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Some Fatigue Tests on Notched Specimens with Programme Loading for a "Ground-Attack" Aircraft

by

W. A. P. Fisher, B.A., A.F.R.Ae.S.

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ROYAL AIRCRAFT ESTABLISHMENT

SOME FATIGUE TESTS ON NOTCHED SPECIMENS WITH PROGRAMME LOADING FOR A "GROUND-ATTACK" AIRCRAFT

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RAE Ref: Structures E4/13489/WAPF

SUMMARY

A series of programme loading tests was made in a hydraulic fatigue testing machine to check the validity of the Miner Cumulative Damage Hypothesis for a structural light alloy notched specimen in axial tension. Test loads were based on an accelerometer load spectrum and calculated according to ultimate factors of 11, 13 and 15. Additional tests were made with higher peak loads.

For this type of specimen and for this shape of load spectrum the Miner Hypothesis is somewhat conservative.

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1 INTRODUCTION

In the ideal fatigue test on an aircraft structure, loads of various magnitudes would be applied in a random order, as occurs in service. In the laboratory, however, it is more convenient to apply a programme of loads in which the number of applications varies with the magnitude of the load in the same ratios as those determined by accelerometer counts over a long flight period. In such a programme the load is usually raised or lowered in steps. The programme is chosen to represent a convenient number of flying hours and is repeated over and over again until failure occurs. The endurance is expressed in terms of "programmes to failure". There is evidence,[1,2] that programme loading is virtually equivalent to random loading provided there is an adequate number of repetitions of the programme before failure. A.O. Payne[2] has reported tests on aircraft wings which give close agreement between the results of programme tests and random loading.

The present Note gives the results of some programme fatigue tests on notched aluminium alloy bars in axial loading using a hydraulic testing machine. The loading spectrum was appropriate to a ground attack type of aircraft, and the assumption was made that the load spectrum, in terms of g acceleration, was determined by the pilot only and was independent of the ultimate strength of the aircraft. Ultimate design factors of 11, 13 and 15 were chosen and the applied loads arranged to give the appropriate nominal stresses for each of these three cases. The results were compared on the basis of the simple Palmgren - Miner hypothesis* of cumulative fatigue damage.

2 TEST SPECIMENS

The test specimen used is shown in Fig.1. For cheapness and simplicity, a wing spar is represented by a notched light alloy rectangular bar designed to have a theoretical stress concentration factor of 3.65. The material was obtained from one manufacturer in the form of extruded bar to specification DTD 363A**. All the material had been supersonically crack detected.

3 SCOPE OF TESTS

Two specimens were first tested statically to determine the static failing load.

Next, fourteen specimens were tested under repeated loading from a minimum load of 1 Tonne (0.982 tons) to various percentages of the static failing load so as to obtain the appropriate endurance curve.

In the subsequent programme loading, series (a), (b) and (c) the nominal stresses in the specimen were calculated for ultimate design factors of 11, 13 and 15 respectively, using, in each case, the same basic load spectrum (curve 'a' in Fig.2) in terms of 'g'***.

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* According to this hypothesis, failure occurs when $\frac{n_1}{N_1} + \frac{n_2}{N_2} + \dots = 1$, where, in complex loading, n_1 n_2 ... etc. are the respective numbers of cycles applied at load levels 1, 2 ... etc. and in simple loading N_1, N_2 ... etc. are the numbers of repetitions of the individual loads 1, 2 ... etc. to failure. The hypothesis is often called, simply, the 'Miner hypothesis'.

** A summary of this Specification is given in the Appendix.

*** These nominal design factors are used as a convenient measure of working stress levels.

The number of load applications per programme was chosen so as to give a theoretical life of about 10 programmes for the 11g case, and therefore more than 10 for the higher factors (i.e. lower stresses) of cases (b) and (c).

The programme loads and cycles were obtained as follows:-

The acceleration range in Fig.2, i.e. from 2g to 7g, was divided into ten equal intervals of 0.5g as shown. The number of occurrences for each interval was obtained by subtraction (see table on Fig.2).

For any interval x to x+ 0.5g, the continuously changing load was represented by a fixed load of x + 0.2g as a weighted mean.

The ten values of fixed load (2.2g, 2.7g, 3.2g .... 6.7g) were converted to percentages of ultimate design load according to the ultimate design factor. These percentages were then, for convenience of testing, arranged in ascending and descending order of magnitude, giving the load programmes shown in Fig.3.

