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For Information Only

[Reg. Docket No. 1518; Supp. No. 20]

PART 6—ROTORCRAFT AIRWORTHINESS; NORMAL CATEGORY SERVICE LIFE OF MAIN ROTORS

Appendix A—Main Rotor Service Life Determination

The policy expressed in presently effective § 6.250-1 sets forth by reference to Appendix A acceptable methods of compliance with the provisions of § 6.250 related to the establishment of service life of main rotors.

Appendix A contains those fatigue evaluation procedures which are acceptable methods for determining the service life of main rotors. However, the present Appendix was not filed with the Office of the Federal Register and is therefore not presently set forth in the Code of Federal Regulations. The purpose of this regulatory action is to revise the current Appendix by up dating the fatigue evaluation procedures in line with current industry practice and to publish the revised Appendix in the FEDERAL REGISTER. In connection with the revision to Appendix A, a minimum reduction of 20 percent in the S-N test data curve has been introduced to account for the scatter inherent in the results of fatigue life tests. This reduction in the S-N curve makes the revised Appendix more conservative than the present Appendix and corresponds with the procedures which are typical of present practice by the rotorcraft industry.

Since this regulatory action relates only to a statement of policy, notice and public procedure hereon are unnecessary and it may be made effective on less than 30 days' notice.

In consideration of the foregoing, Appendix A to Part 6 of the Civil Air Regulations (14 CFR Part 6), is hereby revised to read as hereinafter set forth, effective December 14, 1962:

(Sec. 313(a), 601, 603; 72 Stat. 752, 775, 776; 49 U.S.C. 1354, 1421, 1423)

Issued in Washington, D.C., on December 10, 1962.

N. E. HALABY,
Administrator.

(As published in the Federal Register /27 F.R. 12400/ December 14, 1962

Appendix A

MAIN ROTOR SERVICE LIFE DETERMINATION

1. *Introduction.* The fatigue evaluation procedures outlined in this appendix are acceptable to the Federal Aviation Agency for showing compliance with the fatigue evaluation requirements of CAR 6.250. However, the information contained in this appendix is for guidance purposes only and is not mandatory.

(a) The rotorcraft is perhaps more directly affected by fatigue than any other type of aircraft. The primary structural elements and systems are subject to vibratory stresses in practically every regime of flight. In addition, being a highly maneuverable aircraft that is capable of forward, rearward, sideward, vertical, and rotational flight, operating limitations due to fatigue are possible in practically all flight situations. For those reasons, it is important that special attention be focused on the fatigue strength evaluation of the essential parts of the rotorcraft.

(b) Although a uniform approach to fatigue evaluation is desirable, it is recognized that in such a complex problem, new design features and methods of fabrication, or new approaches and configurations may require variations and deviations from the procedures described herein. Engineering judgment should therefore be exercised for each particular application.

(c) There is some question whether a completely rational method exists for the prediction of fatigue life in a built-up structure subject to random loading. Nevertheless, 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 approximation involved in the hypothesis and the lack of an adequate theory connecting the hypothesis with more basic properties of the materials, it attempts to take more factors into account than any other method developed thus far.

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

(1) The stresses associated with the flight maneuvers and operating conditions expected;

(2) The frequency of occurrence of specific loadings expected; and

(3) The fatigue strength characteristics of the structure.

2. *Flight strain measurement program.* It is generally agreed that it is not possible at present to determine analytically the stress levels associated with normal rotorcraft operation and the correlation of occurrence of critical stresses with specific maneuvers or operating conditions. Therefore, the stress levels and occurrence of critical stresses must be determined by a carefully controlled flight strain measurement program.

(a) *Instrumentation.* The instrumentation system used in the flight strain measurement program should accurately measure and record the critical strains and test conditions associated with normal operation and specific maneuvers. The location and distribution of the strain gages should be based on a rational evaluation of the critical stress areas. This may be accomplished by a qualitative study by means of brittle coatings (such as stresscoat), by photoelastic methods, or by appropriate analytical means. In any event, the distribution and number of strain gages should define the load spectrum adequately for each part essential to the safe operation of the rotorcraft.

