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FATIGUE EVALUATION OF WING AND ASSOCIATED STRUCTURE ON SMALL AIRPLANES

Engineering and Manufacturing Division
Airframe Branch

MAY 1973

FINAL REPORT

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16. Abstract

Methods for evaluating the wing and associated structure for fatigue under the "safe life" and "fail safe" concept are outlined. Detailed procedures and scatter factors are given for full scale fatigue test, component test and the analytical methods of substantiation. Loading spectra are given for the various types of aircraft and usage.

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PREFACE

The aviation community both domestic and international has developed the need for a fatigue evaluation of the wing and carry-through structure under either the "safe life" or "fail safe" concepts. Recognizing this need, the FAA and the General Aviation Manufacturers Association (GAMA) formed independent teams to study the problem and develop more detailed procedures for accomplishing this evaluation. The FAA team consisted of A. Anderjaska, AFS-120, H. Nauert, ACE-212, and H. Leybold of NASA, Langley. This report essentially reflects the findings of the FAA team after review of the GAMA team recommendations. The input and advice of the GAMA project group, chaired by R. Christian of Aero Commander Division, North American Rockwell Corp. and of the Australian Department of Civil Aviation is appreciated.

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SECTION 1. INTRODUCTION

1. PURPOSE. Methods of fatigue evaluation are developed which can be used to evaluate the adequacy of small airplane structure from both a fail-safe and safe-life point of view.

2. BACKGROUND.

   a. It is recognized that, in such a complex problem as fatigue evaluation, new design features and methods of fabrication, new approaches to fatigue evaluation, and new configurations may require variations and deviations from the procedures described herein. Engineering judgment, guided by the extensive literature on the subject, should, therefore, be exercised for each particular application. The primary structure of the wing and carry-through is usually evaluated both for the original design and after any subsequent design changes which affect the loading spectra, internal stresses or stress concentrations. It should be noted that changes which are minor from a static strength standpoint can have a major effect on fatigue characteristics.

   b. A basic understanding of fatigue phenomena is necessary for an adequate fatigue evaluation. In general, fatigue is a progressive failure of a part under repeated, cyclic, or fluctuating loads. Any one application of any of these cyclic loads will not result in structural failure. The criterion for fatigue (Reference a) is the simultaneous action of cyclic stress, tensile stress and plastic strain.

   c. There are many investigations that collectively help give a better understanding of fatigue. The pioneering investigations in fatigue started over a hundred years ago. Yet, at this time, a fatigue prediction approach has not yet been developed for a complex fluctuating stress histories which does not require experimental support. The various cumulative damage theories are discussed and presented in their elementary forms in Reference (a) which also explains the technical reasoning and logic behind their formulation.

   d. The most widely used cumulative damage fatigue theory, in spite of its limitations, is the one developed by Palmgren-Miner hypothesis or Miner's Linear Cumulative Damage Theory which will be used in this report. The basic philosophy of Miner's Theory states that the fatigue damage introduced by a given stress level

is proportional to the number of applied cycles at that stress level divided by the total number of cycles to failure at the same stress level. This ratio is called a cycle ratio and is used to measure damage.

e. If various levels of stress amplitude are applied, the total damage is the sum of the different cycle ratios. Failure is predicted to occur when the sum of all the cycle ratios equals one. There are three distinct parameters that affect a fatigue damage. These are load application which includes the order and stress levels, the damage caused by a continuous loading at the same level, and the stress concentration present in the structure.

SECTION 2. FATIGUE STRENGTH EVALUATION

3. GENERAL. Under this procedure, conservative loading spectra are established, the mean fatigue life of the structure for the spectra is determined and a scatter factor is applied to the mean life to establish the safe life for the structure. Conservatism in the development of the loading spectra plus the use of the scatter factor is intended to assure that there is an extremely low probability that any individual airplane will experience a loading spectrum in excess of its fatigue life for that spectrum. The scatter factors specified vary with the type of substantiation (e.g. full scale testing, component testing, and analyses alone) to account for the difference in the degree of certainty associated with the substantiation method.

4. LOADING SPECTRA.

a. In any static structural analysis, only the magnitude of the load is required. In a fatigue analysis, the frequency and magnitude of the loads must be established. These loads are usually measured at the center of gravity of the aircraft. The principal types of loads are:

(1) Flight loads, comprised of gust and maneuver occurrences.

(2) Ground-Air-Ground (GAG) effects.

(3) Taxi loads.

(4) Landing impact loads.

b. The loading spectrum is a function of "type" of aircraft (e.g., single or twin engine, and pressurized or non-pressurized); and the utilization (e.g., "General" or "Special"). Spectra for both "General" and "Special" usage are to be developed for the particular

type of airplane involved. Special usage is intended to include low level overland survey and patrol and aerial application. General usage is intended to include all other usage. The following repeated loadings are to be included in the spectra under the following conditions unless shown to be insignificant:

(1) Flight loads. The spectra given for gust and maneuver loadings are based on VGH data (References b and c) for the particular type of aircraft and usage. The spectra based on Reference (b) were developed by increasing the mean frequency of exceedance shown in Reference (b) by a factor to account for variation in load spectra between individual aircraft of the same type and usage. This factor was based on the following general guidelines with judgment applied where the recorder data were considered to be untypical for the operational role involved. The standard deviation is derived from the variation in frequency of exceedance between individual airplanes in the operational role being recorded.

Single engine general usage -
mean executive spectrum plus 1.5 standard deviations

Twin engine general usage -
mean executive spectrum plus 2.0 standard deviations

Aerial application or low level survey -
mean spectrum plus 1.0 standard deviation

Whereas executive use was considered to be a typical role for twin engine airplanes, it was considered to be a more severe than average role for single engine airplanes. The spectrum from Reference (c) was considered sufficiently conservative without an additional factor.

(a) Gust loading. Spectra are given in Figures 1 through 5 for the various types of aircraft and usage. The spectra apply to both pressurized and non-pressurized aircraft unless otherwise stated. The spectra are expressed in terms of load factor ratio A_N/A_{NLLF} (incremental load factor at operating weight $\div$ incremental design limit gust load factor at maximum gross weight) exceedance per nautical mile. A_{NLLF} must be calculated as follows in order to be consistent with the derivation of the curves:

$$A_{NLLF} = \frac{30\ K\ Vm}{498\ (W/S)}$$

$K = 1/2\ (W/S)^{\frac{1}{4}}$ (for W/S < 16 p.s.f.)

$K = 1.33 - \frac{2.67}{(W/S)^{3/4}}$ (for W/S > 16 p.s.f.)

V = Airplane design cruising speed V_c, knots

M = Lift curve slope, C_L per radian

W/S = Wing loading, p.s.f. at maximum gross weight

(b) Maneuver loading. Spectra are given in Figures 6 through 8 for the various types of aircraft and usage. (The maneuver spectrum for aerial application will be added to this advisory when additional VGH data is collected.) The spectra apply to both pressurized and non-pressurized aircraft unless otherwise stated. The spectra are expressed in terms of load factor ratio A_N/A_{NLLF} (incremental load factor at operating weight ÷ incremental design limit maneuvering load factor at maximum gross weight).

(2) Ground-Air-Ground. A large stress change, which causes fatigue damage, occurs once per flight due to the operational cycle in which the airplane has negative or low positive loads while on the ground, encounters variable loads while taxiing, flies at 1g load level, is subjected to variable gust and maneuver load increments, fuel burnoff occurs and finally the airplane experiences the landing and taxi loads. This stress cycle is the ground-air-ground cycle which is defined as the cycle from the minimum (largest negative or smallest positive) stress to the maximum stress experienced on the average of once per flight. Where applicable, the maximum and minimum stress are to be taken from different environment, i.e., flight and ground. The length of flight to be used for the various types of aircraft and usage as follows:

(a) Single engine - General .65 hrs.;

Special - Low level survey 2.0 hrs.;

aerial application (will be added after collection of additional data).

(b) Twin engine - General .65 hrs.;

Special 3.0 hrs.

(c) Pressurized - General 1.1 hrs.;

Special (See (a) or (b) above)

(3) Landing impact loads. The spectra of landing impact loads are shown in Figure 9.

(4) Taxi loads. The spectra for taxi loads, based on References (e) and (f), are shown in Figure 10.

(5) Aircraft velocity. For determination of gust loads and miles flown (to be used in developing maneuver and gust spectra) the aircraft velocity is not to be less than 0.9 V_{no} (V_{MO}) for General usage or less than 100 knots or 0.9 Va, whichever is less, for Special usage. For determination of maneuver loads, the aircraft velocity is not to be less than 0.9 Va for General usage or less than 100 knots or 0.9 Va, whichever is less, for Special usage.

(6) Gross weight and load distribution. The gross weight and distribution of disposable load are to be based on conservative estimates of typical operating conditions.

(7) Positive and negative load cycles. While positive and negative load cycles are considered to occur in a random manner in service, the high positive and negative loads of a given type of repeated loading (e.g., gust, maneuver, etc.) tend to occur at the same time. Normally the high positive load cycles, for a given type of loading, are combined with the high negative load cycles of the same frequency. This procedure is to be followed in the case of gust and taxi loads.