3.1 Additional programme tests

In some further tests, given in Table 4, the programme in each series was identical with the programme for the intermediate (i.e. 13g) case, except that the single maximum load was stepped up progressively from 51.7% Static Failing Load, as in series (b), to

(d) 56.7% S.F.L.
(e) 61.7% S.F.L.
and (f) 66.7% S.F.L.

4 METHOD OF TEST

The tests were made in a 100 ton Losenhausen hydraulic fatigue testing machine. The specimen was held in the wedge grips of the machine and carefully centralized.

Repeated loading (Minimum load 1 Tonne) was carried out, using the pulsator for applying the smaller loads and manual operation for the high loads.

The programme testing was done manually throughout because the rate of change of load was too high for automatic operation.

5 TEST RESULTS

Results of the two static tests and endurances under steady fatigue loading are given in Table 2. The mean U.T.S. was 37 tons/in^2. The endurance curve (50% probability) is plotted on Fig.4.

The results of the tests to the load spectrum are given in Table 3. Fig.5 shows a fractured specimen and the appearance of the fractured surfaces. It can be seen that the fatigue area is only a small fraction of the total cross-sectional area.

The results for the modified load spectra are given in Table 4.

6 DISCUSSION OF RESULTS

In Table 3, the reason why the 'total life', column 4, always shows an odd $\frac{1}{2}$ programme is that final rupture always occurred at or near the peak tensile load, i.e. near the middle of a programme.

From Table 3 it will be seen that in series (a) and (b) the arithmetic mean life for three specimens is somewhat greater than the predicted life and in series (c) the mean life is almost exactly equal to the predicted life.

The ratios of mean experimental life/predicted mean life are:-

Series	(a)	1.19
"	(b)	1.23
"	(c)	1.0

The scatter in each case was small. The only case in which the experimental life was below the predicted value was in series (c) specimen 2F - 5E, which reached 83% of the predicted life.

For this form of specimen and loading, therefore, the Miner Cumulative Damage Rule, with an allowance for scatter, appears to be a safe rule for estimating aircraft service life despite the high range of load.

In tests made in Australia[2] on Mustang wings, and in the U.S.A.[3] on outer wing panels of a jet engined fighter, experimental values of $\Sigma n/N$ of the order of 1.5 have been obtained. Doubtless, in a complete wing, redundancy often plays a part in relieving highly stressed members. When it is considered that the tests of Ref.3 were done for a mixture of 60, 80 and 100 per cent limit load, it is fairly certain that more extensive plastic deformation occurred at the maximum load. But the difference from the tests reported here for simple notched specimens is not very significant, since for specimen 2EF - 1D series (f) Table 4, one of three in which the peak load was equal to the limit load, the ratio of experimental life to predicted life was 1.325.

Table 4 shows that, in the range 50 - 67% of static failing load, an occasional high load does negligible damage, and the cumulative damage rule is somewhat conservative when applied to the whole spectrum.

The greatest scatter found was in series (d); yet the lowest life ($15\frac{1}{2}$ programmes) out of four tests is only 30% below the mean and 1/2.5 of the highest. In series (a), (b), (e) and (f) the scatter is quite small.

7 CONCLUSIONS

General conclusions cannot yet be drawn, but within the scope of these tests the mean experimental $\Sigma n/N$ was never less than 1, indicating that the Palmgren - Miner Cumulative Damage Hypothesis is conservative. Tests are required on different forms of specimen, in particular, one typical of bolted spar joints.

LIST OF REFERENCES

Ref No.	Author	Title, etc.
1	Gassner, E.	'Betriebsfestigkeit von Flügelbauteilen' (Fig.17) 'Fatigue in Flight Structures' International Conference held by Columbia University, New York. January, 1956.
2	Payne, A.O.	'Random and Programmed Load Sequence Fatigue Tests on 24 S-T Aluminium Alloy Wings'. Commonwealth of Australia, Dept. of Supply Aer. Res. Lab. Report No. SM 244. 1956.
3	Carl, R.A. Wegeng, T.J.	'Investigations concerning the fatigue of aircraft structures'. Proc. A.S.T.M. Vol. 54 p.903 (1954)

APPENDIX 1

EXTRUDED ALUMINIUM ALLOY BAR TO SPECIFICATION D.T.D. 363A (ZINC-BEARING ALLOY)

Specified chemical composition

Zinc	-	Not less than 4.0 per cent nor more than 8.0 per cent
Copper	-	Not more than 3.0 per cent
Magnesium	-	Not more than 4.0 per cent
Manganese	-	Not more than 1.0 per cent
Chromium	-	Not more than 1.0 per cent
Iron	-	Not more than 0.6 per cent
Silicon	-	Not more than 0.6 per cent
Titanium	-	Not more than 0.3 per cent
Aluminium	-	The remainder

Specified Heat Treatment

Solution treated and artifically aged, i.e. heated to 460°C ±10°C, quenched, and aged by heating at 135° ±10°C for requisite period.