(1) The corresponding flight parameters (airspeed, rotor rpm, center of gravity accelerations, etc.) should also be recorded simultaneously by appropriate methods. This is necessary in order to correlate the loads and stresses with the maneuver or operating condition at which they occurred.

(2) The instrumentation system should be adequately calibrated and checked periodically throughout the flight strain measurement program in order to insure consistent results. Sufficient calibration data should be submitted with the fatigue evaluation program to substantiate the results obtained.

(b) *Parts to be strain-gaged.* The main rotor blades, rotor hub assembly, controls, tail rotor, and directional control system should be strain-gaged. For rotorcraft of unusual or unique design, special consideration might be necessary to insure that all of the essential parts are evaluated.

(c) *Flight regimes and conditions to be investigated.* The flight regimes to be investigated in the flight strain measurement program for power-on and power-off operation are shown in figures I and II. For clarity, the parameters which define these regimes are included in these figures. As noted on figure I, complete coverage at 111% V_{NE} should be demonstrated for power-on operation. However, for power-off operation, figure II, complete coverage at 111% V_{NE} for maximum and minimum design rpms need not be obtained if points are obtained at V_{NE} at both maximum and minimum design rpm and at 111% V_{NE} at both maximum and minimum placarded rpms as indicated in the figure. In addition, if the high speed points are not obtainable at the low rpms, it is acceptable to vary the V_{NE} and 111% V_{NE} speed with rotor rpm as shown in the figures.

(1) The determination of flight conditions to be investigated in the flight strain measurement program should be based on the anticipated use of the helicopter and, if available, on past service records for similar designs. In any event, the flight conditions considered appropriate for the design and application should represent those which will occur in actual operation. Suggested flight conditions for single-engine helicopters used in normal operation are shown in table I, which should be used as a guide in making this determination. In the case of multi-engine helicopters, the flight conditions concerning partial engine-out operation should be considered in addition to complete power-off operation. The flight conditions to be investigated should be submitted, in a form similar to table I, in connection with the flight evaluation program.

(2) The severity and rapidity of control movement used in control reversals, and the extent of blade stall investigated during the flight strain measurement program, should be at least as severe as that which would occur in service. In determining the severity and rapidity of control movement and blade stall, consideration should be given to inadvertent overshoots during training as well as normal service.

(3) All flight conditions considered appropriate for the particular design should be investigated over the complete rpm, airspeed, center of gravity, altitude, and weight ranges in order to determine the most critical stress levels associated with each flight condition. In order to account for data scatter and to determine the stress levels present, a sufficient number of measured strain points should be obtained at each flight condition. In some instances, the critical weight, center of gravity, and altitude ranges for the various maneuvers can be based on past experience with similar designs. This procedure is acceptable where adequate flight tests are performed to substantiate such selections. The combinations of flight parameters that produce the most critical stress levels should be used in the fatigue evaluation.

3. *Frequency of loading.* (a) At best, the determination of the percentage of total operating time associated with each flight maneuver can only be accomplished by a statistical approach and will of necessity be a function of the purpose for which the particular helicopter is intended. Obviously, a helicopter used only for crop dusting would have a different time distribution than one used for mail or passenger service.

(b) The importance of establishing representative percent of occurrences for each flight condition cannot be overemphasized. Therefore, the times alloted should be based on sound engineering judgment and past service history if available. Table I, which contains suggested percent of occurrences along with suggested flight condition for single-engine helicopters used in normal operation, should be used as a guide in establishing appropriate time to be alloted for the various maneuvers.

4. *Fatigue strength.* The third phase of the fatigue evaluation program is the determination of the fatigue strength of the parts. Although there is information available on the fatigue strength characteristics of material specimens, the direct application of such information to built-up structures is questionable. However, data from tests of "perfect" specimens can undoubtedly be an important tool in design if corrected by appropriate stress concentrations and safety factors. Nevertheless, there are various other factors which affect the fatigue strength of a built-up structure which cannot be accounted for to a reasonable degree of accuracy. Therefore, it is usually necessary that the essential parts be subjected to repeated load tests simulating the critical loading conditions determined in the flight strain measurement program. Special operational or functional characteristics which could affect the fatigue strength should also be considered in the service life evaluation. Such factors as high blade operating temperatures due to tip jets or turbine exhaust impingement on the tail rotor should be considered as well as other special operating conditions. In addition, effects of special purpose use such as hoist and sling operation, spraying, surveying, etc., should be considered if appropriate to the particular type. The fatigue strength should be determined by either of the following methods, but the testing method is recommended because of the limitations of the analytical approach:

