NASA Technical Memorandum 80114

NASA-TM-80114 19790018256

EFFECTS OF TRUNCATION OF A PREDOMINANTLY COMPRESSION LOAD SPECTRUM ON THE LIFE OF A NOTCHED GRAPHITE/EPOXY LAMINATE

Edward P. Phillips

June 1979

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

1 Report No	2 Government Accession No	3 Recipient's Catalog No
NASA TM-80114		

4 Title and Subtitle	5 Report Date
EFFECTS OF TRUNCATION OF A PREDOMINANTLY COMPRESSION LOAD SPECTRUM ON THE LIFE OF A NOTCHED GRAPHITE/EPOXY LAMINATE	June 1979
	6 Performing Organization Code

7 Author(s)	8 Performing Organization Report No
Edward P. Phillips	

9 Performing Organization Name and Address	10 Work Unit No
NASA Langley Research Center Hampton, VA 23665	506-17-23-03
	11 Contract or Grant No

12 Sponsoring Agency Name and Address	13 Type of Report and Period Covered
National Aeronautics and Space Administration Washington, DC 20546	Technical Memorandum
	14 Sponsoring Agency Code

15 Supplementary Notes

Presented at the ASTM Symposium on the Fatigue of Fibrous Composite Materials, San Francisco, California, May 22-23, 1979.

16 Abstract

The fatigue behavior of a notched, graphite/epoxy (T300/5208) laminate subjected to predominantly compressive loading was explored in a series of constant-amplitude and transport wing spectrum tests. Results of these tests indicated that (1) the amount of local (near the notch) buckling allowed in the tests significantly affected fatigue life, (2) spectrum truncation of either the high- or low-load end of the spectrum produced lives greater than those obtained in the baseline, complete-spectrum test, but life was much more sensitive to truncations at the high-load end, and (3) the Palmgren-Miner linear cumulative damage theory always predicted lives much longer than the actual spectrum loading test lives.

17 Key Words (Suggested by Author(s))		18 Distribution Statement
Composite materials	Fatigue (materials)	Unclassified - Unlimited
Constant life fatigue diagrams	Graphite composites	
Variable-amplitude loading	Compression tests	
	Spectrum loading tests	Subject Category 39

19 Security Classif (of this report)	20 Security Classif (of this page)	21 No of Pages	22 Price*
Unclassified	Unclassified	26	$4.50

* For sale by the National Technical Information Service, Springfield Virginia 22161

EFFECTS OF TRUNCATION OF A PREDOMINANTLY COMPRESSION LOAD SPECTRUM ON THE LIFE OF A NOTCHED GRAPHITE/EPOXY LAMINATE

Edward P. Phillips
NASA Langley Research Center
Hampton, Virginia 23665

ABSTRACT

The fatigue behavior of a notched, graphite/epoxy (T300/5208) laminate subjected to predominantly compressive loading was explored in a series of constant-amplitude and transport wing spectrum tests. Results of these tests indicated that (1) the amount of local (near the notch) buckling allowed in the tests significantly affected fatigue life, (2) spectrum truncation of either the high- or low-load end of the spectrum produced lives greater than those obtained in the baseline, complete-spectrum test, but life was much more sensitive to truncations at the high-load end, and (3) the Palmgren-Miner linear cumulative damage theory always predicted lives much longer than the actual spectrum loading test lives.

KEY WORDS composite materials, fatigue (materials), compression tests, graphite composites, epoxy laminates, constant life fatigue diagrams, variable amplitude loading tests, spectrum loading tests

INTRODUCTION

In most airframe development programs, the results of structural fatigue tests are used to decide if the structure satisfies the fatigue

N79-26427#

life design requirement. Years of experience with tests on aluminum coupons and structures have led to satisfactory procedures for defining suitable test load spectra, that is, spectra which yield a representative life estimate in a reasonably short test time. In particular, a general understanding of the effects of truncating load spectra at the high- and low-load levels has evolved. For composite materials, however, a similar base of test experience has not yet been accumulated.

The current work was undertaken to explore (1) the effect of spectrum truncations on the fatigue life of notched, quasi-isotropic graphite/epoxy coupons and (2) the capability of the Palmgren-Miner linear cumulative damage theory to predict the truncation effects. Since cyclic compressive loading is generally more detrimental to composites than cyclic tension [1-3], a load spectrum representative of the upper surface of a transport wing was used in the current work. Constant-amplitude tests were conducted to explore the sensitivity of fatigue life to the amount of local (near the notch) buckling allowed in the tests and to provide data for cumulative damage calculations.

EXPERIMENTAL PROCEDURE

Materials and Specimens

Specimens were cut from 16-ply, $[45/0/-45/90]_{2s}$ (quasi-isotropic) sheets made from T300/5208 graphite/epoxy unidirectional tape. The cured laminate had an average ply thickness of 0.14 mm, a fiber volume of

64 percent, and a void content of 0.36 percent. Specimens were stored and tested in an ambient laboratory air environment. Measurements taken on several specimens during the test program indicated a 0.6 to 0.7 percent moisture content by weight.

The test specimen configuration is shown in figure 1. The central 6.35-mm hole was made by a diamond-coated, ultrasonically vibrating drill. Generally, this drilling procedure produced very clean hole surfaces, but some interlaminar cracking was detected between the first two plies on the drill exit surface of most of the specimens. Nondestructive examinations showed the depth of the delaminations did not exceed 0.3 mm, and subsequent test observations indicated the delaminations did not play a significant role in the failure process.

Test Machines

Specimens were tested to complete rupture in axial-load, closed-loop, servohydraulic testing machines having about 45 kN force capacity. Constant-amplitude and variable-amplitude tests were run at a nearly uniform loading rate so that loading frequency ranged from about 3 to 20 Hz depending upon load amplitude. Load sequences were generated by a small, on-line digital computer.

Load Spectra

The standardized test TWIST (Transport WIng Standard Test) [4] was used in the spectrum loading tests. TWIST was developed jointly by the

Laboratorium für Betriebsfestigkeit in Germany and the National Lucht-en Ruimtevaartlaboratorium in The Netherlands. A list of the most significant features of this test is given in table 1 and a tabulation of the load spectrum is given in table 2. In addition to the baseline TWIST spectrum, several truncated versions were used. The truncated versions were created by deleting the following load levels: (1) the lowest, (2) the two lowest, (3) the four lowest, (4) the two highest, and (5) the four highest and the lowest. In the truncation process, flights were never completely eliminated even though all of the flight loads included in the baseline version of the flight were scheduled to be deleted (see tables 1 and 2). For such flights, the truncated version of the flight consisted of a ground-air-ground cycle which was bounded by the ground load and the highest flight load in the baseline version of the flight (see fig. 2). The sequence of flight types remained the same as the baseline TWIST sequence in all tests, but the sequence of loads within flights changed as a result of each spectrum truncation.