(8) Stress levels. For one g the purpose of this evaluation, it can be assumed that the 1-g and the stress/g are equal provided the effects of buckling and yielding at high load factors and significant dynamic effects are accounted for. These stress values may be determined analytically or by strain-gage data from flight test and should represent the nominal (i.e. effect of stress concentration not included) stress at the critical area.

5. MEAN FATIGUE LIFE DETERMINATION. The following methods individually or in combination are acceptable to determine the mean fatigue life under the spectra established in paragraph 4 above for General or Special usage:

a. Full scale spectrum testing. The complete wing (left and right to nominally provide two specimens) and carry-through primary structure are to be tested under the selected usage spectrum of paragraph 4. Consideration is to be given to the possible consequences of omitting secondary structures. This is the most realistic method of determining the mean fatigue life, and the most realistic method of load application is a random application of loads on a flight-by-flight basis. The loads may, however, be applied in ordered loading blocks. Block length should be no greater than the number of flight hours (unfactored) that can be repeated ten times during the expected life and the sequence of loads should be from low to high to low within each block. At least six load levels should be used. The highest load level to be applied should not exceed limit load nor the load which will be equaled or exceeded only ten times in the expected life of the specimen. The test loads are to extend to the lowest level that causes significant fatigue damage unless such loads are otherwise accounted for. Fatigue damage resulting from omitting the lower levels can be simulated by a limited number of load applications at higher load levels in order to expedite testing, provided that sufficient cycles are applied to account for the effects of fretting. The ground-air-ground cycles are to be applied either individually or in frequent blocks, unless the fatigue damage is accounted for by the methods of paragraph 5c and corresponding scatter factors. The mean fatigue life for the General or Special usage spectrum is the mean test life indicated for the most critical area. The mean life for category not tested may be obtained by extrapolating from these tests results by the analytical methods of paragraph 5c.

b. Component testing. If less than the complete wing and carry-through primary structure is tested, great care is to be taken to assure that the test stresses are valid and that all critical portions are tested. Consequently, such an approach is limited to simple and determinate structure which is free of stresses due to excentricities, assembly preload, etc., unless sufficient adjacent structure is included to assure valid test stresses; or it is conclusively shown that the test stresses (including the peak local stresses) are valid by strain survey comparisons with complete structures for the test loading conditions. It will be acceptable to use component specimens which include portions of immediately adjacent structural elements such as skins, webs, rib attachments and fasteners (without separately

loading such elements) to simulate fretting action. The test specimen and surface conditions should be representative of the production article. The design, stress analysis, static test, strain surveys, tests of similar structural configurations, and service experience should be carefully reviewed to assure that all structural details, which are significant from a fatigue standpoint, are identified and that the most critical location of each type of detail is determined. Special attention should be given to areas of stress concentration such as joints, changes in section, sharp corners and rough surface finish. For spectrum testing, at least three specimens representative of the most critical location of each type of detail are to be tested under the selected spectrum of paragraph 4 using the test procedures of paragraph 5a. For S-N testing, a sufficient number of such specimens to define the mean S-N curve is to be tested. The mean fatigue lives for General and Special usage are to be determined as outlined in paragraph 5a if the components are subjected to spectrum tests, or paragraph 5c if component S-N curves are established.

c. Analytical Method. The locations to be analyzed are to be determined in the same manner as outlined for determination of the test areas in paragraph 5b. If the structure involved is conventional builtup aluminum structure with no fittings (other than continuous splice fittings), or parts with high residual stresses, or unique structural features, or theoretical stress concentrations (Reference d) greater than $K_T = 4$, then the S-N curves of Figure 11, adjusted for the proper mean stress per Figure 12 may be used. For other structures, such as lugs, joints, and fittings, applicable full scale and component S-N data may be used if the data are sufficient to define the S-N curve fully and accurately and if the specimens and loadings are sufficiently similar. A cautious and conservative approach should be used in determining whether data is applicable. The Miner-Palmgren hypothesis (linear damage accumulation such that $\sum n/N = 1$ at failure) is to be used to calculate the mean fatigue life under the General and Special loading spectra of paragraph 4. The appropriate loading spectrum should be divided into load intervals no larger than 0.2g when calculating $\sum n/N$. For aluminum structures, stress cycles below the S-N endurance limit at 3×10^7 cycles need not be considered. An example problem is given in the Appendix.

6. ESTABLISHMENT OF SAFE LIFE.

a. The safe life is to be based on the component found to have the lowest fatigue life. Safe lives are to be established for both General and Special usage by dividing the mean fatigue lives,

determine in paragraph 5 for the spectra of paragraph 4, by the following factors for conventional aluminum structure:

(1) Full scale spectrum testing - 3 to 4.
For the usual case, a scatter factor of four should be used for full scale spectrum testing. The factor may be reduced to three if equivalent safety is provided by determining crack location and growth rate and prescribing an inspection program based on this information that will assure that catastrophic failure will not result from initiation and growth of fatigue cracks. The specified inspection program should include specific information on when, where, and how to inspect the critical portions of the structure. The inspection openings and techniques should be adequate and appropriate to the inspection capability for the category of airplane involved.

(2) Component testing -5 to 7.
The factor will depend on the experience level of the applicant adjudged on the degree to which he develops a test loading and a specimen which accurately simulates operational loading and stress distributions and the full scale structure. This should include consideration of spectrum loading, realism of the spectrum, and the degree to which the test structure support and loading simulates that of the full-scale structure. The upper value would apply to the usual S-N test, while the lower value would apply to an exceptional realistic spectrum test of components.

(3) Analysis alone -7 to 8.
For the usual case a scatter factor of eight should be used for analysis alone. Where the designer presents data which shows that his knowledge of the stresses and fatigue properties of his structure is comprehensive based on flight measurements and on previous test and use of the type of construction in similar designs, a scatter factor as low as seven may be used.

b. If additional specimens are tested, the above test factors may be reduced by dividing by the following factor:

$$\text{antilog}\ (3.511 \times 0.14\ (1+1/N_s)^{\frac{1}{2}} - 3.511\ \sigma\ (1+1/N_t)^{\frac{1}{2}})$$

where -

N_s = number of specimens specified

N_t = number of specimens tested

σ = standard deviation of log of test life = 0.14 unless sufficient specimens tested to conclusively establish standard deviation.

c. Should an airplane that has previously been evaluated with a safe life be subjected to a mission change, gross weight increase, or gross weight increase with structural material added (without changing existing stress concentrations), to decrease the operating stress level, the scatter factor used in original evaluation would be applicable to adjust the previously established safe life.

SECTION 3. FAIL-SAFE STRENGTH EVALUATION

7. GENERAL. The fail-safe strength evaluation of the wing structure is intended to insure that should a serious fatigue failure occur, the remaining structure can withstand reasonable flight loads without excessive structural deformation. The fail-safe evaluation generally encompasses establishing the components which are to be made fail-safe, defining the loading conditions and extent of damage for which the structure is to be designed, conducting structural tests and analyses to substantiate that the design objective has been achieved, and establishing inspection programs aimed at detection of fatigue damage. Design features which may be used in attaining a fail-safe structure are:

   a. Use of multipath construction and the provision of crack stoppers to limit the growth of cracks.

   b. Use of composite duplicate structures so that a fatigue failure occurring in one-half of the composite member will be confined to the failed half and the remaining structure will still possess appreciable load-carrying ability.

   c. Use of backup structure wherein one member carries all the load, with a second member available and capable of assuming the extra load if the primary member fails.

   d. Selection of materials and stress levels that provide a contolled slow rate of crack propagation combined with high residual strength after initial of cracks.

   e. Arrangement of design details to permit easy detection of failures in all critical structural elements before the failures can become dangerous or result in appreciable strength loss, and to permit replacement or repair.

8. IDENTIFICATION OF PRINCIPAL STRUCTURAL ELEMENTS. Principal structural elements are those which contribute significantly to carry flight loads and whose failure can result in catastrophic failure of the aircraft.
Typical examples of such elements are:

   a. Attachment fittings.

   b. Integrally stiffened plates.

   c. Primary fittings.

   d. Principal splices.

   e. Skin or reinforcement around cutouts or discontinuities.

f. Skin-stringer combinations.

g. Spar cap.

h. Spar web.

9. EXTENT OF FAIL-SAFE DAMAGE. Each particular design should be carefully assessed to establish appropriate damage criteria. In any fatigue damage determination, when it is not possible to establish the extent of damage in terms of an "obvious partial failure," the damage should be considered in terms of the complete failure of the single element involved. Thus, an obvious partial failure can be considered to be the extent of the fail-safe damage, provided a positive determination is made that the fatigue cracks will propagate in the open; for example, cracks that occur in exterior skins and which can be detected by a visual inspection at an early stage of the crack development. Typical examples of wing fatigue damage which should be considered are outlined below:

a. Skin cracks emanating from the edge of structural openings or cutouts which can be readily detected by visual inspection of the area.

b. Failure of one element where dual construction is utilized in components such as spar caps and wing attach fittings.

c. The presence of a fatigue crack in at least the tension portion of the spar web or similar elements.

d. Failure of primary attachments.