(a) *Analytical method.* Although it has been pointed out that correlating material fatigue data with that of a built-up structure is difficult, it is recognized that if maximum allowable stress levels are established by acceptable means, and the maximum stresses measured in flight are lower than these established levels, no fatigue testing is necessary. The following technique, based on the use of the Goodman diagram, is considered acceptable for establishing this maximum allowable stress level:

(1) Determine the endurance boundary for the perfect specimen from material data obtained from laboratory tests. The perfect laboratory specimen should be representative of the material used in the actual structure in regard to basic strength properties, and without stress concentrations. Referring to figure III, the endurance boundary for the perfect specimen may be represented by a straight line drawn through the yield stress (point A on the horizonal axis) and the maximum oscillatory stress which the particular specimen can withstand for an infinite number of cycles (point B on the vertical axis.) The maximum oscillatory stress should be based on laboratory specimens tested without failure to at least 5 x 10^7 cycles for nonferrous materials or 10^7

cycles for ferrous materials. The line AB then represents the upper boundary of the combinations of oscillatory and steady stresses which the perfect specimen can withstand without failure.

(2) The allowable full reversal stress as determined in (1) should then be reduced to account for the stress concentrations that are present in the actual part. The stress concentration factor chosen should be adequate to account for surface conditions, fabrication methods, fretting, and size and shape effects, as well as stress concentrations around bolts, threads, fillets, notches, and rivets. The resulting line AC on the Goodman diagram represents the failure boundary line for the actual part. The selection of an adequate stress concentration factor to account for the above conditions, particularly size and shape effects and fretting should be based on sound engineering judgment and past experience.

(3) A factor of safety of 3 should then be applied to the failure boundary line to establish the operating boundary line AD. The slope of line AD would be one-third of line AC.

(4) If the flight strain measurements indicate that all of the operating stresses fall below the operating boundary line (AD), no fatigue testing is necessary. When the measured stresses are above the operating boundary line, however, fatigue testing of the actual parts is necessary.

(b) *Limitations of the analytical method.* Caution should be exercised in the application of the Goodman diagram method, particularly when the following items are involved:

(1) Large parts in proportion to the laboratory specimens;

(2) Irregularly shaped parts containing numerous fillets, holes, threads, or lugs;

(3) Parts of unique design for which no past service experience is available.

(4) Parts subject to fretting; and

(5) Bolted or pinned connections.

In view of these limitations and the difficulty in selecting an adequate overall stress concentration factor, many helicopter manufacturers establish the operating boundary line, AD (figure III) on the Goodman diagram from data based on actual tests and service experience. This method is considered acceptable provided sufficient data are used to substantiate the allowables.

(c) *Testing methods.* The fatigue strength characteristics of the essential parts of the helicopter should be determined by any of the test methods described below or other test methods which can be shown to provide similar results. (Since lack of quality control can 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 tests.)

(1) *S–N Curves.* (i) The establishment of a family of S–N curves is an acceptable method for determining fatigue strength of the essential parts. The establishment of each S–N curve involves testing a sufficient number of parts at the same steady stress level and varying the oscillatory stress. Thus, in figure IV, if at a steady stress level A and an oscillatory stress of level B, the part is tested until failure, failure occurring at N_1 cycles, a point on the S–N curve for steady stress of level A is determined. Additional points on the S–N curve representing a steady stress of level A may be determined by choosing a different oscillatory stress level and testing the part to failure. If no failure occurs for a specific loading condition, after 10^7 cycles for ferrous materials or after 5 x 10^7 cycles for nonferrous materials, the part can be considered to have infinite life at that stress level. However, in the case of nonferrous materials, it is acceptable to test to 10^7 cycles provided the extension of the curve to 5 x 10^7 cycles is established by suitable means.