Specimen Antibuckling Procedure

Specimens were restrained from column buckling during compressive loading by sandwiching the specimen between two aluminum plates (coated with a trifluoroethylene resin plastic). This antibuckling procedure was chosen because it was the simplest of the methods described in the literature [1-3,5,6] and because no method had a clear advantage. In the instances where the "plate-sandwich" antibuckling procedure was used in studies reported in the literature [1,3,5], the degree to which local (near the notch) buckling was allowed varied over a wide range. That is, either the plates covered the entire specimen surface or some portion of the plates was cut away to free the test section. Because the size of the windows (cutouts) in the antibuckling plates could reasonably be expected to influence fatigue life, preliminary tests were conducted to explore the magnitude of the effect.

RESULTS AND DISCUSSION

Specimen Buckling Restraint Tests

The four antibuckling plate configurations used in the tests and corresponding median test lives for one stress level are shown in figure 3. Test results are tabulated in table 3. As expected, the more specimen surface covered by the antibuckling plates, the longer the fatigue life. The life for tests in which the antibuckling plates had no window was about 30 times that for tests in which the windows were 32-mm wide and either 32- or 64-mm long. For higher compressive loading, the difference in lives was even more pronounced. The results in figure 3 illustrate the need to consider the influence of the antibuckling procedures in making comparisons among compressive-loading fatigue data from the literature.

The antibuckling plates having a 32- by 32-mm window were selected for all subsequent tests in the current work. This selection was based on the feeling that local buckling associated with delamination around the hole should not be restrained, but that the specimen should be restrained from general-section buckling so that testing can cover the full material strength range. In the current tests, the delamination zone exceeded the 19-mm window size in many tests but rarely exceeded the 32-mm size. The choice of any antibuckling procedure is, of course, rather arbitrary in the sense that none of the procedures will realistically simulate buckling conditions in all structural configurations.

Constant-Amplitude Tests

Constant-amplitude tests were conducted at several R values $\left(R = \frac{\text{minimum stress}}{\text{maximum stress}}\right)$ to provide data for the cumulative damage calculations. Results of the tests are tabulated in table 4 and shown in figure 4 in the form of a constant life diagram. Results of tension and compression static strength tests of this specimen configuration are also plotted in figure 4. As expected, the constant life diagram shows that compressive mean stresses had a deleterious effect on fatigue life. All of the constant life lines proceed down and to the left from the R = -1 line (that is, in the compressive mean stress half of the diagram). This trend in the constant life lines means that for a constant alternating stress, the higher the compressive mean stress, the lower the life. By contrast, the opposite trend is evident over at least a portion of the tensile mean stress half of the diagram; that is, the higher the tensile mean stress, the higher the life. In general, the trends in the data show that the laminate tested in the current program is more susceptible to compressive fatigue loading than to tensile fatigue loading.

The failure process in all compression-dominated tests was progressive delamination followed by failure in a crippling mode. In most tests, the surface plies split and buckled near the hole at about 10 to 20 percent of the life. The shadow Moiré [7] photograph in figure 5 illustrates this failure mode. The area of out-of-plane displacement is outlined by the closely spaced fringes near the hole. The more widely spaced fringes away from the hole are "initial condition" fringes and do not represent out-of-plane displacement of the specimen. During each test, delaminations initiated and grew at all of the ply interfaces at the hole. The extent of the delaminated zone near the end of a test (at 93 percent of the life) is shown in the dye-enhanced x-ray radiograph [8] in figure 6. At equal lives, the extent of delamination at rupture (especially of the surface ply) was greater for tests with the greater tensile stress in the loading cycle. This is illustrated in the photograph of failed specimens in figure 7.

Spectrum Loading Tests

The flight mean stress (spectrum reference level, see Table 2) in the current tests was -111 MPa (based on the gross section). This stress level was chosen because it produced test lives representative of transport aircraft design goals (about 60,000 flights of the baseline spectrum). A few preliminary tests at a flight mean stress of -95 MPa had produced lives of about 10^6 flights. At the -111 MPa flight mean stress, the maximum stress in the baseline spectrum was -289 MPa, or 89 percent of the median compressive static strength. Corresponding gross-section strain levels for the flight mean and maximum stresses in the spectrum were 0.002100 and 0.005450, respectively. The appearance of the failures and the extent of delamination at failure in the spectrum loading tests were similar to those for predominantly compression constant-amplitude tests at the same maximum compressive load.

The results of the tests to explore truncation effects are shown in figure 8 and tabulated in table 5. A comparison of the median lives (in flights) indicates that truncation of the spectrum at either the high- or low-load end produced lives greater than those obtained in the baseline spectrum test. However, life was much more sensitive to truncations at the high-load end. Omission of the four highest load levels plus the lowest level produced lives eleven times the baseline spectrum life. With these omissions, the effect on life appears to be largely due to omission of the high loads (which constitute less than 0.007 percent of the total number of load cycles in the spectrum) since omission of the lowest load alone did not produce lives longer than the baseline spectrum. Indeed, omission of the four lowest load levels produced lives only two times the baseline spectrum life even though the omitted load cycles constitute more than 99 percent of the loads in the baseline spectrum.

In defining the test spectrum for composite wing structures, the data in figure 8 suggest that the high-load end of the spectrum should not be truncated to the same extent as has been the practice for metallic structures. On the other hand, truncation of the low-load end of the spectrum shows promise for achieving large reductions in test time without significantly changing the test result.

Linear Cumulative Damage Evaluation

To explore the usefulness of the Palmgren-Miner linear cumulative damage theory for composite applications, the median lives from the spectrum loading tests were compared to the corresponding lives calculated from the theory. The cumulative damage calculations were based on the constant-amplitude data represented in figure 4 and on the "rainflow" method [9] of defining the random load history in terms of constant-amplitude cycles.

Predicted and measured lives are plotted in figure 9 for comparison. As can be seen, the linear damage theory always predicted longer lives than the actual test lives, that is, predictions were always on the unsafe side. The ratios of predicted to test life ranged from 6.6 to 19.8. Rosenfeld and Huang [1] and Schutz and Gerharz [3] have reported the same trend for tests on other graphite/epoxy laminates using fighter-wing load spectra.

The results in figure 9 also show that the linear damage theory failed to predict the trend toward longer life for progressively larger truncations of the spectrum at the low-load end. For truncations at the high-load end of the spectrum, however, the theory did predict the correct trend in life.

CONCLUDING REMARKS

The fatigue behavior of a notched, graphite/epoxy (T300/5208), $[45/0/-45/90]_{2S}$ laminate subjected to predominantly compressive loading was explored in a series of constant-amplitude and transport wing spectrum tests. These exploratory tests support the following conclusions and observations:

1. The amount of local (near the notch) buckling allowed in the tests significantly affected fatigue life. This result indicates that the influence of antibuckling procedures used in a test must be accounted for to make meaningful comparisons among compressive-loading fatigue data from the literature.