10. INACCESSIBLE AREAS. In cases where inaccessible or blind areas are unavoidable, emphasis should be placed on determining crack propagation and residual strength of the particular fatigue-damaged structure in order to assure continued airworthiness of the structure with reasonable inspection methods and controls by the operator. Alternative procedures would be to provide additional fatigue strength to preclude fatigue cracking in the blind element or to conduct fatigue tests of the blind areas to establish that a high service life is provided.

11. TESTING OF PRINCIPAL STRUCTURAL ELEMENTS.

a. The nature and extent of tests on complete structures and/or portions of the primary structure will depend upon previous experience with similar types of structures regarding tests of this nature and the

crack propagation characteristics of the structure. Single elements or members such as stringers and spar caps, should be completely severed and 1.15 times the critical fail-safe load applied after severing. In cases where definite evidence is furnished that the dynamic failure effects are not present, the 1.15 factor may be eliminated or reduced in accordance with the effects noted. Alternatively, the fail-safe loads may be applied to the structure before severing and the 1.15 factor omitted.

b. In the case of distributed members such as a sheet-stringer combination or an integrally stiffened tension skin, a cut may be made to represent an initial crack in the element under test. If there is no failure, the length of the cut may be increased and the fail-safe load applied until either:

(1) The fail-safe damage has been simulated, or

(2) The crack propagation rate decreases due to redistribution of load path, or

(3) The crack propagation stops due to a crack stopper.

c. The simulated cracks should be as representative as possible of actual fatigue damage. In cases where it is not practical to produce actual fatigue cracks, damage may be simulated by cuts made with a fine saw, sharp blades, or a guillotine. In those cases where it is necessary to simulate damage at joints of fittings, bolts may be removed to simulate the failure if this condition would be representative of an actual failure.

12. ANALYSIS OF PRINCIPAL STRUCTURAL ELEMENTS.

a. In some instances, the fail safe characteristics may be shown analytically. The analytical approach may be used when the structural configuration involved is essentially similar to one already verified by fail-safe tests, whether conducted on a previously approved type design, or whether conducted on other similar areas of the design currently being evaluated.

b. The analytical approach may also be used when conservative failures are assumed such that the failure would be detected considerably before the critical crack length is approached and the margins of safety resulting from the analysis are well in excess of the fail-safe residual static strength level. In any such analysis, the 1.15 factor should be included unless it can be shown as indicated in paragraph 14 that this factor is not required.

13. SELECTION OF A CRITICAL AREA. Single principal structural elements and detail design points requiring investigation are identified under paragraph 8. The process of actually locating where fail-safe damage should be simulated in an element, such as a wing spar chord, requires use of sound engineering judgment that takes into account a variety of factors, such as:

a. Conducting an analysis to locate areas of maximum stress and low margin of safety.

b. Conducting strain gage surveys on undamaged structure to establish points of high stress concentration as well as the magnitude of such concentration.

c. Examining static test results to determine locations where excessive deformations occurred.

d. Determining from repeated load tests where failure may have initiated or where the crack propagation rate is a maximum.

e. Selecting locations in an element (such as spar cap) where the stresses in adjacent element (such as the spar web or wing skin) would be the maximum with the spar failed.

f. Selecting points in an element (such as a spar web) wherein high stress concentrations are present in the residual structure with the web failed.

g. Assessing detail design areas which service experience records of similarly design components indicate are prone to fatigue damage.

14. DYNAMIC EFFECTS. The dynamic magnification factor of 1.15 should be applied to all loads unless fail-safe tests are performed under load or the dynamic effects are shown to be negligble by dynamic test data from a similar structure.

15. INSPECTION. Detection of fatigue cracks before they become dangerous is the ultimate control in insuring the fail-safe characteristics of the wing structure. Therefore, the aircraft manufacturer should provide sufficient guidance information to assist operators in establishing the frequency and extent of the repeated inspections of the critical structure. Where these inspections involve more than a general visual inspection of external and easy access areas, then frequency and extent are to be specified in a required document (placards, markings, or manuals).

16. REFERENCES.

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e. "A Note on the Loads Imposed on Fixed Wing Light Aircraft During Agriculture Operations." Canadian National Aeronautical Establishment Report LTR-ST-422, August 1970.

f. "Airplane Strength and Rigidity Reliability Requirements, Repeated Loads, and Fatigue," Military, Specification MIL-A-8866 (ASG), May 1960.

APPENDIX 1. SAMPLE FATIGUE LIFE CALCULATION

1. ASSUMPTIONS:

   a. Single engine, non-pressurized aircraft.

   b. General usage.

   c. Construction is such that S-N curve of Figure 12 applies.

   d. Average speed = .9 V_{no} = 139.5 knots.

   e. A_{NLLF} = 2.5g for gust, +2.8, -2.5 for maneuver.

   f. All stresses are nominal stresses at critical point under conservative operating weights and load distributions.

   g. 1g flight stress = +5,500 psi.

   h. Sa = alternating stress = 5,500 x Δg for gust and maneuvers.

   i. Critical point is inboard of landing gear.

   j. Sm = mean stress = +2,750 psi for landing cycle.

   k. Sa for maximum landing cycle = 1,200 + 1,200 x sink speed (ft./sec.) for sink speeds equal to or greater than 1 ft./sec. and = 2,400 psi for lower sink speeds.

   l. Sm for taxi cycle = +850 psi.

   m. Sa for taxi cycle = 5,150 x Δg.

   n. 1.54 flights/hour.

2. GENERAL PROCEDURE. Calculate fatigue damage per hours due to gusts, maneuver, taxi, landing impact and ground-air-ground cycle separately and add to obtain total. The number of load levels considered should be sufficient to accurately represent the spectrum, and the selection of load levels should be such that the maximum damage results. A. 0.2 g load increment with the lowest increment selected equal to the endurance limit of the SN curve at 3 x 10^7 cycles has generally been adequate. Conventionally, the positive load exceedances for a given load level are combined with a negative load exceedances for a level which occurs at the same frequency and the mean and alternating stress are determined from the combined cycle.

TABLE I

GUST DAMAGE CALCULATION

Loading Spectrum of Figure 1 Used And Considered Essentially Symmetrical
With Sm = 5,500 psi

(1) A_n/A_{NLLF}	(1A)	(2) Fig. 1 CUM. /~~KT.~~ nmi	(3) n FREQ. /~~KT.~~ nmi	(4) 139.5 x (3) FREQ./HR.	(5) 2.5 x (1A) An	(6) 5500 x (5) Sa	(7) Fig 11 N x 10^6	(8) (4) ÷ (7) n/N x 10^{-6} /hr.
.092		.60						
AVG =	.136		.49	68.36	.340	1875	30	2.2787
.18		.11						
	.23		.078	10.88	.50	2750	7.2	1.5111
.24		.032						
	.28		.0268	3.739	.70	3850	2.1	1.7805
.32		.0052						
	.36		.0041	.5720	.90	4950	.80	.7150
.40		.0011						
	.44		.00078	.1088	1.10	6050	.35	.3109
.48		.00032						
	.52		.00022	.03069	1.30	7150	.19	.1615
.56		.00010						
	.60		.000065	.009068	1.50	8250	.11	.0824
.64		.000035						
	.68		.000020	.00279	1.70	9350	.068	.0410
.72		.000015						
	.76		.0000096	.001339	1.90	10450	.042	.0319
.80		.0000054						
	.84		.0000054	.0007533	2.10	11550	.028	.0269
							TOTAL	6.9400

TABLE II

MANEUVER DAMAGE CALCULATION

Loading Spectrum of Figure 6 used which is unsymmetrical with a variable Sm

A_N/A_{NLLF}		CUM/KT.	FREQ/KT.	n FREQ./HR.		An	Sm	Sa	N x 10^{-6}	n/N x 10^{-6}
	+.14	.012								
-.07	+.18		.0085	1.186	- .18	+ .50	6380	1870	23	.0516
	+.22	.0035								
-.13	+.26		.0023	.3209	- .33	+ .73	6600	2915	4.2	.0764
	+.30	.0012								
-.20	+.34		.00084	.1172	- .50	+ .95	6737	3988	1.25	.0938
	+.38	.00036								
-.25	+.42		.00022	.03069	- .63	+1.18	7012	4978	.48	.0693
	+.46	.00014								
-.31	+.50		.000088	.01228	- .78	+1.40	7205	5995	.235	.0523
	+.54	.000052								
-.36	+.58		.000032	.004464	- .90	+1.62	7480	6930	.125	.0357
	+.62	.00002								
-.42	+.66		.0000012	.001674	-1.05	+1.85	7700	7975	.070	.0239
	+.70	.000008								
-.47	+.74		.0000051	.0007115	-1.18	+2.07	7947	8938	.040	.0178
	+.78	.0000029								
-.53	+.82		.0000029	.0004046	-1.33	+2.30	8167	9983	.027	.0150
									TOTAL	.4358

3. CALCULATION OF TAXI AND LANDING DAMAGE. The stress / g due to one g taxi and landing accelerations is commonly assumed to be equal to the 1-g or steady stress condition, but as illustrated by the given Sa values, which are based on measured values, considerable stress amplification can occur when large masses and/or flexible structures are involved. The amplitude and number of load factor cycles per landing and their variation with landing sink speed is determined from drop tests or flight measurements and the resulting stress by flight measurements or analysis which includes the effect of wing lift. For purposes of this illustration it will be assumed that there are two cycles per landing with the second cycle having an amplitude of .6 of the maximum cycle as suggested by measurements in one case. The sink speed spectrum should be based on the critical major use anticipated for the design. For purposes of this illustration it will be assumed that training is a major use for the design being analyzed.