Specified minimum strengths

	Test pieces* representing:-	
	Extruded sections greater than $\frac{3}{8}$" in thickness	Extruded sections not greater than $\frac{3}{8}$" in thickness
0.1 per cent proof stress	Not less than 33 tons per sq.in.	Not less than 30 tons per sq.in.
Ultimate tensile stress	Not less than 38 tons per sq.in.	Not less than 35 tons per sq.in.
Elongation	Not less than 5 per cent	Not less than 5 per cent

* For bars up to and including $1\frac{1}{8}$" dia. or width across flats, the tensile test piece shall be machined concentrically from the test sample; above this diameter or width its longitudinal axis shall be not less than 9/16" from the surface of the test sample.

TABLE 1

Programmes based on 11, 13 and 15g ultimate calculations

NOTE In all cases minimum load is ZERO

Block No.	Series (a) 11g Ultimate		Series (b) 13g Ultimate		Series (c) 15g Ultimate		Cycles	Total Cycles
	Max Load % Ult.	Load Tonne (Metric)	Max Load % Ult.	Load Tonnes	Max Load % Ult.	Load Tonnes		
1	20.0	13.2	16.9	11.1	14.7	9.7	13	13
2	24.6	16.2	20.8	13.7	18.0	11.9	10	23
3	29.2	19.2	24.6	16.2	21.4	14.1	9	32
4	33.6	22.1	28.5	18.8	24.6	16.2	7	39
5	38.2	25.1	32.3	21.2	28.0	18.4	6	45
6	42.8	28.2	36.2	23.8	31.4	20.6	4	49
7	47.4	31.2	40.0	26.3	34.7	22.7	3	52
8	52.0	34.2	43.9	28.9	38.2	25.2	2	54
9	56.6	37.3	47.9	31.5	41.5	27.3	1	55
10	61.2	40.3	51.7	34.0	44.8	29.5	1	56
9	56.6	37.3	47.9	31.5	41.5	27.3	1	57
8	52.0	34.2	43.9	28.9	38.2	25.2	2	59
7	47.4	31.2	40.0	26.3	34.7	22.7	3	62
6	42.8	28.2	36.2	23.8	31.4	20.6	4	66
5	38.2	25.1	32.3	21.2	28.0	18.4	6	72
4	33.6	22.1	28.5	18.8	24.6	16.2	7	79
3	29.2	19.2	24.6	16.2	21.4	14.1	9	88
2	24.6	16.2	20.8	13.7	18.0	11.9	10	98
1	20.0	13.2	16.9	11.1	14.7	9.7	13	111

Tested Ultimate Load = 65,85 metric tonnes = 64.7 tons (see Table 26)
(1 tonne = 2,200 lb.)

TABLE 2

(a) Static tests

Spec. No. 2 EF - 3A. Failed at 65.0 tonnes at bolt holes
Spec. No. 2 EF - 1D. Failed at 66.7 tonnes at bolt holes

Average S.F.L. = 65.85 tonnes

(b) Repeated loading tests

Note: 1 tonne = 1,000 Kg = 2,200 lb.

Load Range % S.F.L.	Load Range Tonnes	Cycles to Failure
1.5 - 10	1 - 6.85	127,870
"	(1300 to 8750 lb/in^2)	89,410
1.5 - 26	1 - 17.1	6,150
"		5,050
"	(1300 to 21,900 lb/in^2)	4,520
1.5 - 36	1 - 23.8	1,610
"	(1300 to 30,400 lb/in^2)	1,210
"	"	1,400
1.5 - 46	1 - 30.3	540
"	(1300 to 38,800 lb/in^2)	440
1.5 - 60	1 - 39.5	125
"	(1300 to 50,500 lb/in^2)	106
0 - 73	1 - 50.0	48
"	(1300 to 64,000 lb/in^2)	52