2. Spectrum truncation at either the high- or low-load end of the spectrum produced lives greater than those obtained in the baseline, complete-spectrum test. However, life was much more sensitive to truncations at the high-load end. The results suggest that in defining test spectra for composite wing structures, the high-load end of the spectrum should not be truncated to the same extent as has been the practice for metallic structures. Also, truncation of the low-load end of the spectrum shows promise for achieving large reductions in test time without significantly changing the test result.

3. The Palmgren-Miner linear cumulative damage theory always predicted lives much longer than the actual spectrum loading test lives; that is, predictions were always on the unsafe side. However, the theory predicted the correct trends for effects of truncating the high-load end of the spectrum.

REFERENCES

[1] Rosenfeld, M. S.; and Huang, S. L.: Fatigue Characteristics of Graphite/Epoxy Laminates Under Compression Loading. Technical Papers, Volume A, American Institute for Aeronautics and Astronautics, Inc., New York, 1977, pp. 423-427.

[2] Ramani, S. V.; and William, D. P.: Notched and Unnotched Fatigue Behavior of Angle-Ply Graphite/Epoxy Composites. Fatigue of Filamentary Composite Materials, ASTM STP-636, American Society for Testing and Materials, 1977, pp. 27-46.

[3] Schutz, D.; and Gerharz, J. J.. Fatigue Strength of a Fibre-Reinforced Material. Composites, vol. 8, no. 4, October 1977, pp. 245-250.

[4] de Jonge, J. B.; et al.: A Standardized Load Sequence for Flight Simulation Tests on Transport Aircraft Wing Structures. NLR-TR-73029U, National Lucht-en Ruimtevaartlaboratorium, Amsterdam, The Netherlands (also LBF-Bericht-FM-106, Laboratorium fur Betriebsfestigkeit, Darmstadt, German Federal Republic), 1973.

[5] Walter, R. W.; et al.: Designing for Integrity in Long Life Composite Aircraft Structures. Fatigue of Filamentary Composite Materials, ASTM STP-636, American Society for Testing and Materials, 1977, pp. 228-247.

[6] Ryder, J. T.; and Walker, E. K.. Effect of Compression on Fatigue Properties of a Quasi-Isotropic Graphite/Epoxy Composite. Fatigue of Filamentary Composite Materials, ASTM STP-636, American Society for Testing and Materials, 1977, pp. 3-26.

[7] Theocaris, P. S. Moire Fringes of Isopachics. J. of Sc. Instruments, vol. 41, no. 3, 1964, pp. 133-138.

[8] Chang, F. H.; et al.: Application of a Special X-Ray Nondestructive Testing Technique for Monitoring Damage Zone Growth in Composite Laminates. Composite Reliability, ASTM STP-580, American Society for Testing and Materials, 1975, pp. 176-190.

[9] Endo, T., et al.. Damage Evaluation of Metals for Random or Varying Loading - Three Aspects of Rain Flow Method. Proceedings of the 1974 Symposium on Mechanical Behavior of Materials, Volume I, The Society of Materials Science, Kyoto, Japan, August 21-24, 1974, pp. 371-380.

TABLE 1 - Description of TWIST (Transport WIng Standard Test).

Significant Features

- Flight-by-flight loading.
- Constant flight mean load.
- All ground loads represented by a single load event equal to minus one-half the flight mean load.
- Each flight load reversal occurred at 1 of 20 discrete levels; 10 levels above the flight mean and 10 levels below the flight mean (see Table 2).
- Ten flight types (severities)--each characterized by the number of load levels involved and the number of occurrences of each level.
- Sequence of flight types and sequence of loads within each flight determined by random draw without replacement.
- Random draw of flight loads restricted so that successive loads must generate a mean-level crossing, but no other restrictions on magnitudes of successive loads, that is, the sequence is generated in the random half-cycle fashion.
- Load sequence repeats after 4 000 flights, that is, the block length is 4 000 flights.

TABLE 2 - Frequency of occurrence of flight types and load cycles within each flight in TWIST.

Flight type	Frequency of occurrence of each flight type in one block of 4 000 flights	Frequency of occurrence of flight load cycles[a] at the ten alternating load levels										Total number of cycles per flight
		1.60[b]	1.50	1.30	1.15	.995	0.84	0.685	0.53	0.375	0.222	
A	1	1	1	1	4	8	18	64	112	391	900	1 500
B	1		1	1	2	5	11	39	76	366	899	1 400
C	3			1	1	2	7	22	61	277	879	1 250
D	9				1	1	2	14	44	208	680	950
E	24					1	1	6	24	165	603	800
F	60						1	3	19	115	512	650
G	181							1	7	70	412	490
H	420								1	16	233	250
I	1 090									1	69	70
J	2 211										25	25
Total number of load cycles per block of 4 000 flights		1	2	5	18	52	152	800	4 170	34 800	358 665	
Cumulative number of load cycles per block of 4 000 flights		1	3	8	26	78	230	1 030	5 200	40 000	398 665	

[a]In this table, the frequency of occurrence numbers indicate full cycles.

[b]Ratio of alternating load to the flight mean load.

TABLE 3 - Results of tests[a] to explore the effect of local buckling restraint on fatigue life.

Dimensions of windows in antibuckling plates, mm	Fatigue life, cycles
No window	194 000 1 112 149 2 861 708 5 881 792
19 by 19	102 517 136 372 349 650
32 by 32	61 914 66 029 132 274
32 by 64	45 065 55 742 112 071

[a]Constant amplitude loading, R = -2, minimum gross-section stress = -207 MPa

TABLE 4 - Results of constant amplitude fatigue tests[a].

R (Min. stress / Max. stress)	Minimum gross-section stress, MPa	Fatigue life, cycles
+3	-289	1 600 3 334
	-248	737 968 1 131 736
	-207	>10 000 000 >10 000 000
∞	-289	1 298 1 844
	-248	15 924 78 132
	-207	150 098 527 518 1 076 384
	-165	>10 000 000 >10 000 000
-5	-289	392 1 070
	-248	33 976 43 417
	-207	395 282 546 434
-2	-289	318 350 370
	-248	1 939 2 710 8 141
	-207	61 914 66 029 132 274
	-165	3 133 789 3 917 639 4 925 000
-1	-238	444
	-207	22 263 25 550
	-165	247 012 317 110
	-134	>10 000 000

TABLE 4 - Concluded

R $\left(\frac{\text{Min. stress}}{\text{Max. stress}}\right)$	Minimum gross-section stress, MPa	Fatigue life, cycles
-0.5	-119	123 618
	-103	1 433 070 3 876 274 3 954 528

[a]Window in antibuckling plate was 32 by 32 mm.

TABLE 5 - Results of the spectrum loading fatigue tests[a] using the baseline and truncated spectra.