4. CALCULATION OF GROUND-AIR-GROUND DAMAGE. The GAG cycle is obtained by developing cumulative frequency curves for positive and for negative stress exceedances from all sources and then determining the average exceedance per flight. The overall cumulative frequency curves are obtained by adding the cumulative frequency curves for gusts, maneuvers, landing and taxi.

TABLE III

TAXI DAMAGE CALCULATION

Taxi Spectrum of Figure 10, Which is Symmetrical, Used With S_{m} = 850 psi

g	CUM/LAND	FREQ	n FREQ/HR	Sa	$\times 10^{6}$ N	$\times 10^{-6}$ n/N
.05	350					
.10		266	409.6	515		-
.15	84					
.20		75.3	116.0	1030		-
.25	8.7					
.30		8.23	12.67	1545		-
.35	.47					
.40		.453	.6976	2060		-
.45	.017					
.50		.017	.02618	2575		-

ZERO DAMAGE

TABLE IV

LANDING DAMAGE CALCULATION

Sink Speed Spectrum of Figure 9, which is Symmetrical used with S_m = 2750 psi

SINK SPEED	CUM/LAND	FREQ.	FREQ./HR.	Sa	$\times N_1 10^6$	$\times n/N_1 10^{-6}$	.6Sa	$\times N_2 10^6$	$\times n/N_2 10^{-6}$
0	1.0								
.5		.30	.462	2400	30.	.0154	1440	∞	-
1.0	.70								
1.5		.32	.493	3000	13	.0379	1800	∞	-
2.0	.38								
2.5		.24	.370	4200	3.3	.1321	2520	28.	.0132
3.0	.14								
3.5		.095	.146	5400	1.1	.1327	3240	10	.0146
4.0	.045								
4.5		.033	.0508	6600	.48	.1058	3960	3.8	.0134
5.0	.012								
5.5		.0089	.0137	7800	.24	.0571	4680	2.0	.0069
6.0	.0031								
6.5		.0022	.00339	9000	.13	.0300	5400	1.1	.0035
7.0	.0009								
7.5		.0006	.00924	10200	.075	.0123	6120	.65	.0014
8.0	.0003								
8.5		.0002	.000308	11400	.046	.0067	6840	.45	.0007
9.0	.0001				SUBTOTALS	.5300			.0537

TOTAL DAMAGE $.5837 \times 10^{-6}$/Hr.

TABLE V

GAG DAMAGE

STRESS	GUST CUM./FLT.	MANEUVER CUM./FLT.	LANDING CUM./FLT.	TAXI CUM./FLT.	TOTAL CUM./FLT.
+ 9650	.73	.105	.165	-	1.0
- 900	.037	.007	.336	.59	1.0

Resulting in a GAG Cycle -900 +9650

Sm	Sa	n FREQ./HR	N x 10^6	n/N x 10^{-6}
4375	5275	1.54	.75	2.0533

DAMAGE = 2.0533 x 10^{-6}/HR.

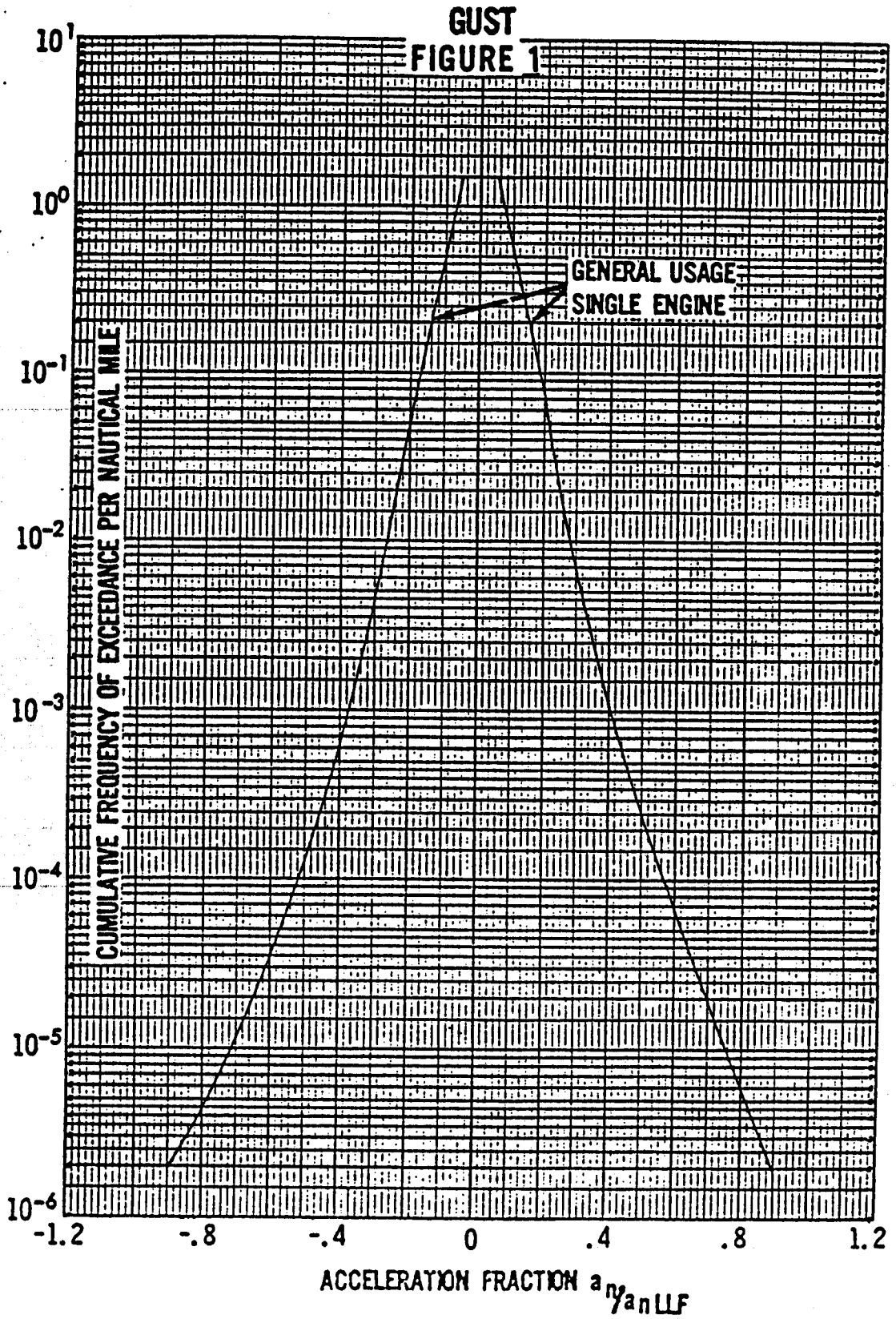
TABLE VI

TOTAL FATIGUE DAMAGE

	$\times 10^{-6}$ /HR.
GUST	6.9400
MANEUVER	.4358
GAG	2.0533
LANDING	.5837
TAXI	-
TOTAL	10.0128×10^{-6} /HR.

UNFACTORED LIFE = 99, 872

SAFE LIFE (ANALYSIS) = 12, 484 HRS.