(ii) To compensate for the scatter usually associated with fatigue testing, a large number of test specimens is desirable in establishing each S–N curve. However, most manufacturers cannot afford the cost and time necessary to obtain such accuracy. Therefore, a minimum of 4 test specimens which will establish a well defined curve over the range of oscillatory stress levels expected to occur in service is considered acceptable in establishing each S–N curve. In order to compensate for the scatter associated with fatigue testing, the mean S–N curves should be reduced by an appropriate factor. This factor, which should be applied to the stress axis, should be based on the type of material being tested, past service experience with the material, and type of design. For materials and designs for which service experience is available, a factor of not less than 20 percent is considered acceptable. However, for new materials or designs this factor should be appropriately increased. The shape of the resulting reduced curve should be based on typical published S–N data and all of the test points should fall above the reduced curve. This curve would then represent the S–N curve for use in determining the fatigue lives. Figure IV represents this method of constructing a typical S–N curve based on test specimens. In this example, a reduction factor of 20 percent was used for explanatory purposes only. A separate S–N curve should be established for each critical steady stress level determined in the flight strain measurement survey. If it is desired to limit the fatigue tests, a single S–N curve based on the highest measured steady stress may be used in the fatigue life calculations. However, if this approach tends to unduly limit the fatigue life, a family of curves may be developed from two established S–N curves by means of Goodman or similar diagrams or by rational methods. Caution should be exercised in extrapolating test data by means of straight line Goodman diagrams, particularly from a lower alternating stress to a higher alternating stress since the results may be unconservative.

(2) *Cyclical units.* The establishment of fatigue life based on cyclical unit method involves the following:

(i) Determining by flight test the damaging stress levels associated with each flight maneuver considered appropriate for the particular helicopter;

(ii) Determining the number of cycles the damaging stress levels occur during each maneuver based on the expected percentage of occurrence; and

(iii) Testing of each essential part at all of the damaging stress levels for the corresponding number of cycles, representing the expected maneuver history. Since the fatigue life of the parts is unknown beforehand, the damaging stress levels must be covered in arbitrarily chosen cyclical units. For example, if cyclical units of 100 hours are chosen, then reference to table I would indicate that the damaging stress levels and number of cycles corresponding to 0.5 hours at rapid increase of rpm on the ground for quickly engaged clutch, 0.5 hours at jump takeoff, 1.0 hours at 20% V_{NE} for level flight, and so on throughout the maneuver history, should be included during each testing unit of 100 hours. A minimum of 4 specimens should be used and the fatigue life of the part or component should be based on the smallest number of completed units. Thus, if the smallest number of completed units for the 4 test specimens is 14, then the fatigue life for this part would be based on 1400 hours. It should be noted that the Cumulative Damage Hypothesis on which this method is based 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 also desirable to keep the units of time at reasonably low levels.

(3) *Combination of S–N curves and cyclical units.* Another method of determining fatigue strength would be by the combination of S–N curves and cyclical units. This would involve the determination of the knee of the S–N curve (endurance limit) and the flight conditions which resulted in stresses below the endurance limit. The stresses which fall below the endurance limit are considered to have no effect on the fatigue life. The method of cyclical units would then be applied only to those flight conditions resulting in stresses which would cause fatigue damage. Thus, if it is established that all level flight conditions result in stresses which are below the endurance limit, the actual testing would be greatly reduced.

(4) *Whirl stand testing.* Another method of determining the fatigue life of the essential parts involves the use of a whirl test stand on which the entire rotor assembly is tested for the loads determined in the flight strain program. The fatigue life would be based on the minimum number of hours completed without failure for the most critical stress levels determined in flight. This method is only valid when the critical loads determined in the flight strain survey can be duplicated accurately.

(d) *Finite service life.* Since actual operating conditions might involve factors which cannot be ascertained by testing, it becomes desirable to establish an operational time limit, or service life, after which the part should be removed from service. Therefore, to compensate for these factors, the service life should be established in accordance with the following formulas as applicable:

(1) Calculated service life, L_c, $\leq$3,350 hours
Service life, $L = 0.75\ L_c$ hours

(2) Calculated service life, L_c, $\geq$3,350 hours
Service life, $L = 0.375\ L_c + 1{,}250$ hours.