Spectrum truncations	Fatigue life, flights
None (Baseline spectrum)	61 655 61 655 65 655 165 655
Lowest load level omitted	34 855 53 655 57 655 157 665
Two lowest load levels omitted	82 855 86 855 118 855 137 655
Four lowest load levels omitted	34 855 134 855 137 655 138 855
Two highest load levels omitted	49 652 130 855 267 982 311 359
Four highest and the lowest load levels omitted	508 105 710 935 711 276 1 107 180

[a]Flight mean gross-section stress = -111 MPa, window in antibuckling plate was 32 by 32 mm.

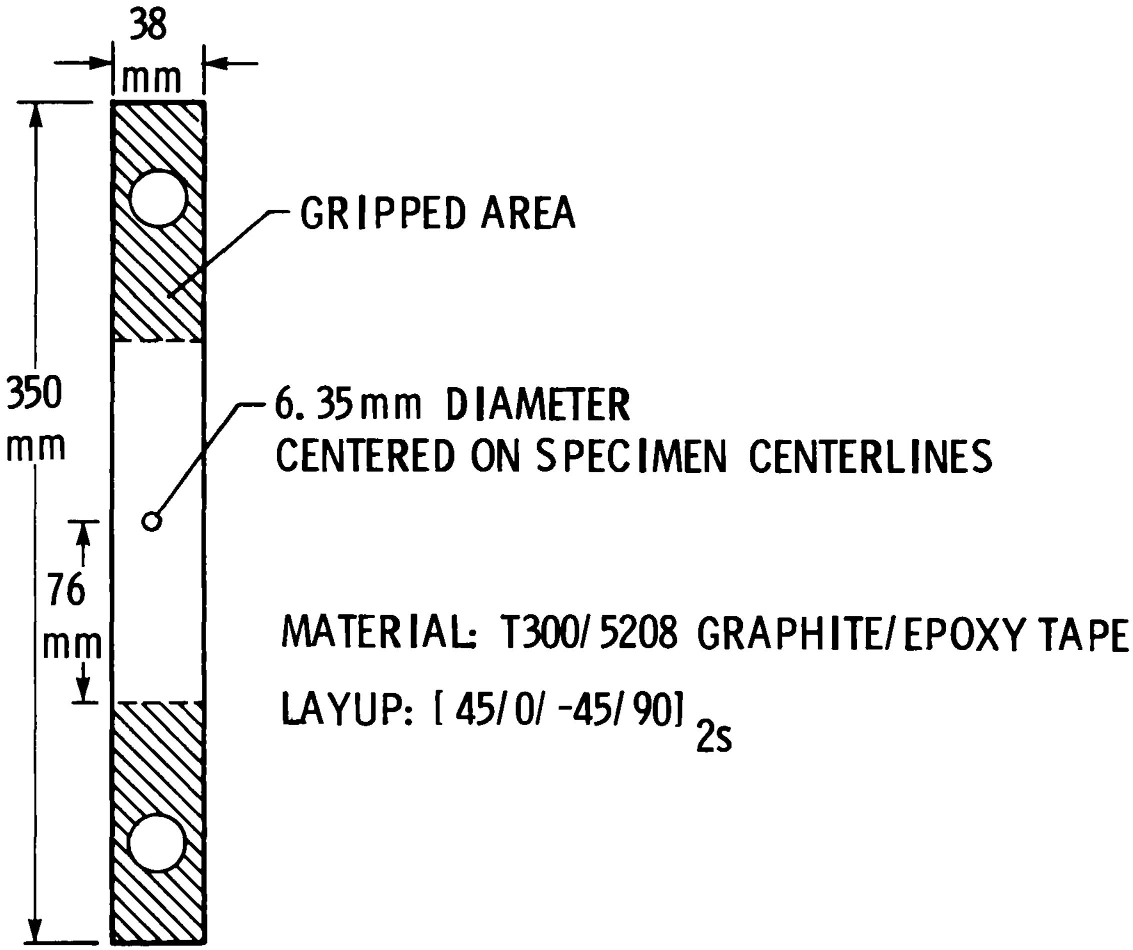

Figure 1.- Specimen configuration.

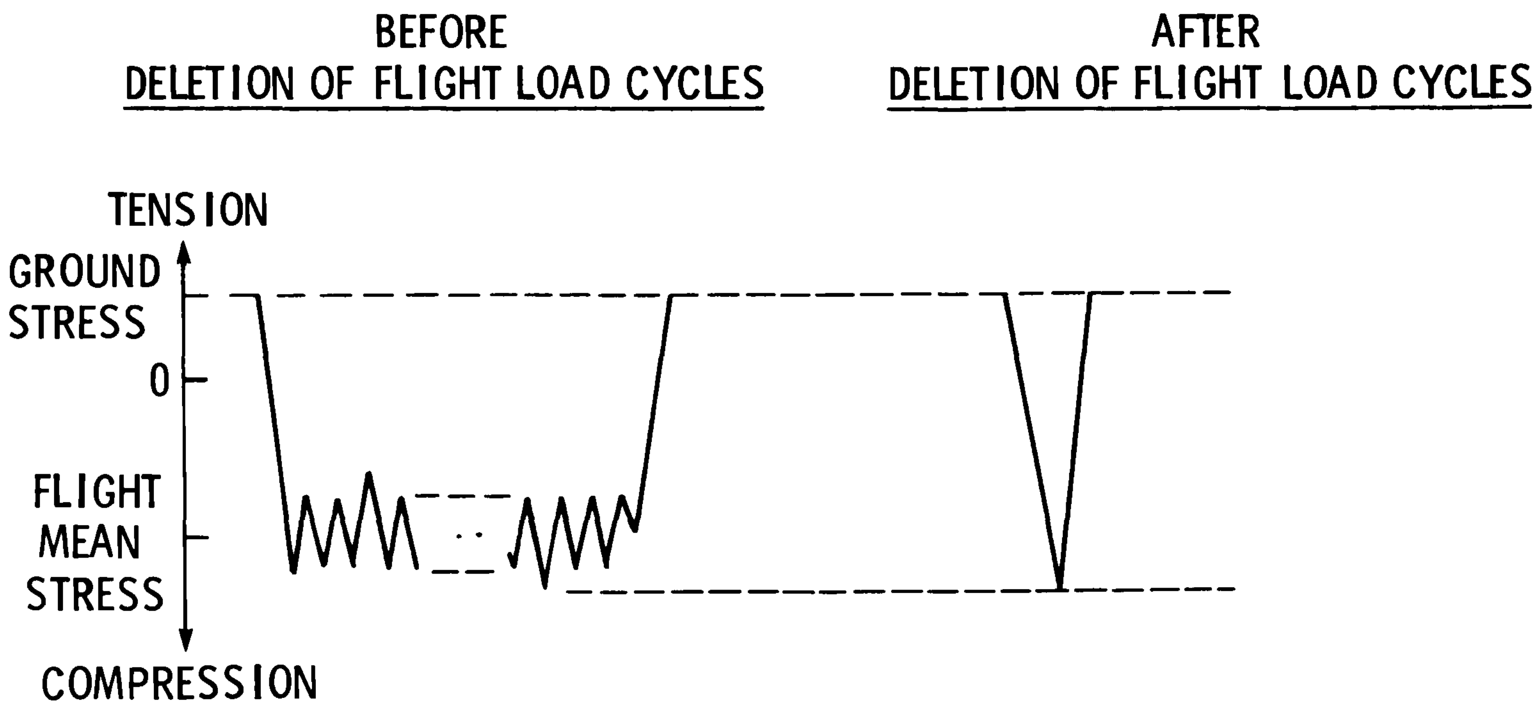

Figure 2.- Example of truncation procedure for flights in which all flight load cycles were scheduled for deletion.

