics, or loading, do not lend themselves to established and reliable methods of analysis. In this regard, proof testing such items as fixed or ground adjustable stabilizing surfaces is not considered a minimum requirement and will not be necessary provided sufficient experience has been accumulated from previous satisfactory designs, methods of analysis and tests to justify acceptance of these components on the basis of structural analysis. Therefore, these components may be regarded structurally the same as any other part of the basic airframe.

§ 6.221–1 *Service life of auxiliary rotor assemblies (CAA interpretations which apply to § 6.221).* The requirement in § 6.221 that vibration stresses in highly stressed metal components of auxiliary rotors must not exceed safe values for continuous operation is interpreted to mean that the service life of such components should be determined by fatigue tests or by other methods found acceptable by the Administrator. The methods of service life determination for main rotor structure outlined under § 6.250–1 are considered to be acceptable in showing compliance with the pertinent portion of § 6.221.

§ 6.231–1 *Distribution of vertical ground reaction loads and determination of angular inertia loads (CAA interpretations which apply to § 6.231 (b) (2)).* (a) Although § 6.231 (b) (2) states that the vertical loads are those specified in § 6.231 (b) (1), the distribution of the vertical loads among the ground reaction points is not necessarily the same for the two subparagraphs since the requirements of § 6.230 must be met. Section 6.230 (a) states, in part, that the external loads shall be placed in equilibrium with the linear and angular inertia loads in a rational or conservative manner.

(b) Compliance with § 6.231 (b) (2) is interpreted to require that a vertical inertia load of nW and a horizontal inertia load of 0.25 nW be applied at the center of gravity. For the level landing with drag on all wheels, the vertical ground reaction loads should be distributed between the forward and rear wheels to place the ground reaction loads in equilibrium with the rotorcraft linear inertia loads. For the level landing with drag on main wheels only, the pitching moments arising from the vertical and horizontal ground reactions should be placed in equilibrium with an angular inertia load about the c. g.

(c) The drag load at each wheel, in both cases, is required to be equal to 0.25 times the respective wheel vertical load.

§ 6.250–1 *Service life of main rotors (CAA policies which apply to § 6.250 (a)).*

Several methods which have been found acceptable by the Administrator for determining the service life of main rotors are outlined in Appendix A of this section for the guidance of the industry in complying with § 6.250 (a).

§ 6.355–1 *Application of loads (CAA policies which apply to § 6.355).* The actual forces acting on seats, berths, and supporting structure in the various flight, ground and emergency landing conditions will consist of many possible combinations of forward, sideward, downward, upward, and aft loads. However, in order to simplify the structural analysis and testing of these structures, it will be permissible to assume that the critical load in each of these directions, as determined from the prescribed flight, ground, and emergency landing conditions, acts separately. If the applicant desires, selected combinations of loads may be used, provided the required strength in all specified directions is substantiated. (TSO C–25 Aircraft Seats and Berths, outlines acceptable methods for testing seats and berths.)

[§ 6.625–1 *Automatic reset circuit breakers (CAA policies which apply to § 6.625).* Automatic reset circuit breakers (which automatically reset themselves periodically) should not be applied as circuit protective devices.[1] They may be used as integral protectors for electrical equipment (e. g., thermal cut-outs) provided that circuit protection is also installed to protect the cable to the equipment.

[[1] Circuit protective devices are normally installed to limit the hazardous consequences of overloaded or faulted circuits. These devices are resettable (circuit breakers) or replaceable (fuses) to permit the crew to restore service when nuisance trips occur or when the abnormal circuit condition can be corrected in flight. If the abnormal circuit condition can *not* be corrected in flight, the decision to restore power to the circuit involves a careful analysis of the flight situation. It is necessary to weigh the essentiality of the circuit for continued safe flight against the hazards of resetting on a possibly faulted circuit. Such evaluation is properly an aircraft crew function which can not be performed by automatic reset circuit breakers. To assure crew supervision over the reset operation, circuit protective devices should be of such design that a manual operation is required to restore service after tripping.]

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 photoelastic 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 I (in next section) 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 I 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 take-off	.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 (Take-off 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 I) fatigue tests of the actual component are required. 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.

[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).

However, flight or whirl stand testing is acceptable in lieu of laboratory testing if they are conducted under controlled conditions.

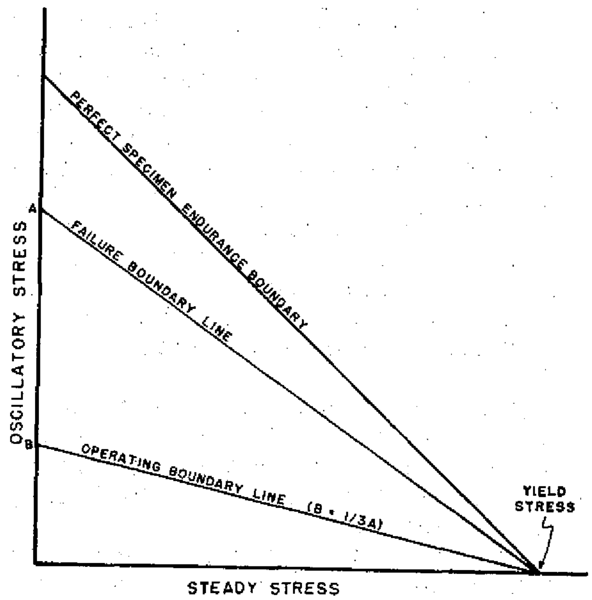

Figure I

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 criterial 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 I 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 I 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 5×10^7 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. 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 I), 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 I, 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 II

	Lateral Reversal Hovering, 300 r. p. m.	*Autorotation Landing 320 r. p. m.*
Flight Condition:		
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 $\left(\frac{(3)}{\text{cpm} \times 60}\right)$	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.

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