TABLE 3

Programmes based on 11, 13 and 15g ultimate calculations

U.T.S. = 65.85 tonnes

Series	Spec. No.	Ult. Design Case	Peak Load % of Ult.	Total Life Programmes	Mean Programme Life	Predicted Life
(a)	2 EF - 10I	11g	61.2	14½	12½	10½
	2 EF - 6D	11g	"	11½		
	2 EF - 16H	11g	"	11½		
(b)	2 EF - 10E	13g	51.7	23½	27	22
	2 EF - 5E	13g	"	25½		
	2 EF - 11A	13g	"	32½		
(c)	2 EF - 2B	15g	44.8	42½	40.8	40.5
	2 EF - 5E	15g	"	33½		
	2 EF - 14C	15g	"	46½		

TABLE 4

Programme as for 13g ultimate calculations with the exception of the peak load details of which are given in the table of results

Series	Spec. No.	Load Range		Life in Programmes
		% U.T.S.	Tonnes	
(d)	2 EF - 14F	0 - 56.7	0 - 37.3	22½
	2 EF - 14A	"	"	15½
	2 EF - 6F	"	"	38½
	2 EF - 11E	"	"	13½
			Arithmetic Mean 22.5	
(e)	2 EF - 2A	0 - 61.7	0 - 40.6	23½
	2 EF - 8A	"	"	24½
	2 EF - 2E	"	"	23½
			Arithmetic Mean 23.8	
(f)	2 EF - 8G	0 - 66.7	0 - 43.9	18½
	2 EF - 1D	"	"	24½
	2 EF - 1B	"	"	23½
			Arithmetic Mean 22.2	

Comparison between predicted and experimental lives

	(1) Predicted Life (Programmes)	(2) Ar. Mean Life (Programmes)	Ratio (2)/(1)
Series (b)	22.0	27.0	1.23
" (d)	20.5	22.5	1.095
" (e)	19.5	23.8	1.22
" (f)	18.5	22.2	1.20

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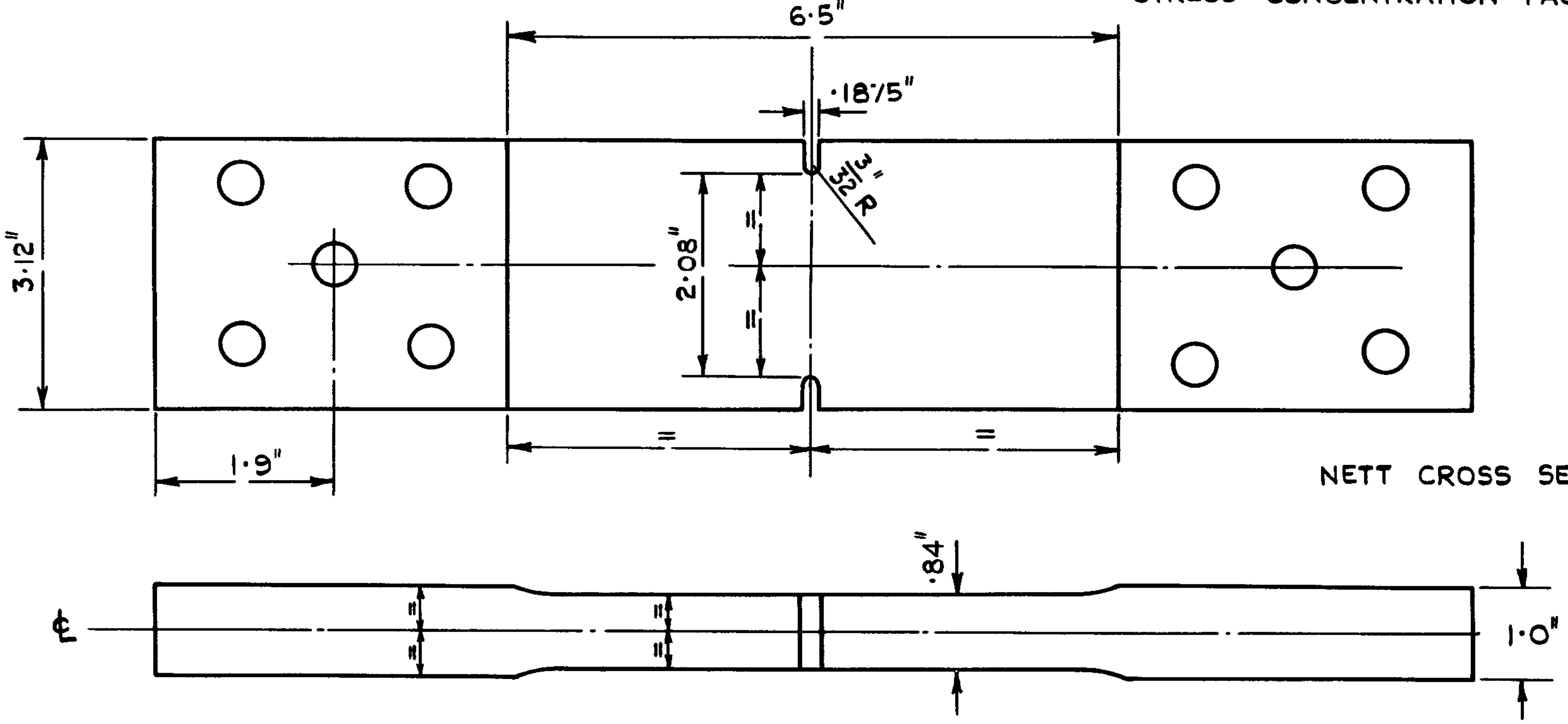
STRESS CONCENTRATION FACTOR $K_\tau = 3{\cdot}65$.