(e) *Infinite service life.* Infinite life of a particular part or component may be established by demonstrating that all of the critical operating stresses, as determined by the flight strain survey, are below the endurance limit. This may be demonstrated by either of the following methods:

(1) If all of the critical operating stresses fall below the operating boundary line on the Goodman diagram (figure III) no fatigue testing is necessary.

(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 x 10^7 cycles for nonferrous materials. The minimum number of test specimens is dependent on the oscillatory test level in the following manner:

(i) A minimum of 4 test specimens if the oscillatory level is chosen at 1.1 times the critical oscillatory stress level;

(ii) A minimum of 3 test specimens if the oscillatory level is chosen at 1.25 times the critical oscillatory stress level;

(iii) A minimum of 2 test specimens if the oscillatory level is chosen at 1.5 times the critical oscillatory stress level; and

(iv) One specimen if the oscillatory level is chosen at 2 times the critical oscillatory stress level.

(f) *Extension of service life.* The following conditions should be met to extend service life beyond the initial retirement life established in accordance with equation (2) of paragraph (d) of this section.

(1) A sufficient number of identical parts which represent an adequate sampling of operation should successfully reach the initial retirement life.

(2) The parts should be thoroughly inspected for wear, fretting, cracking, etc., by appropriate methods.

If these conditions have been satisfied and the parts found to be free from defects, an increase in service life might be granted. The upper limit of service life which might be granted under these extension provisions is 75 percent of the demonstrated fatigue life. It is advisable to approach the 75 percent figure in several increments of life extension.

5. *Sample calculation based on S-N data.* The Cumulative Damage Hypothesis states 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. Thus, if a part is subjected to random loading for n_1 cycles at a stress level of S_1, n_2 cycles at S_2, n_3 cycles at S_3, and so on, and if N_1, N_2, N_3 are the corresponding number of cycles to failure for each stress level, then failure will occur with the summation:

$$\frac{n_1}{N_1}+\frac{n_2}{N_2}+\frac{n_3}{N_3}+\frac{n_i}{N_i}=1, \text{ or } \sum\frac{n_i}{N_i}=1$$

Using this expression, the calculated service life of a part subjected to random loading can be determined if the percent of life used per hour at each damaging stress level is known. The percent of life used per hour at each damaging stress level, can be expressed by

$$1=\frac{an}{N} \qquad (1)$$

where:

- 1 = percent of life used per hour at the damaging stress level;
- a = percent of total operating time allotted to the flight condition during which the damaging stress level was recorded;
- N = total number of cycles of the damaging stress level at failure; and
- n = number of cycles the damaging stress level occurs per hour.

Thus, the calculated service life, L_c, of a particular part or component subjected to a random number of damaging stress levels, would be

$$L_c=\frac{100}{1_1+1_2+1_3+\ldots 1_i}=\frac{100}{\Sigma 1_i} \qquad (2)$$

A sample calculation illustrating this method for determining the calculated service life is shown in table II. In this example, the peak steady and vibratory stress levels associated with each maneuver have been assumed to occur for the duration of the maneuver (columns 3 and 4). In addition, the cycles of oscillatory stress per hour also has been conservatively assumed at the maximum level throughout the flight spectrum (column 5). If this procedure tends to limit the service life unduly, it is acceptable to use the actual measured stress level distributions if proper account of possible variations is provided by repeated maneuvers. The number of cycles to failure for each damaging stress level (column 6) was determined from figure V. As an example, consider flight condition II(c) of table II. The percent of total operating time (a) considered at this maneuver is 0.5%, the damaging oscillatory stress level is 10,500 psi, the number cycles of damaging stress per hour (n) is 23,200 and the number of cycles to failure (N) from the S-N curve (figure V) is 3,200,000 cycles. Then by equation (1) the percent of life used per hour at this damaging stress level would be

$$1=\frac{an}{N}=\frac{0.5\times 23{,}000}{3{,}200{,}000}=0.00362$$

The summation of the individual percentages of life used per hour for each damaging stress level is shown in column 7. Therefore, by equation (2), the calculated service life of this part would be

$$L_c=\frac{100}{\Sigma 1_i}=\frac{100}{0.15289}=654 \text{ hours}$$

The service life of this part would be, as explained in paragraph (d) of section 4.

$$L=0.75\times 654$$
$$L=490 \text{ hours}$$

A summary of the measured stress and percent life used at the various flight conditions should be submitted with the fatigue evaluation program in a form similar to table II.