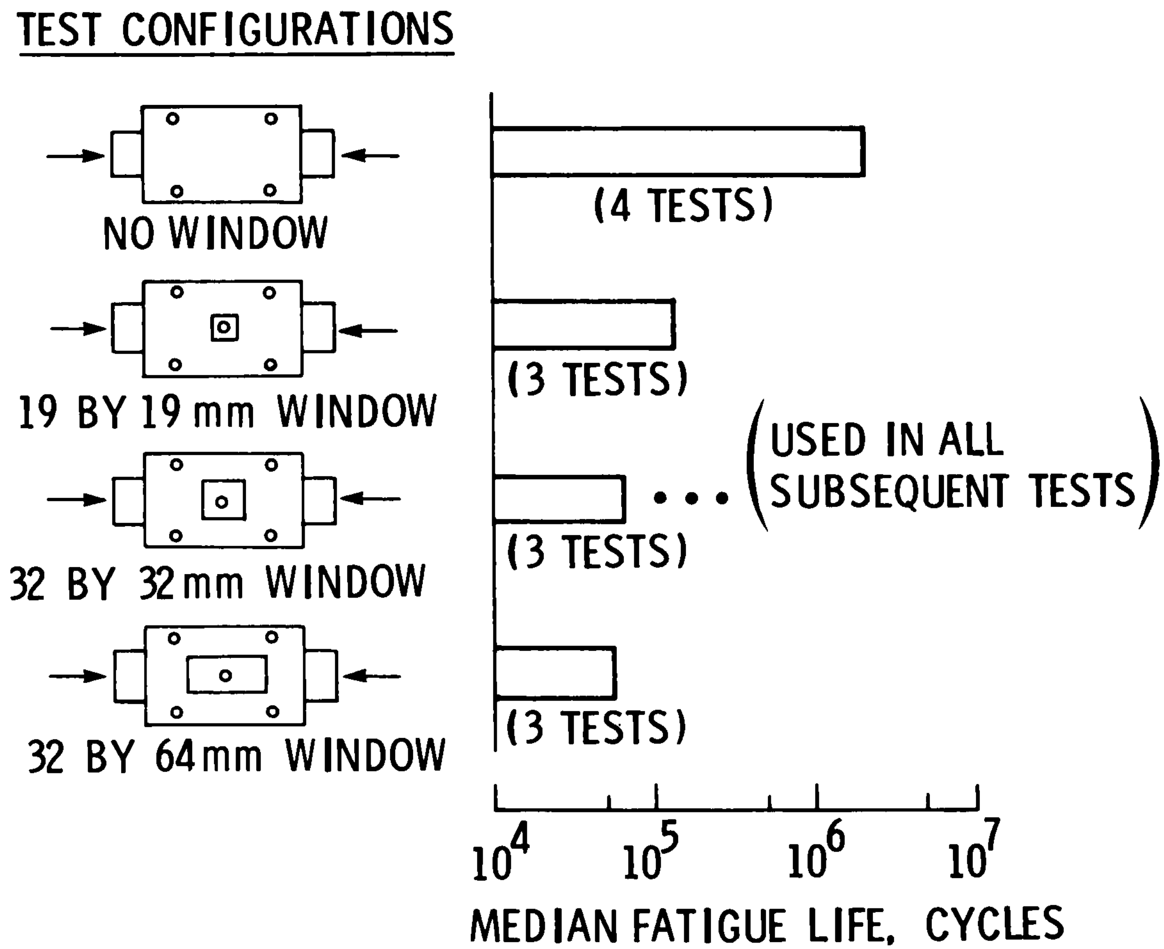

Figure 3.- Effect of antibuckling-plate window size on fatigue life. (Constant amplitude, R = -2, minimum gross-section stress = -207 MPa.)