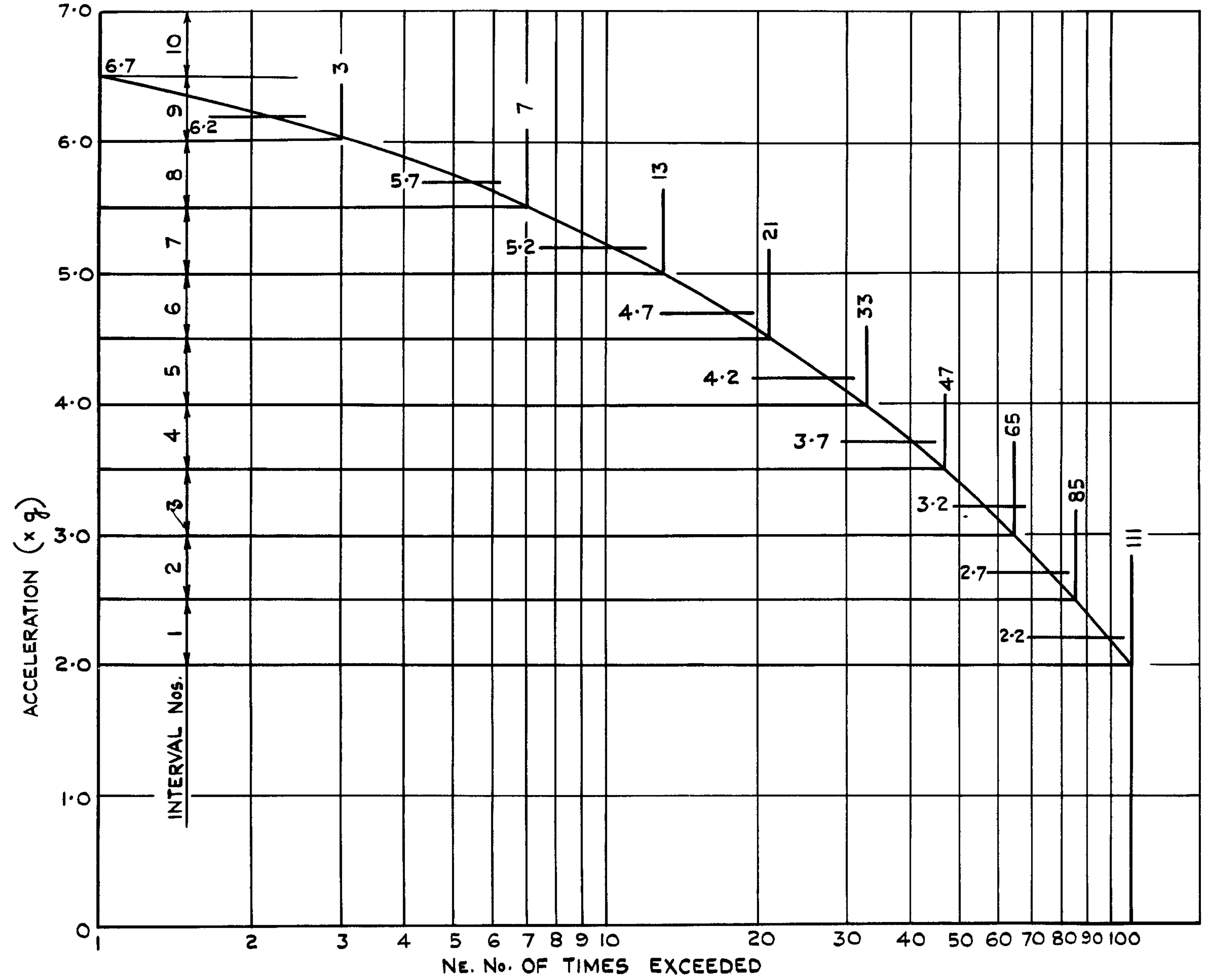


NETT CROSS SECTIONAL AREA THROUGH NOTCH = $1{\cdot}745\ \text{IN}^2$

SCALE $^1/_2$

MATERIAL : EXTRUDED BAR TO D.T.D. 363A

FIG. 1. TEST SPECIMEN.