Table I

Percent Occurrence

I. Ground conditions		1.5
(a) Rapid increase of rpm on ground to quickly engage clutch	0.5	
(b) Taxiing with full cyclic control	.5	
(c) Jump takeoff	.5	
II. Hovering		2.0
(a) Steady hovering	.5	
(b) Lateral reversal	.5	
(c) Longitudinal reversal	.5	
(d) Rudder reversal	.5	
III. Forward flight power on		87.5
(a) Level flight—20% V_{NE}	1.0	
(b) Level flight—40% V_{NE}	3.0	
(c) Level flight—60% V_{NE}	18.0	
(d) Level flight—80% V_{NE}	25.0	
(e) Maximum level flight (but not greater than V_{NE})	15.0	
(f) V_{NE}	3.0	
(g) 111% V_{NE}	.5	

Table I—Continued

Percent Occurrence—continued

(h) Right turns—30, 60, 90% V_{NE}	3.0	
(i) Left turns—30, 60, 90% V_{NE}	3.0	
(j) Climb (takeoff/power)	2.0	
(k) Climb (max. continuous power)	4.0	
(l) Change to autorotation from power-on flight—30, 60, 90% V_{NE}	1.5	
(m) Partial power descent (including condition of zero flow through rotor)	2.0	
(n) Cyclic and collective pull-ups from level flight	1.0	
(o) Lateral reversals at V_H	.5	
(p) Longitudinal reversals at V_H	.5	
(q) Rudder reversals at V_H	.5	
(r) Landing approach	3.0	
(s) Sideward flight	.5	
(t) Rearward flight	.5	
IV. Autorotation—power off		9.0
(a) Steady forward flight	2.0	
(b) Rapid power recovery from autorotational flight	.5	
(c) Right turns—30, 60, 90% V_{NE}	1.0	
(d) Left turns—30, 60, 90% V_{NE}	1.0	
(e) Lateral reversals	.5	
(f) Longitudinal reversals	.5	
(g) Rudder reversals	.5	
(h) Cyclic and collective pull-ups	1.0	
(i) Landings (including flares)	2.0	
	100.0	100.0

Table II—Determination of Service Life

(Sample Calculation)

1 Flight condition Table I	2 Percent occurrence Table I	3 Critical steady stress	4 Critical oscillatory stress	5 Cycles oscillatory stress Cycles/Hr	6 Cycles to failure Figure V	7 Percent of life used per hour
I (a)	0.5	Level A	1900	23,200		
(b)	.5	do	2100	23,200		
(c)	.5	do	2300	23,200		
II (a)	.5	do	2600	23,200		
(b)	.5	do	9600	23,200	5,000,000	0.00232
(c)	.5	do	10500	23,200	3,200,000	.00362
(d)	.5	do	3400	23,200		
III (a)	1.0	do	6500	23,200		
(b)	3.0	do	5100	23,200		
(c)	18.0	do	7700	23,200		
(d)	25.0	do	8100	23,200		
(e)	15.0	do	8300	23,200	16,000,000	.02175
(f)	3.0	do	8900	23,200	8,400,000	.00829
(g)	.5	do	9100	23,200	7,400,000	.00157
(h)	3.0	do	11200	23,200	2,400,000	.02900
(i)	3.0	do	11400	23,200	2,250,000	.03093
(j)	2.0	do	10900	23,200	2,700,000	.01719
(k)	4.0	do	9900	23,200	4,100,000	.02263
(l)	1.5	do	7800	23,200		
(m)	2.0	do	6700	23,200		
(n)	1.0	do	9700	23,200	4,800,000	.00483
(o)	.5	do	7800	23,200		
(p)	.5	do	7900	23,200		
(q)	.5	do	7300	23,200		
(r)	3.0	do	6700	23,200		
(s)	.5	do	6700	23,200		
(t)	.5	do	7900	23,200		
IV (a)	2.0	do	7100	23,200		
(b)	.5	do	9700	23,200	4,800,000	.00242
(c)	1.0	do	9300	23,200	6,300,000	.00368
(d)	1.0	do	9900	23,200	4,100,000	.00566
(e)	.5	do	6800	23,200		
(f)	.5	do	6100	23,200		
(g)	.5	do	5900	23,200		
(h)	1.0	do	7600	23,200		
(i)	2.0	do	7900	23,200		
Total	100.0					.15289