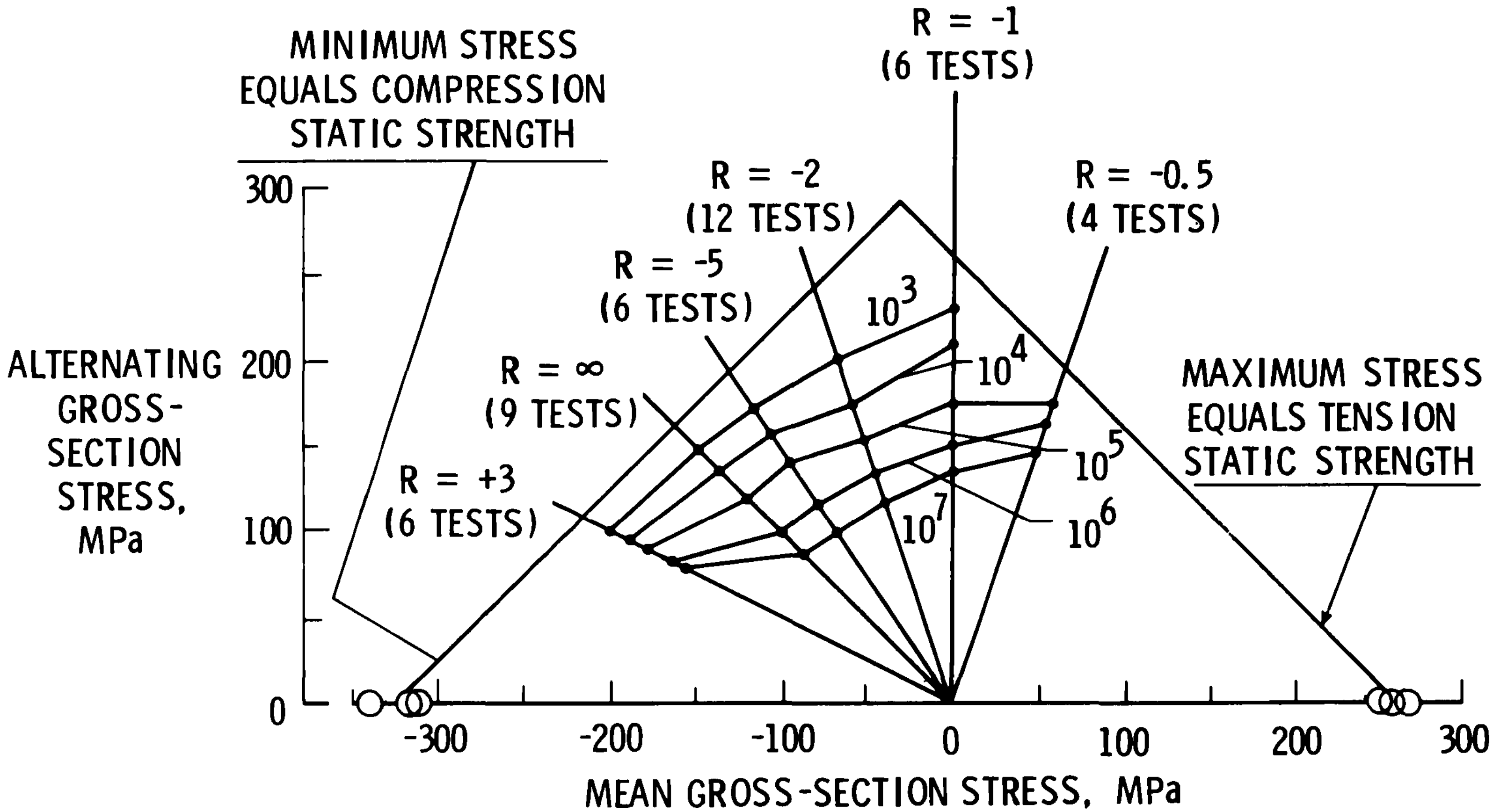

Figure 4.- Constant-life diagram constructed from the results of the constant amplitude tests.

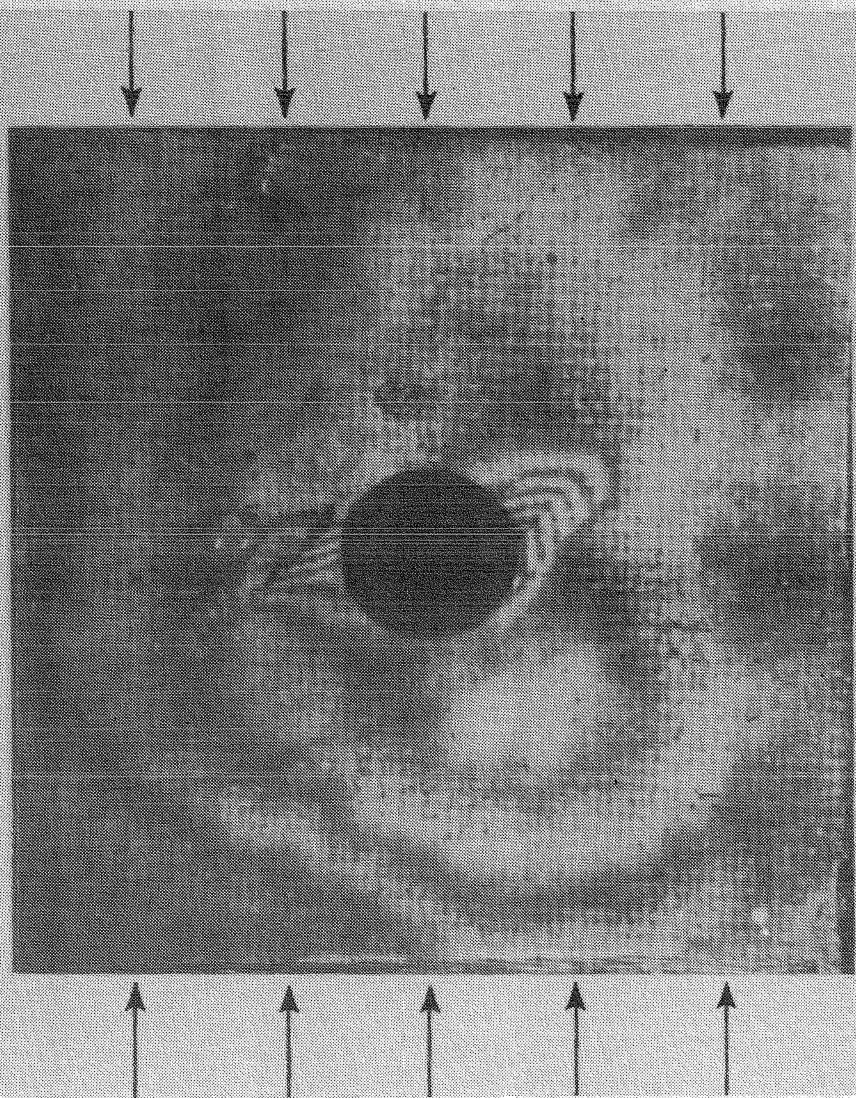

Figure 5.- Typical shadow-Moire photograph showing the fringe pattern caused by out-of-plane displacement of the surface ply near the hole. (Photograph shows the area outlined by the 32- by 32-mm window in the antibuckling plate.)