CYCLES FOR EACH RANGE OBTAINED BY DIFFERENCES AS SHEWN IN TABLE

BASIC TEST PROGRAMME

LOAD No.	INTERVAL RANGE(g)	EQUIVALENT LOAD (g)	NE	DIFFERENCE
1	2 - 2·5	2·2	111	26
2	2·5 - 3	2·7	85	20
3	3 - 3·5	3·2	65	18
4	3·5 - 4	3·7	47	14
5	4 - 4·5	4·2	33	12
6	4·5 - 5	4·7	21	8
7	5 - 5·5	5·2	13	6
8	5·5 - 6	5·7	7	4
9	6 - 6·5	6·2	3	2
10	6·5 - 7	6·7	1	1
			0	111

FIG. 2. BASIC LOAD SPECTRUM.

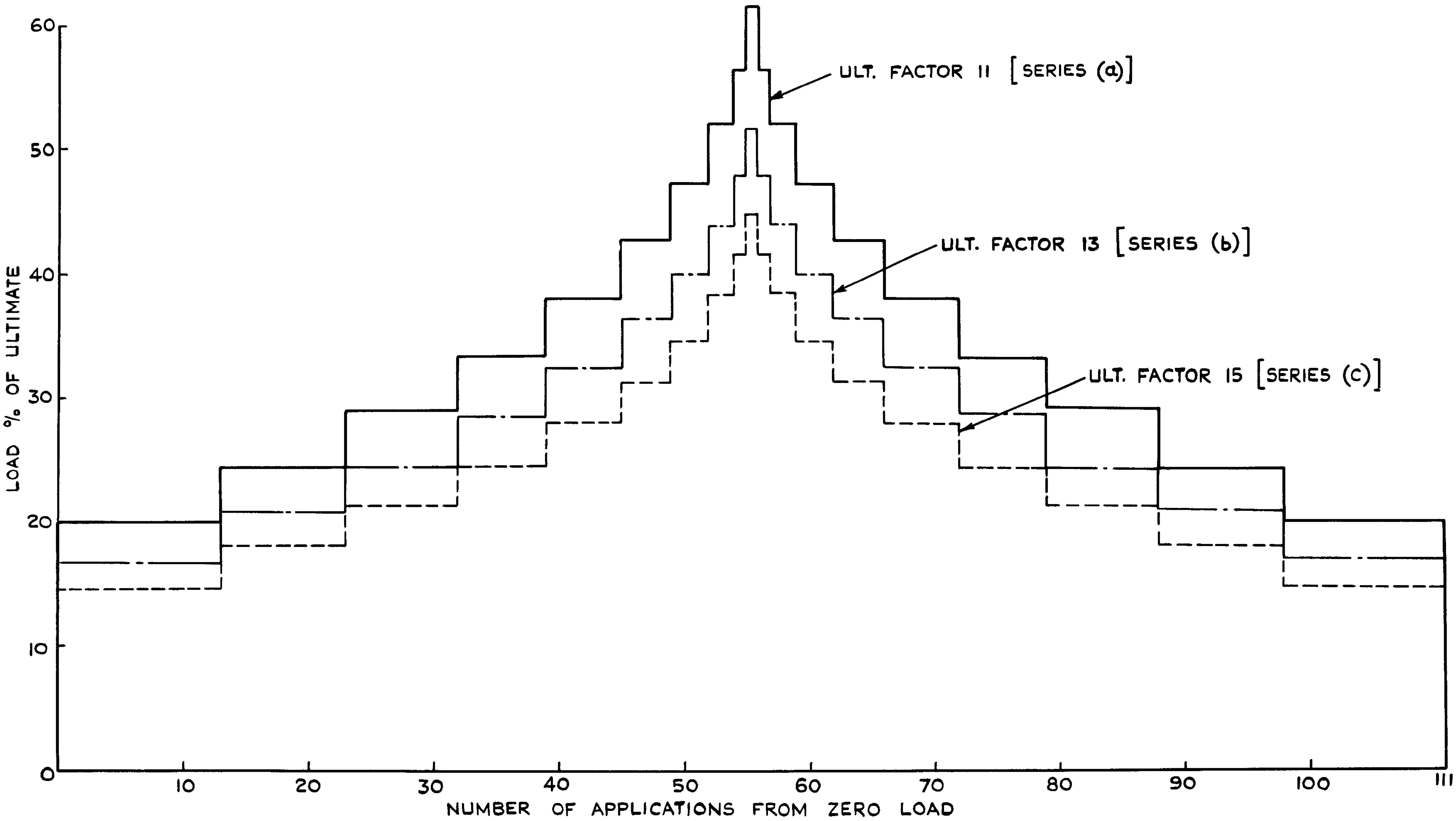


FIG. 3. LOADING PROGRAMME.