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FIGURE I

POWER ON

Rotor R.P.M. Airspeed Envelope

NOTE: Dashed lines indicate test boundaries. Cross-hatched areas indicate operating regimes.

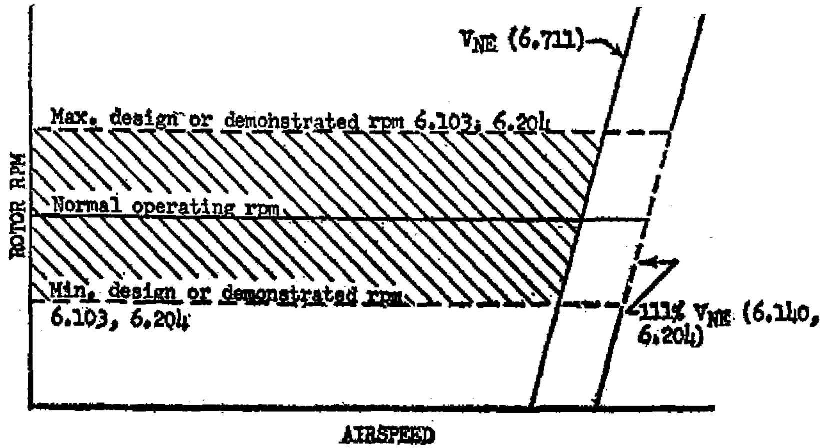


FIGURE III

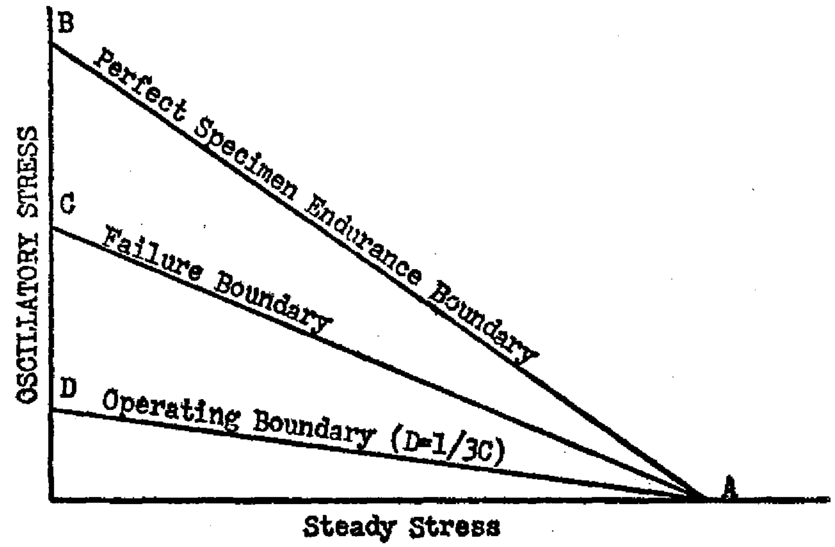