GUST
FIGURE 1
GENERAL USAGE
SINGLE ENGINE
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10^1
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
10^{-6}
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n\,LLF}$

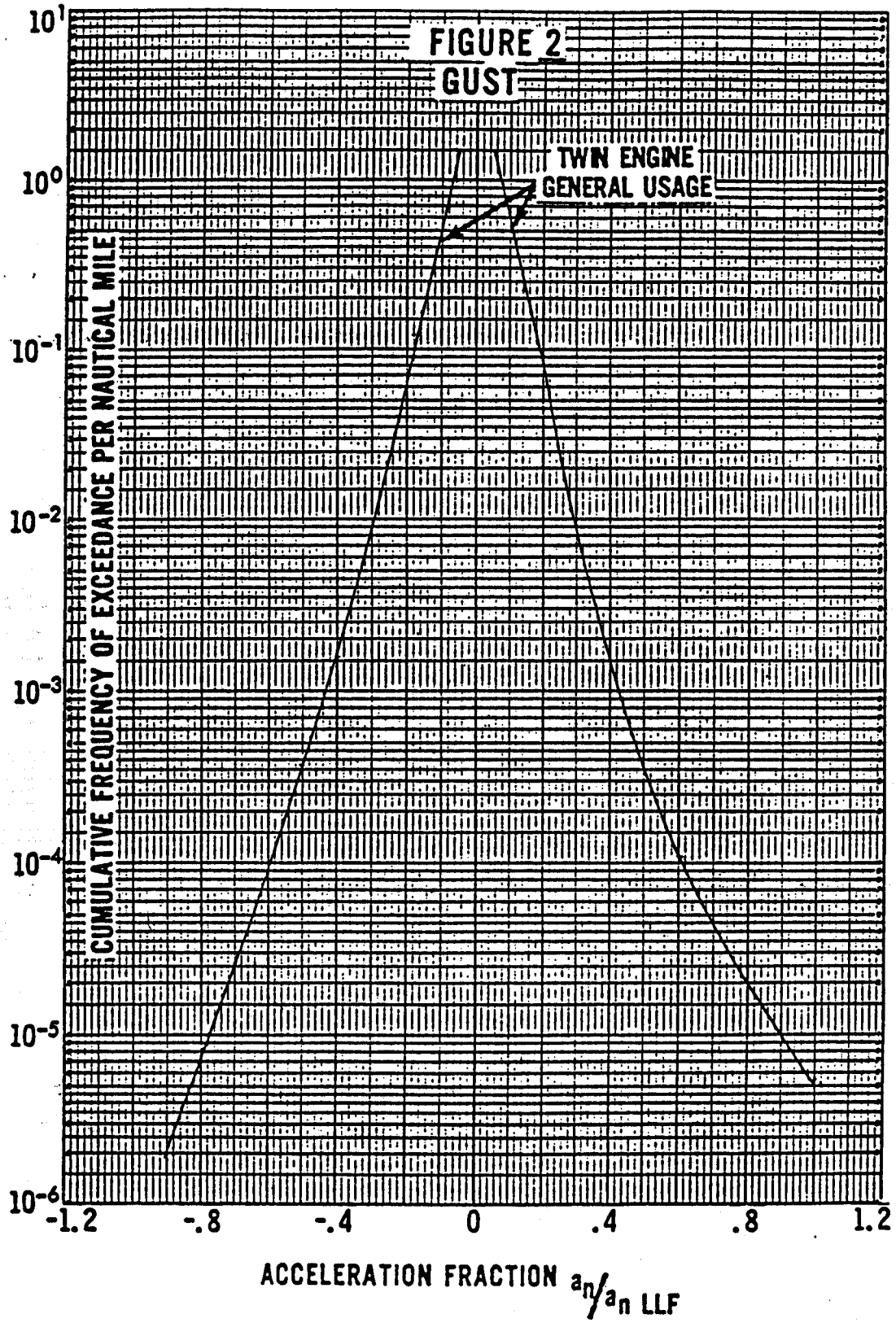


FIGURE 2
GUST
TWIN ENGINE
GENERAL USAGE
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10^1
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
10^{-6}
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n\ LLF}$

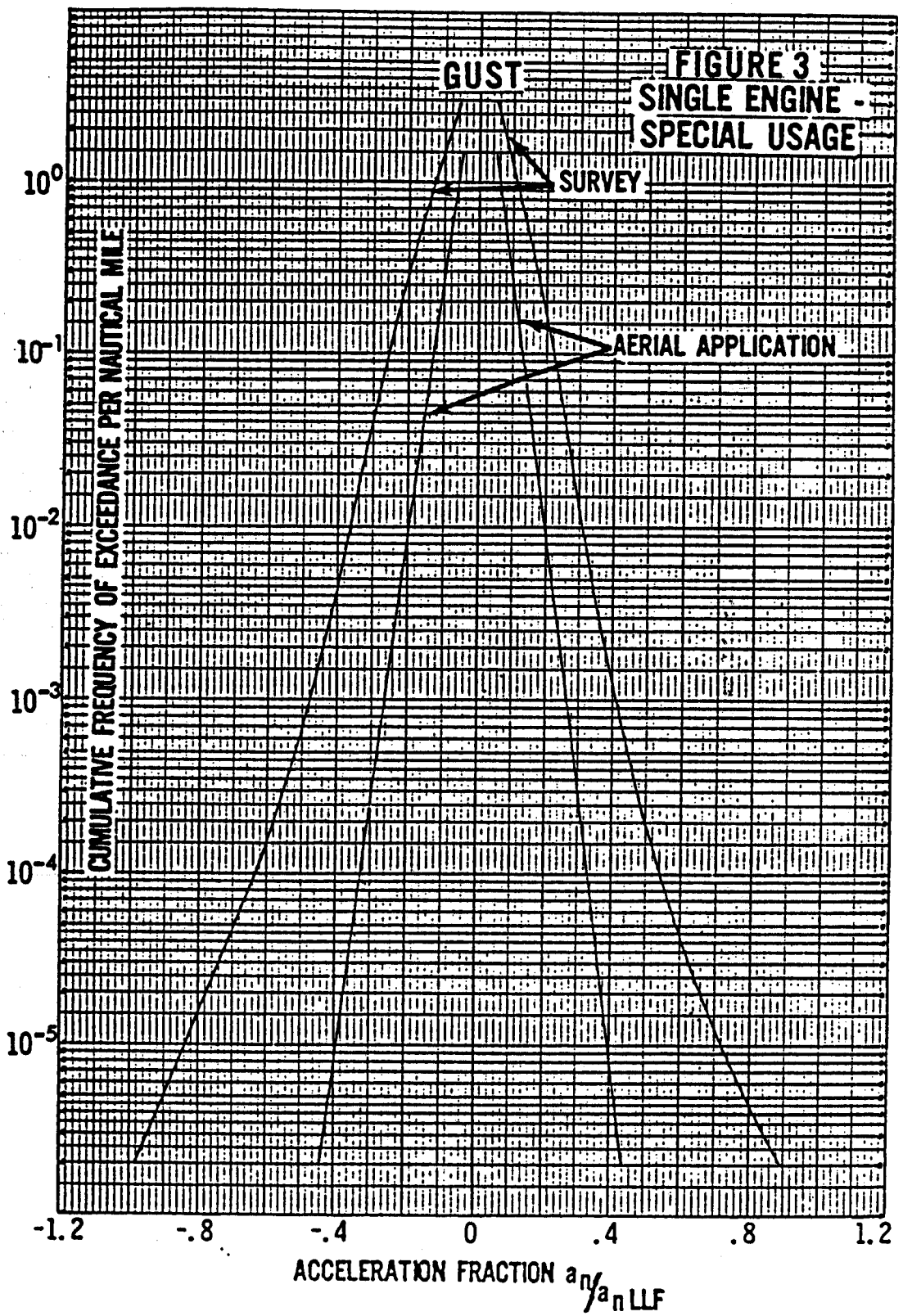


FIGURE 3
SINGLE ENGINE - SPECIAL USAGE
GUST
SURVEY
AERIAL APPLICATION
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n\,LLF}$