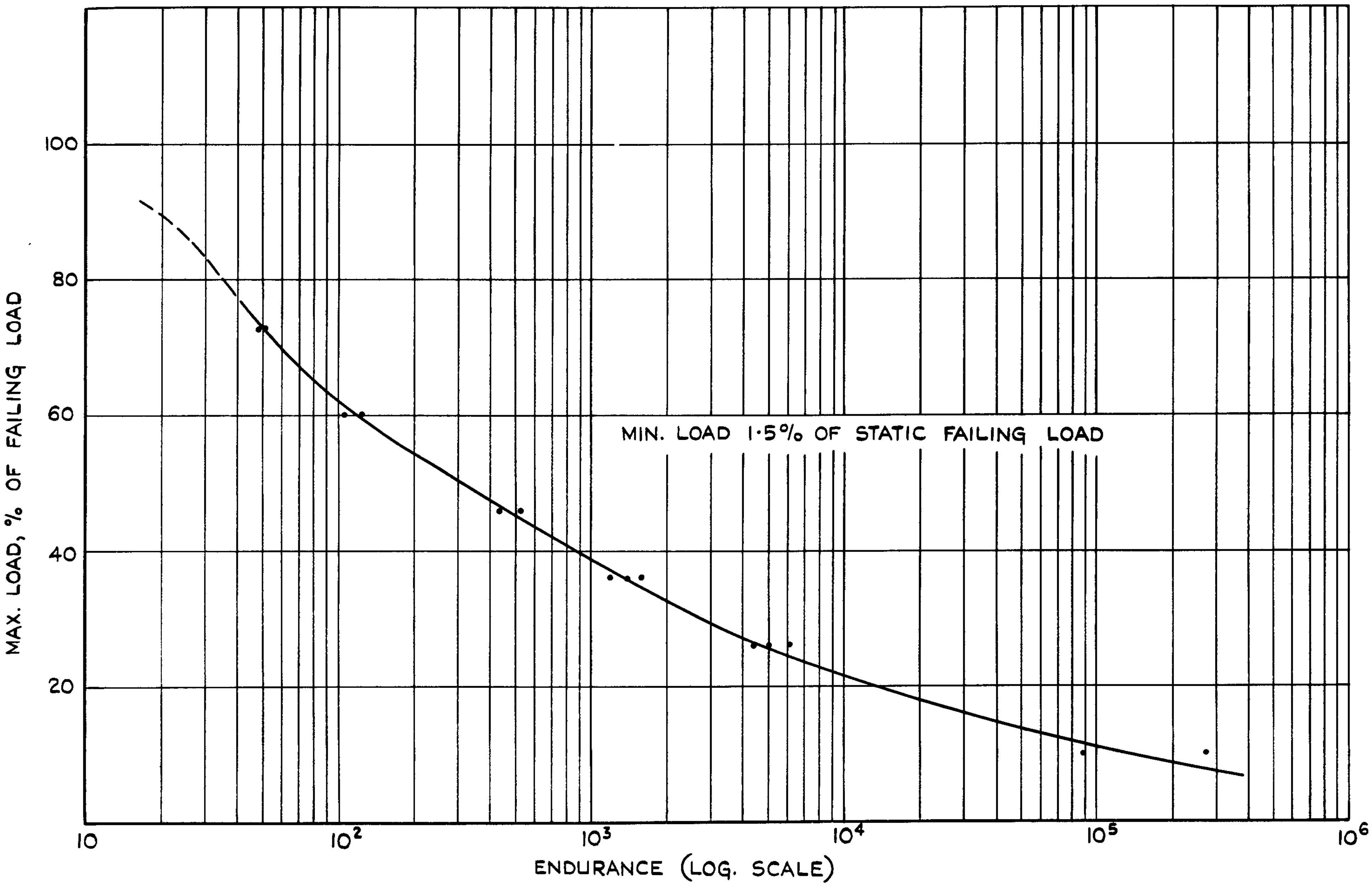
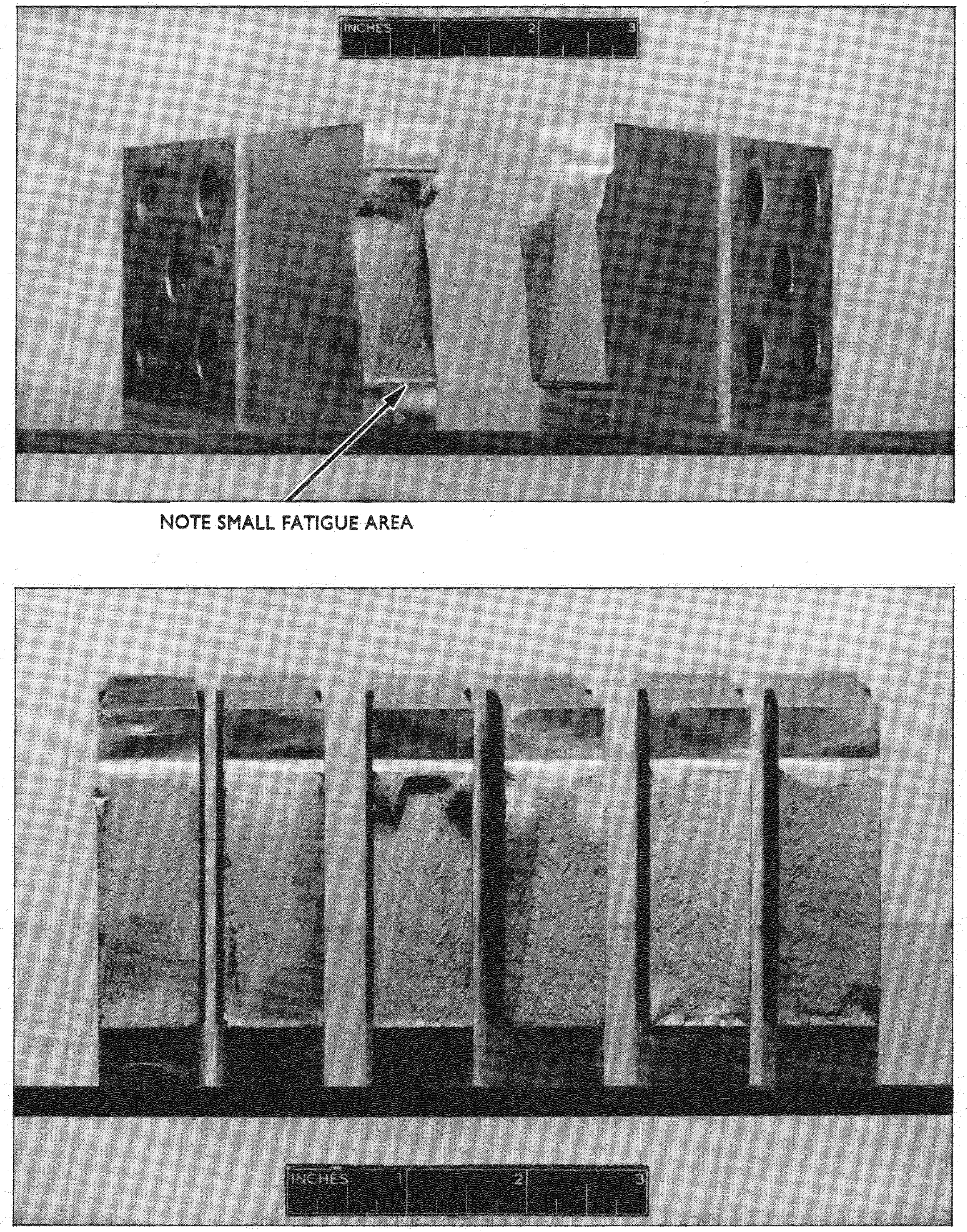


FIG. 4. ENDURANCE CURVE FOR NOTCHED SPECIMENS.



NOTE SMALL FATIGUE AREA

FIG.5. FRACTURED SURFACES

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