FIGURE II

POWER OFF

Rotor R.P.M. Airspeed Envelope

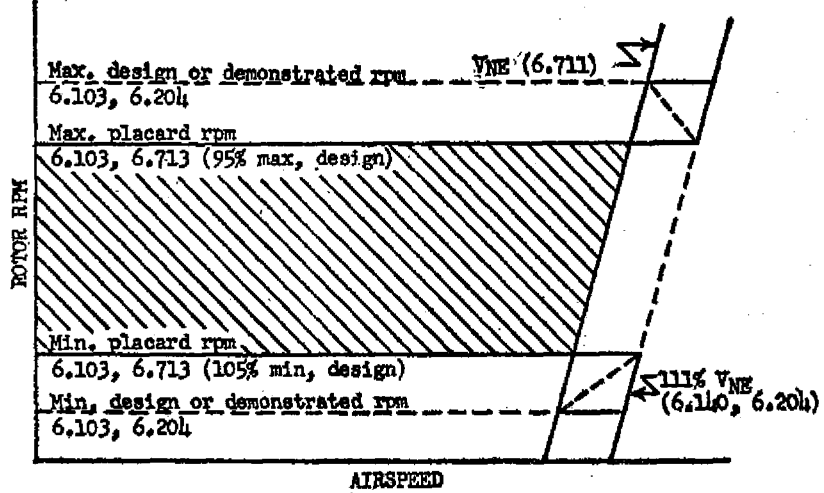


FIGURE IV

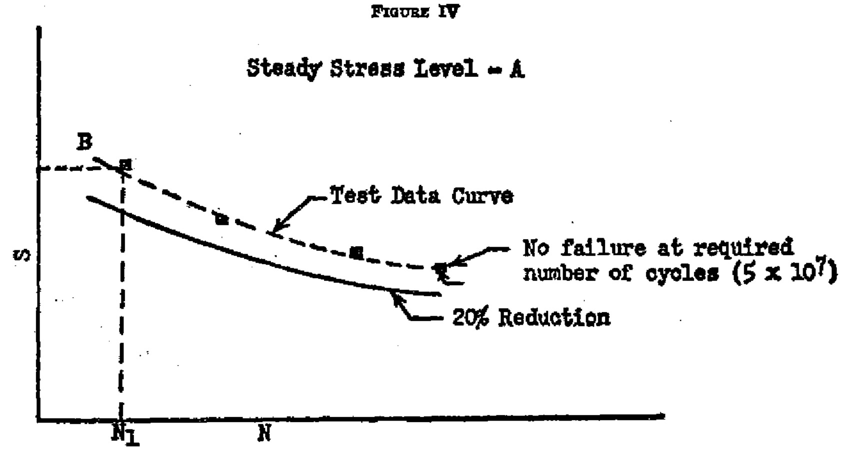
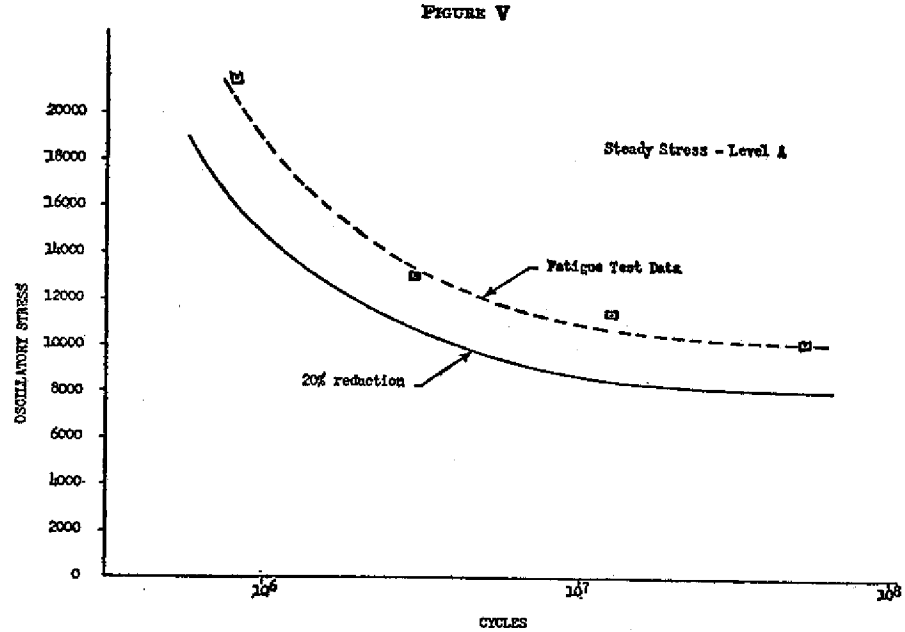


FIGURE V



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