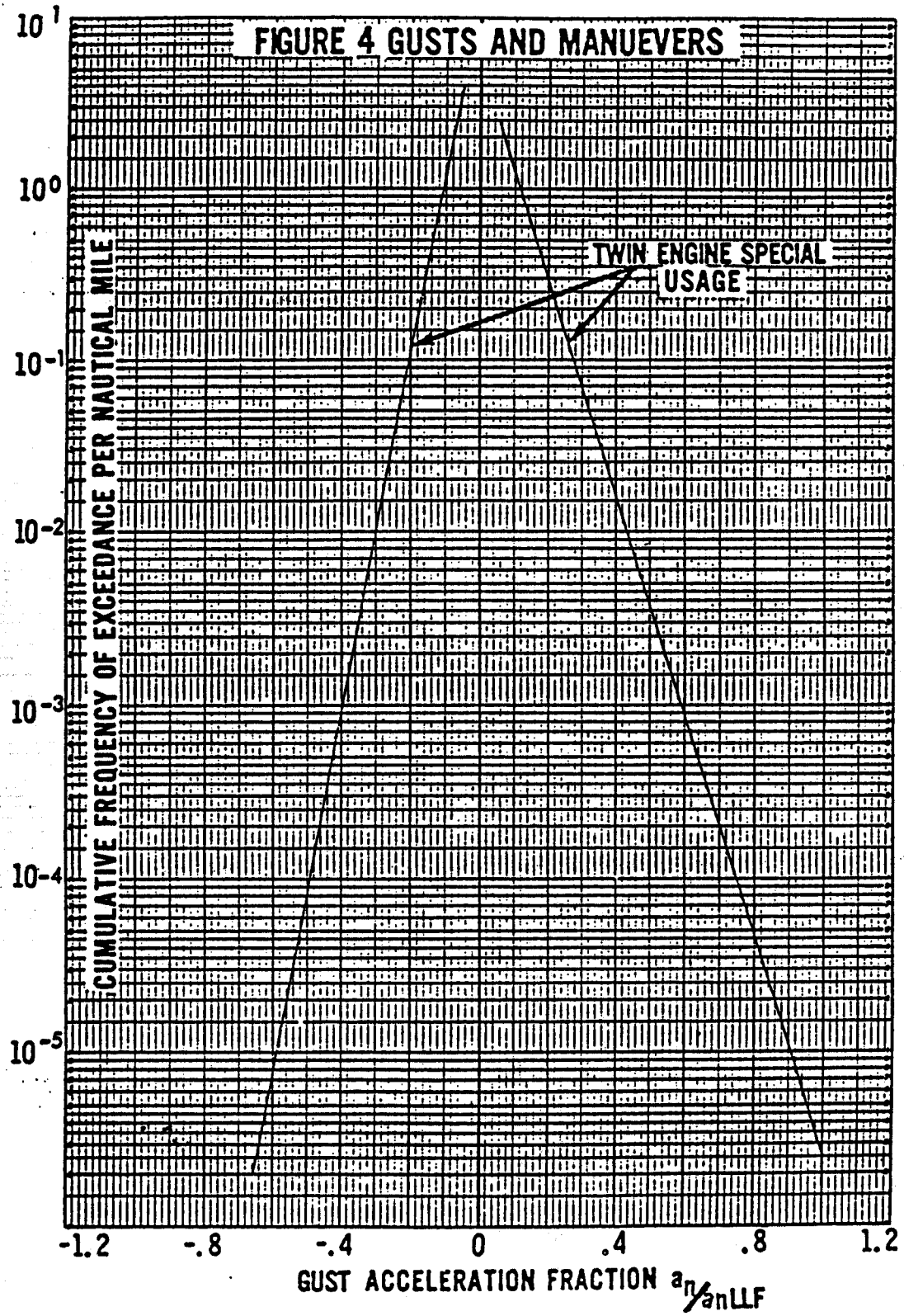
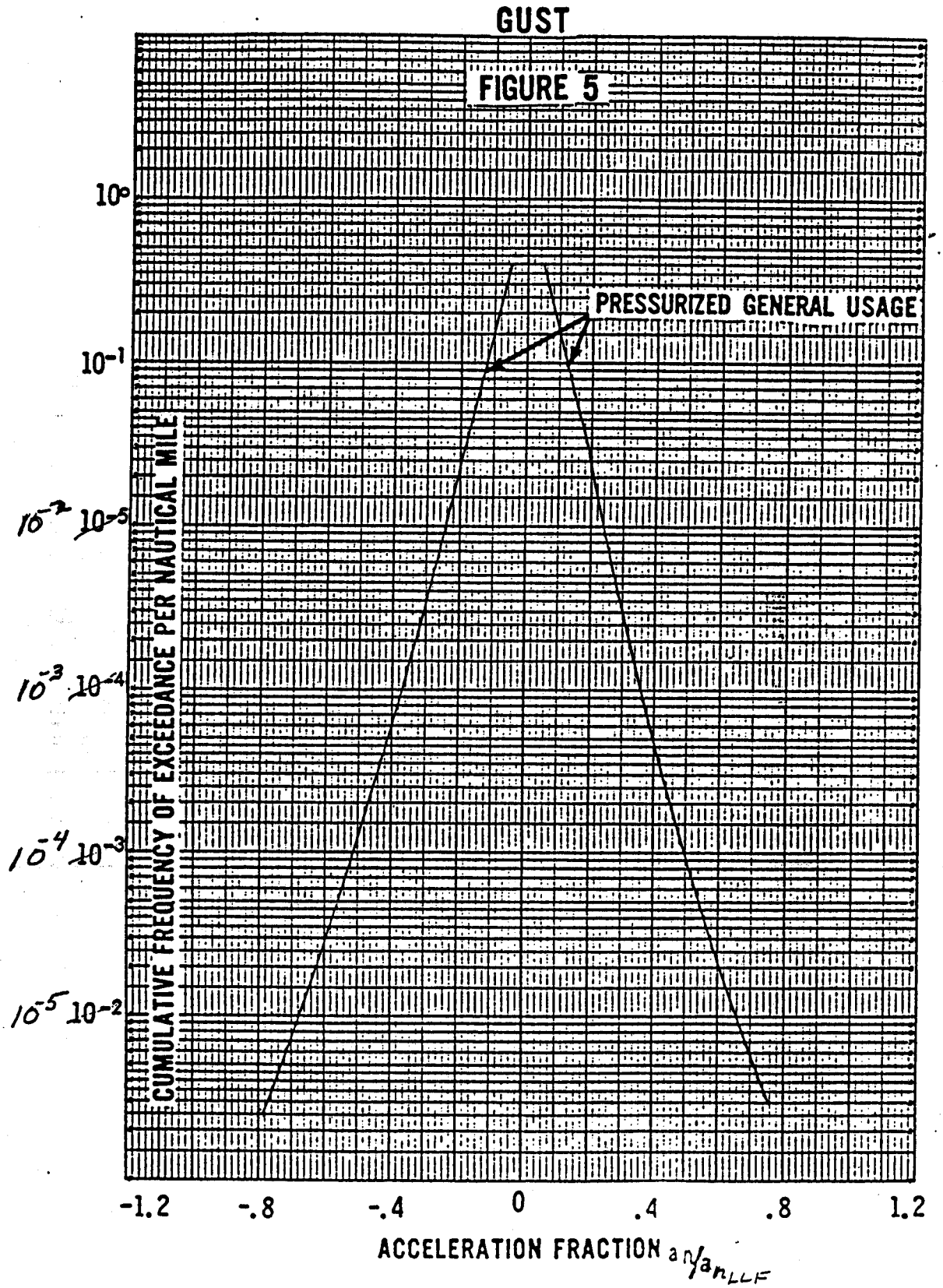
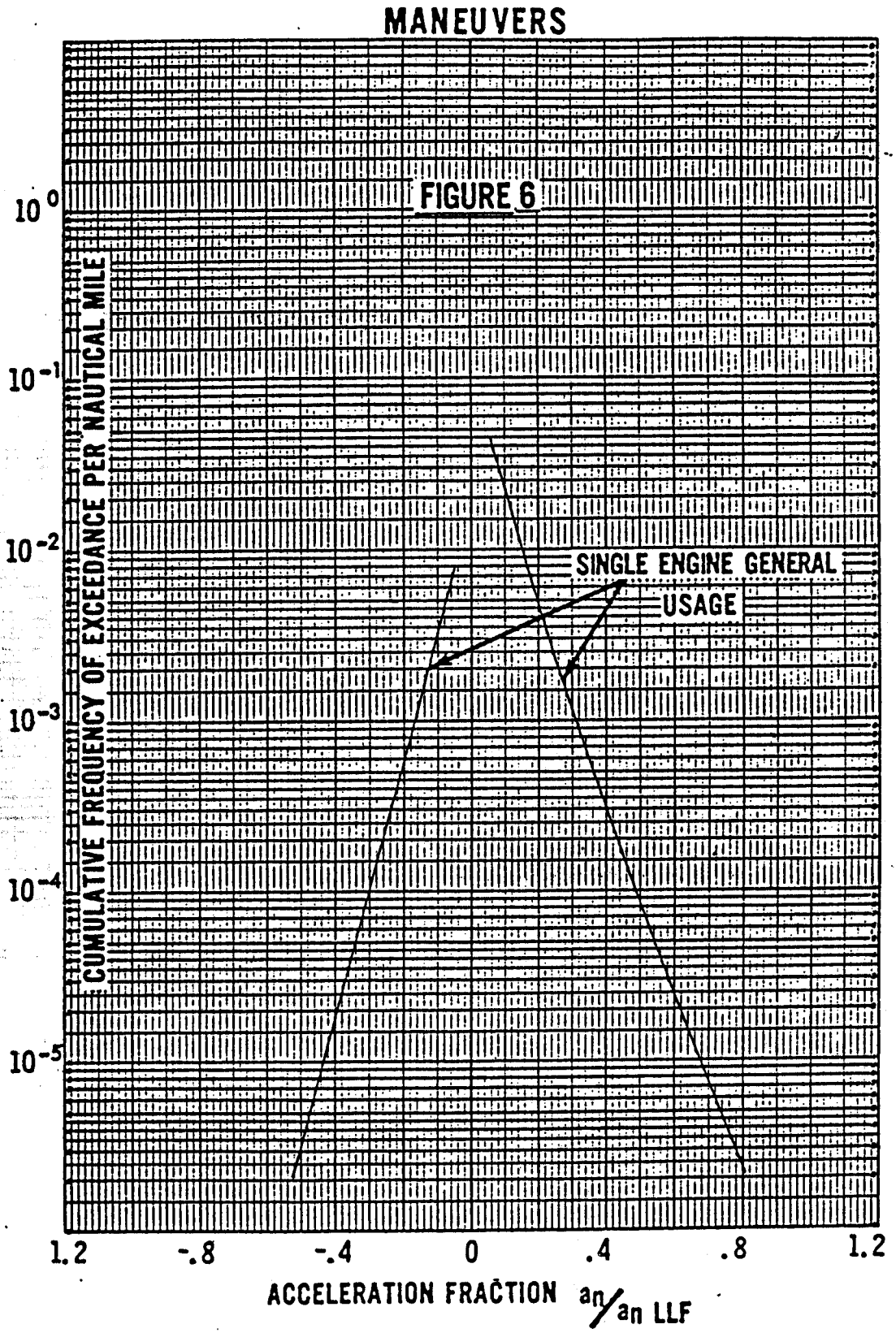