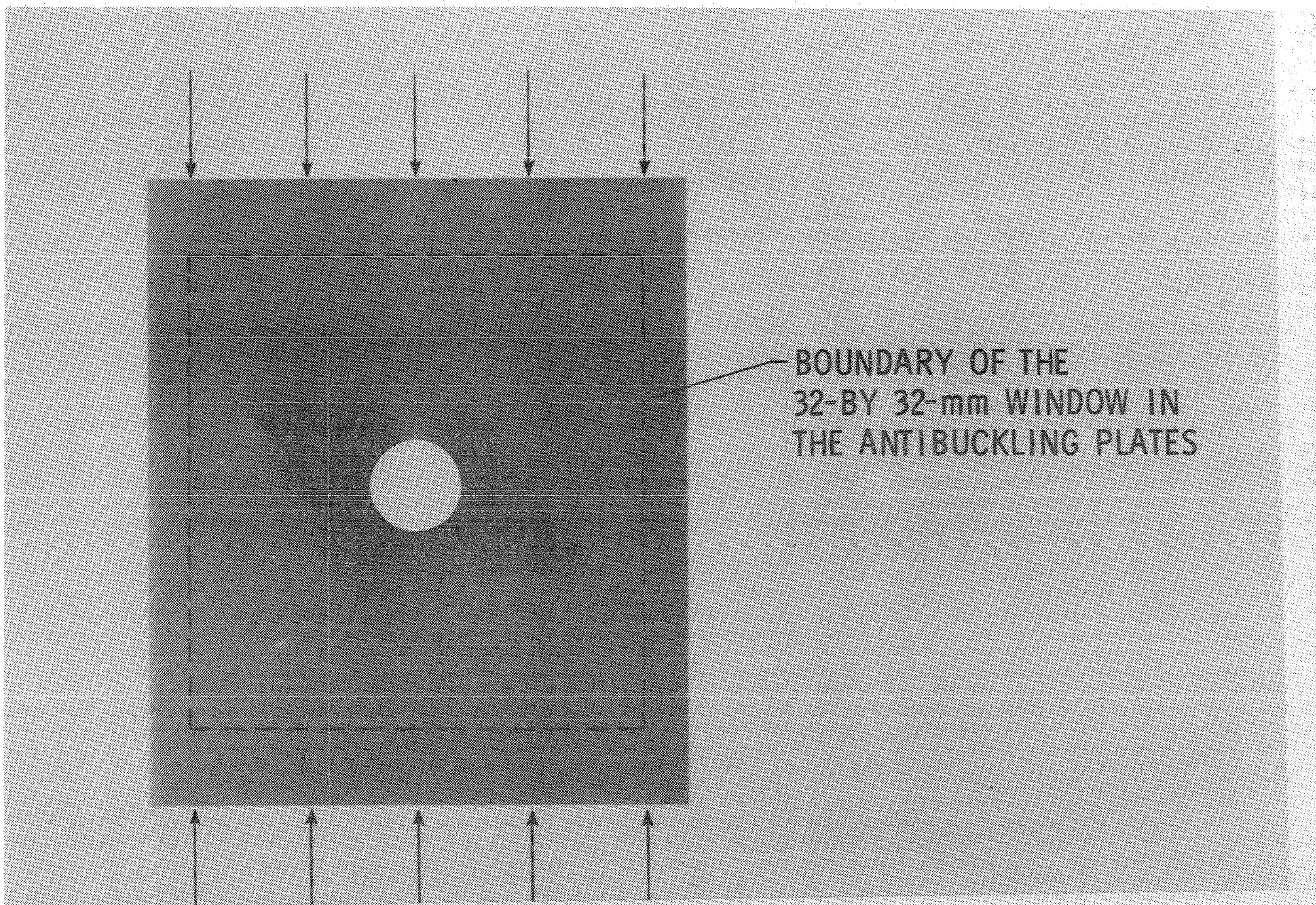

Figure 6.- Dye-enhanced radiograph showing the extent of the delamination and splitting near the end (at 93 percent of the life) of a test.

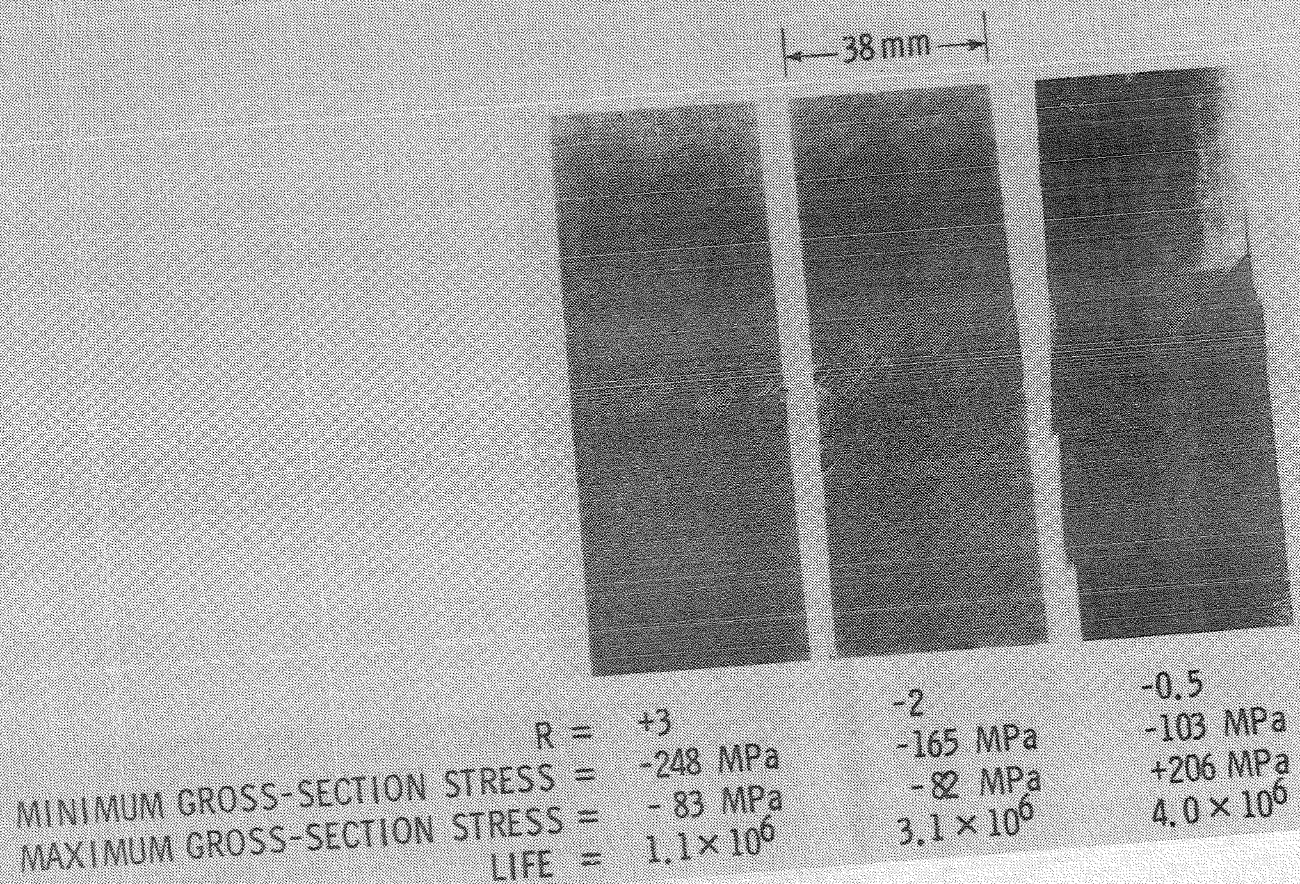

Figure 7.- Appearance of specimens after test at various R values. (T300/5208 graphite/epoxy, $[45/0/-45/90]_{2s}$, 32- by 32-mm window in antibuckling plates.)