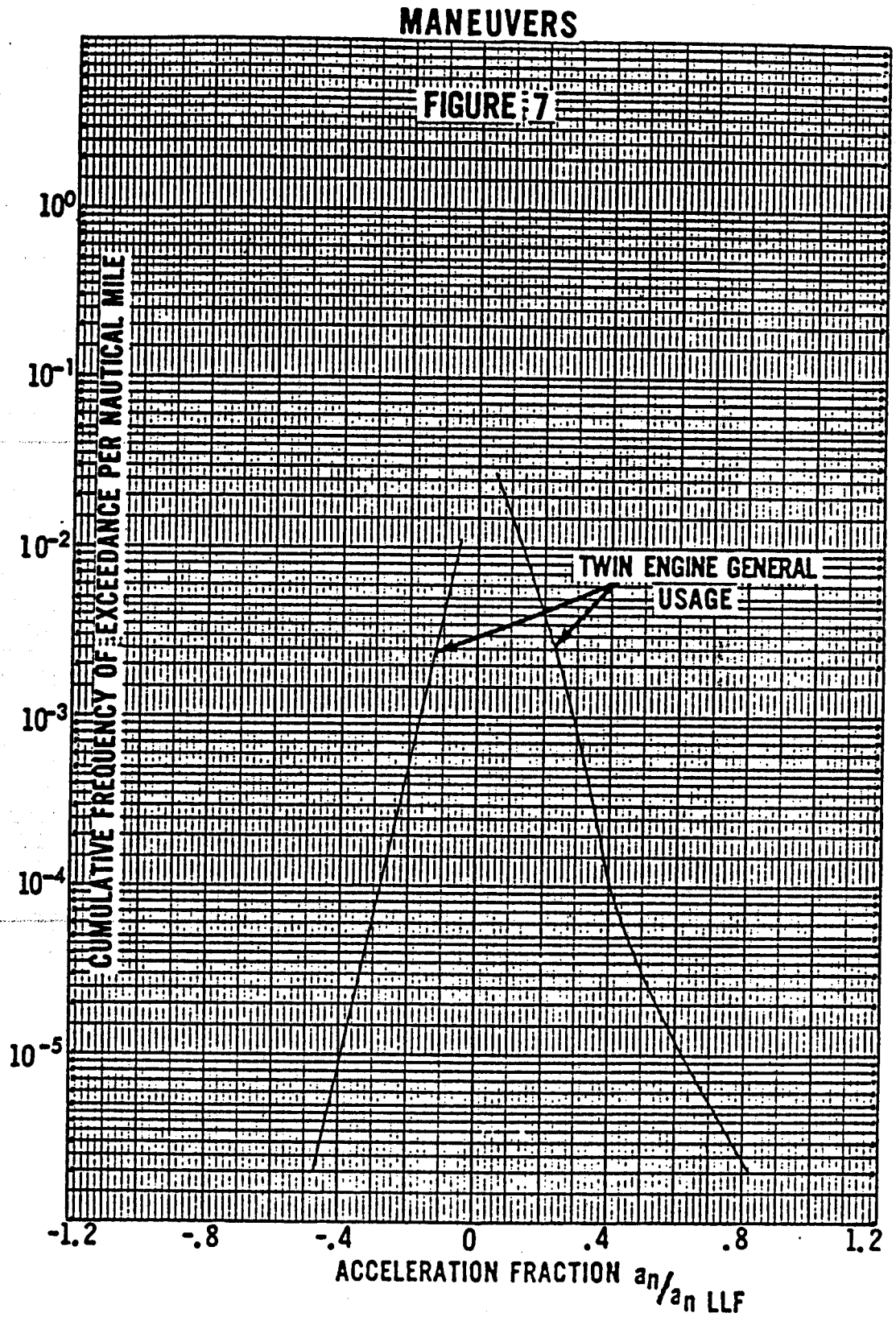
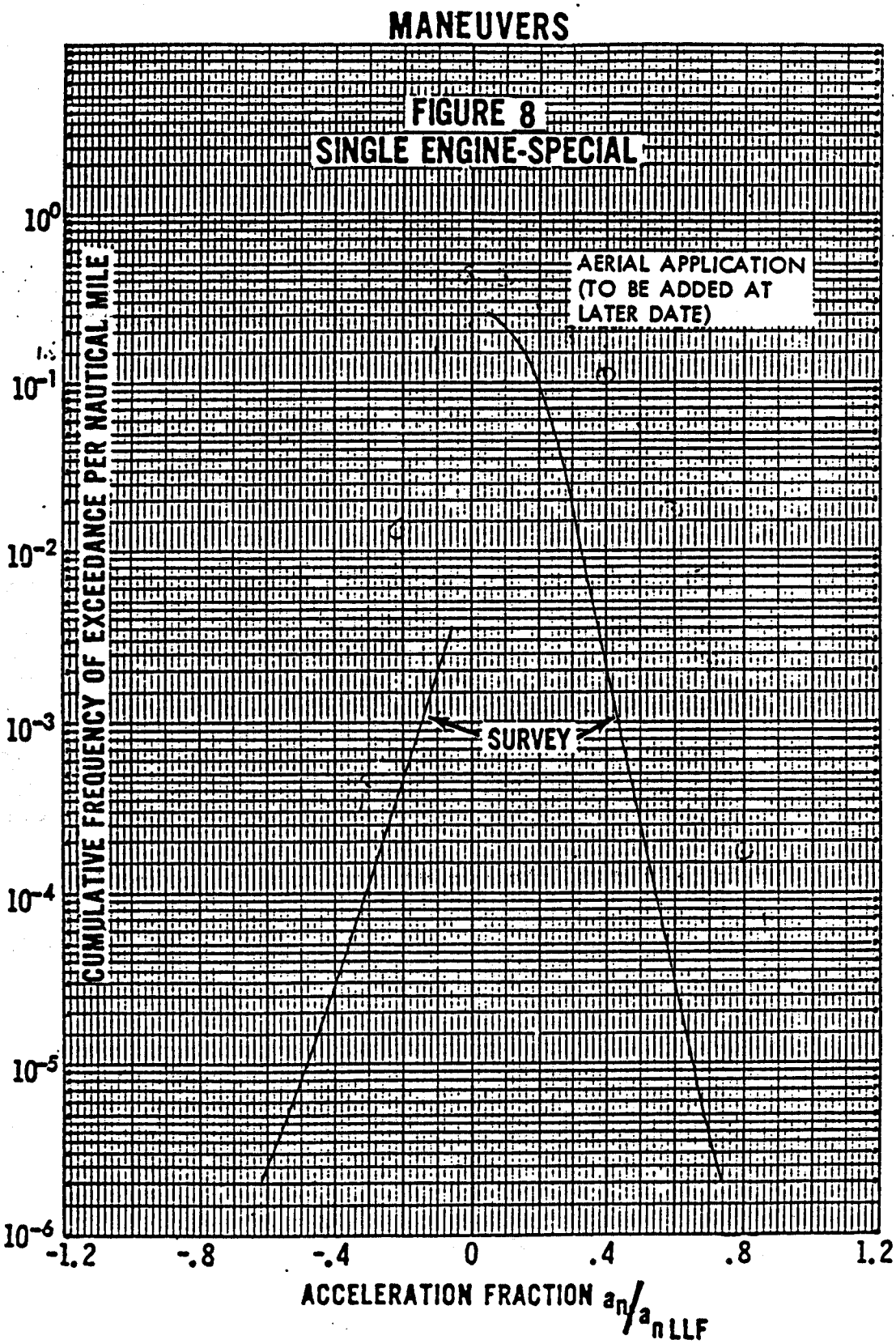