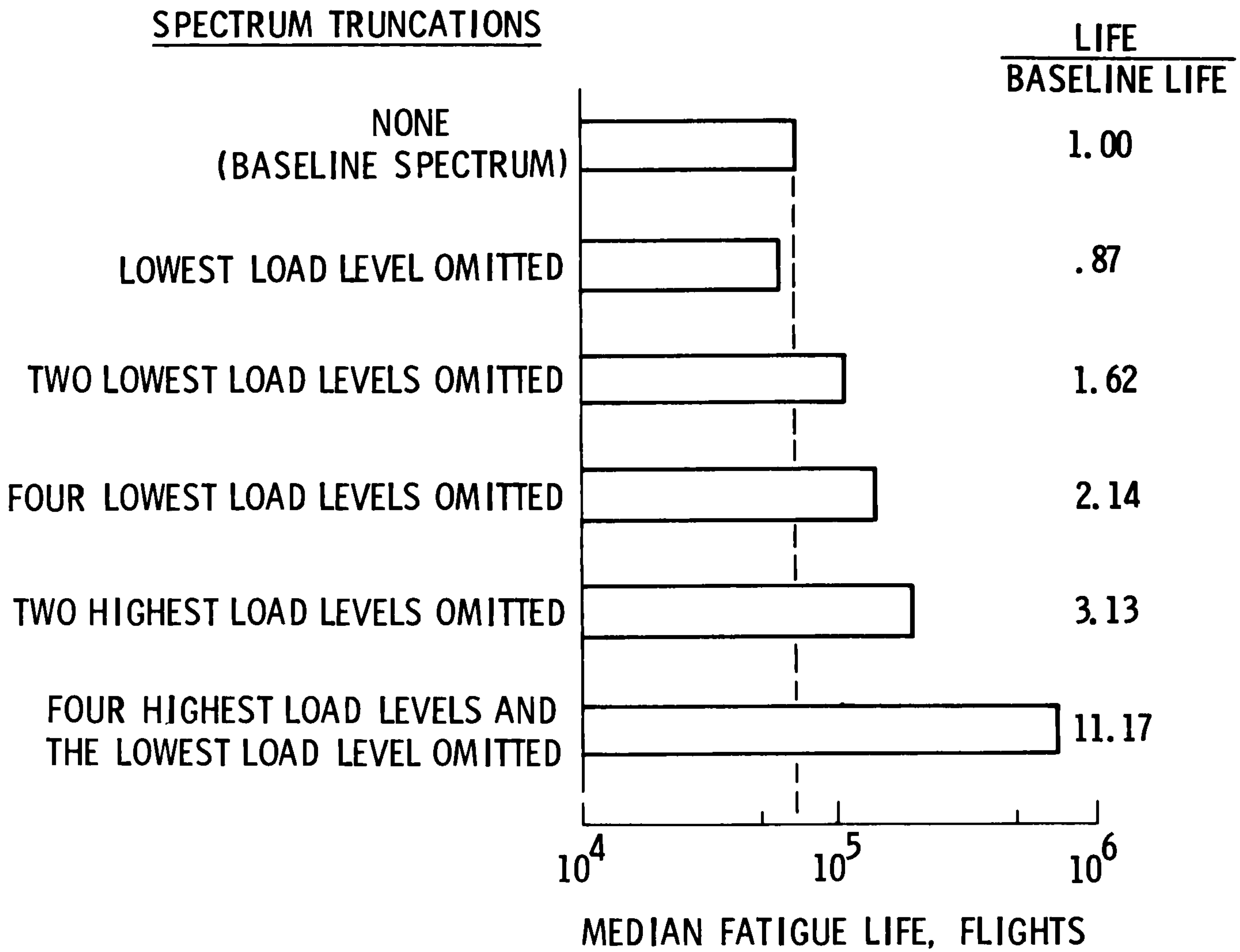

Figure 8.- Effect of spectrum truncations on fatigue life. (Transport upper-wing-surface spectrum, flight mean gross-section stress = -111 MPa, four tests per spectrum.)

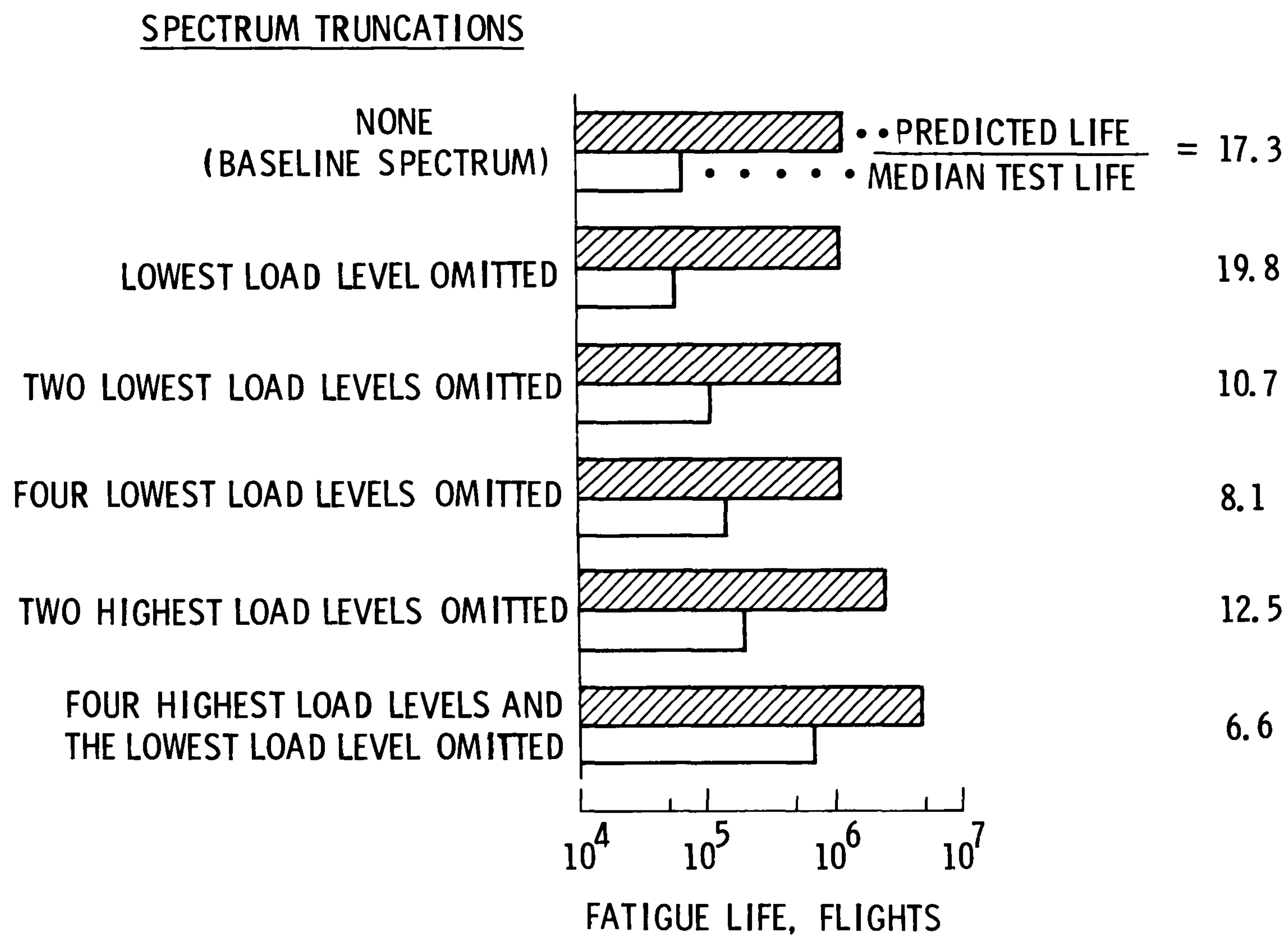

Figure 9.- Comparisons of lives predicted by the linear cumulative damage theory to actual test lives.

End of Document