FIGURE 4 GUSTS AND MANUEVERS
TWIN ENGINE SPECIAL USAGE
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10 1
10 0
10-1
10-2
10-3
10-4
10-5
-1.2
-.8
-.4
0
.4
.8
1.2
GUST ACCELERATION FRACTION an/anLLF


GUST
FIGURE 5
PRESSURIZED GENERAL USAGE
10^0
10^{-1}
10^{-2} 10^{-5}
10^{-3} 10^{-4}
10^{-4} 10^{-3}
10^{-5} 10^{-2}
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n_{LLF}}$


MANEUVERS
FIGURE 6
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10^{0}
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
SINGLE ENGINE GENERAL
USAGE
1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n\ LLF}$


MANEUVERS
FIGURE 7
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
TWIN ENGINE GENERAL USAGE
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n\ LLF}$


MANEUVERS
FIGURE 8
SINGLE ENGINE-SPECIAL
AERIAL APPLICATION (TO BE ADDED AT LATER DATE)
SURVEY
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
10^{-6}
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n\,LLF}$

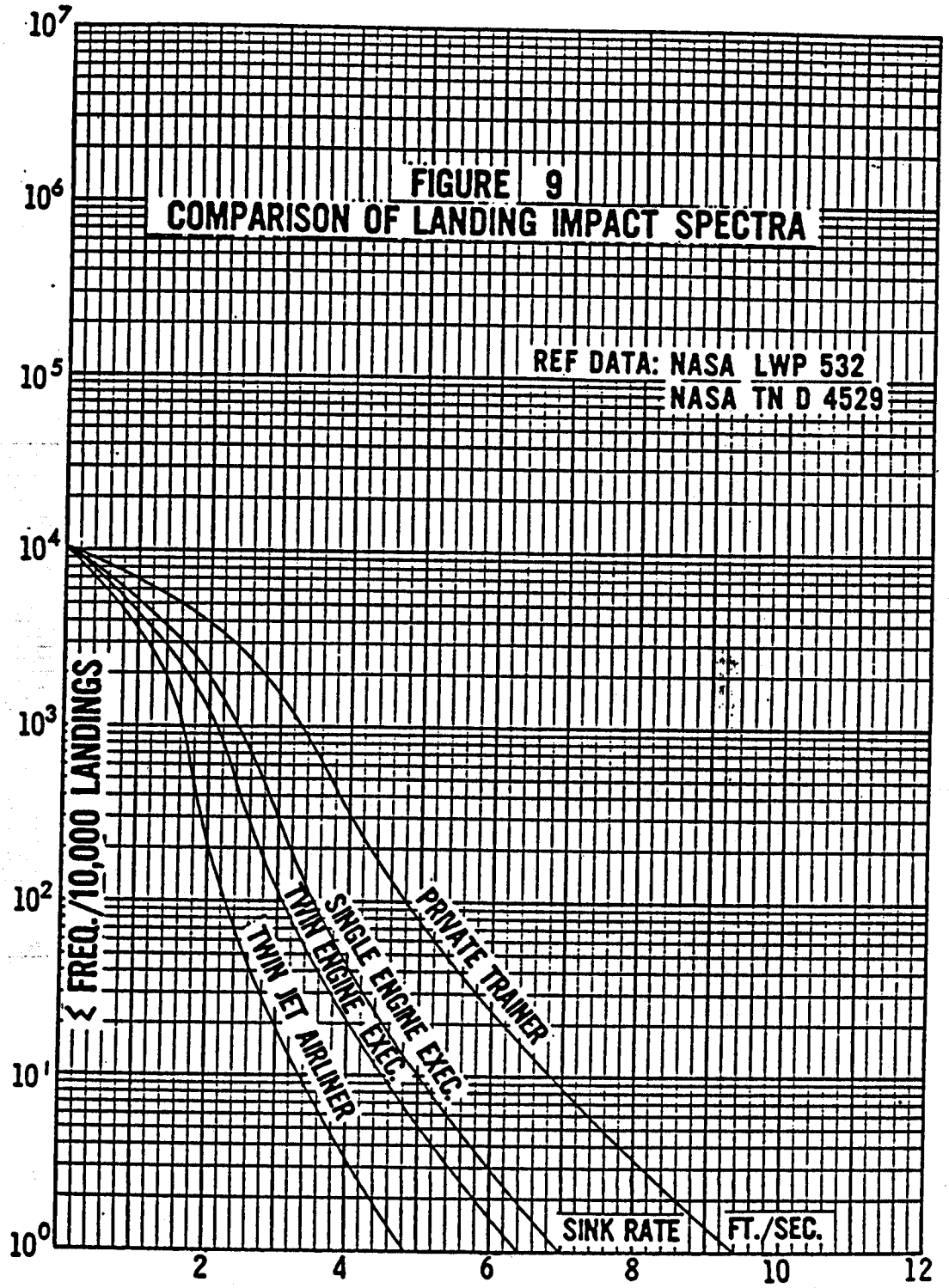


FIGURE 9
COMPARISON OF LANDING IMPACT SPECTRA
REF DATA: NASA LWP 532
NASA TN D 4529
FREQ./10,000 LANDINGS
10^7
10^6
10^5
10^4
10^3
10^2
10^1
10^0
PRIVATE TRAINER
SINGLE ENGINE EXEC.
TWIN ENGINE EXEC.
TWIN JET AIRLINER
SINK RATE FT./SEC.
2
4
6
8
10
12

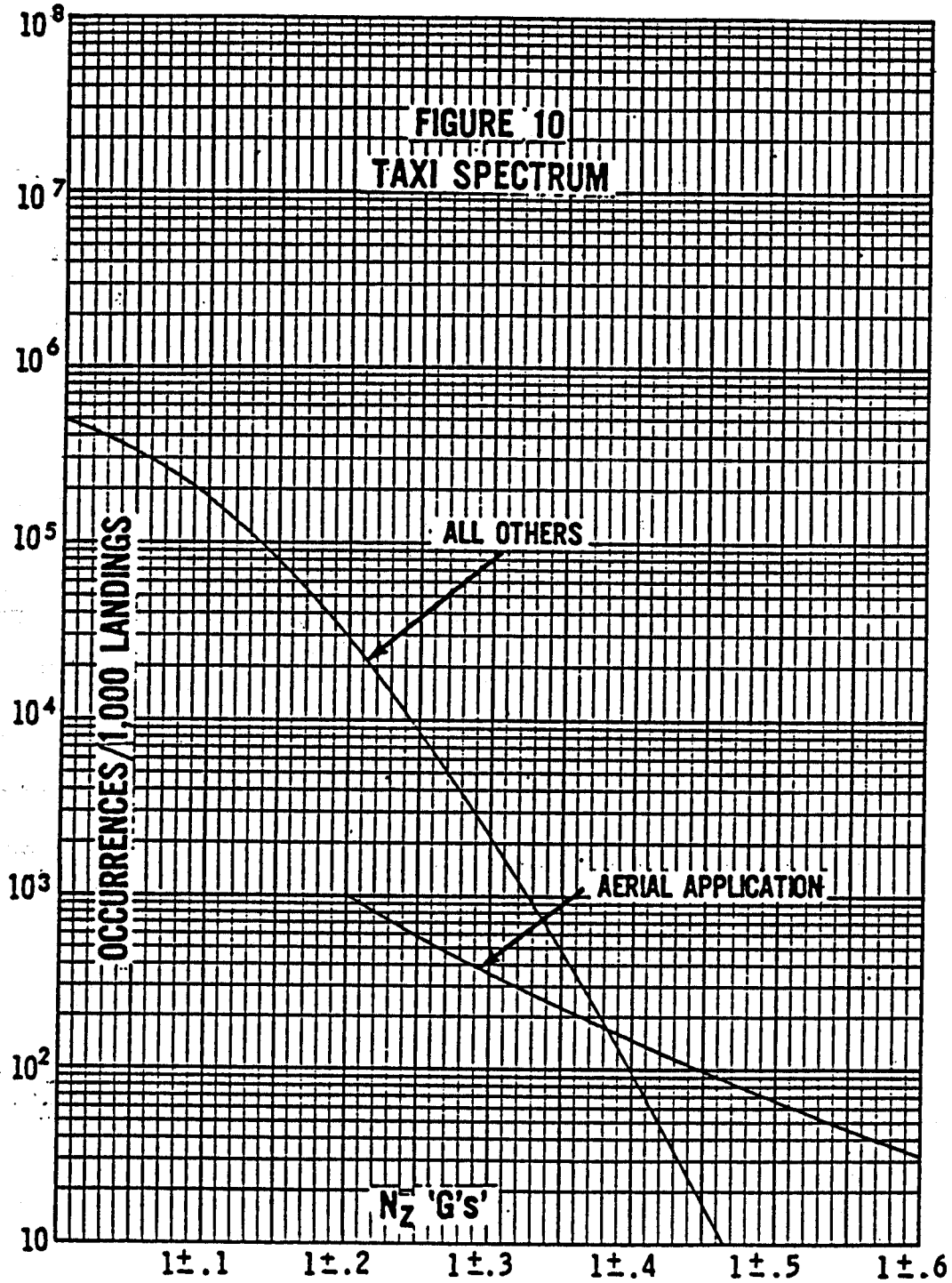


FIGURE 10
TAXI SPECTRUM
OCCURRENCES/1,000 LANDINGS
ALL OTHERS
AERIAL APPLICATION
N_Z 'G's'
10^8
10^7
10^6
10^5
10^4
10^3
10^2
10
1±.1
1±.2
1±.3
1±.4
1±.5
1±.6

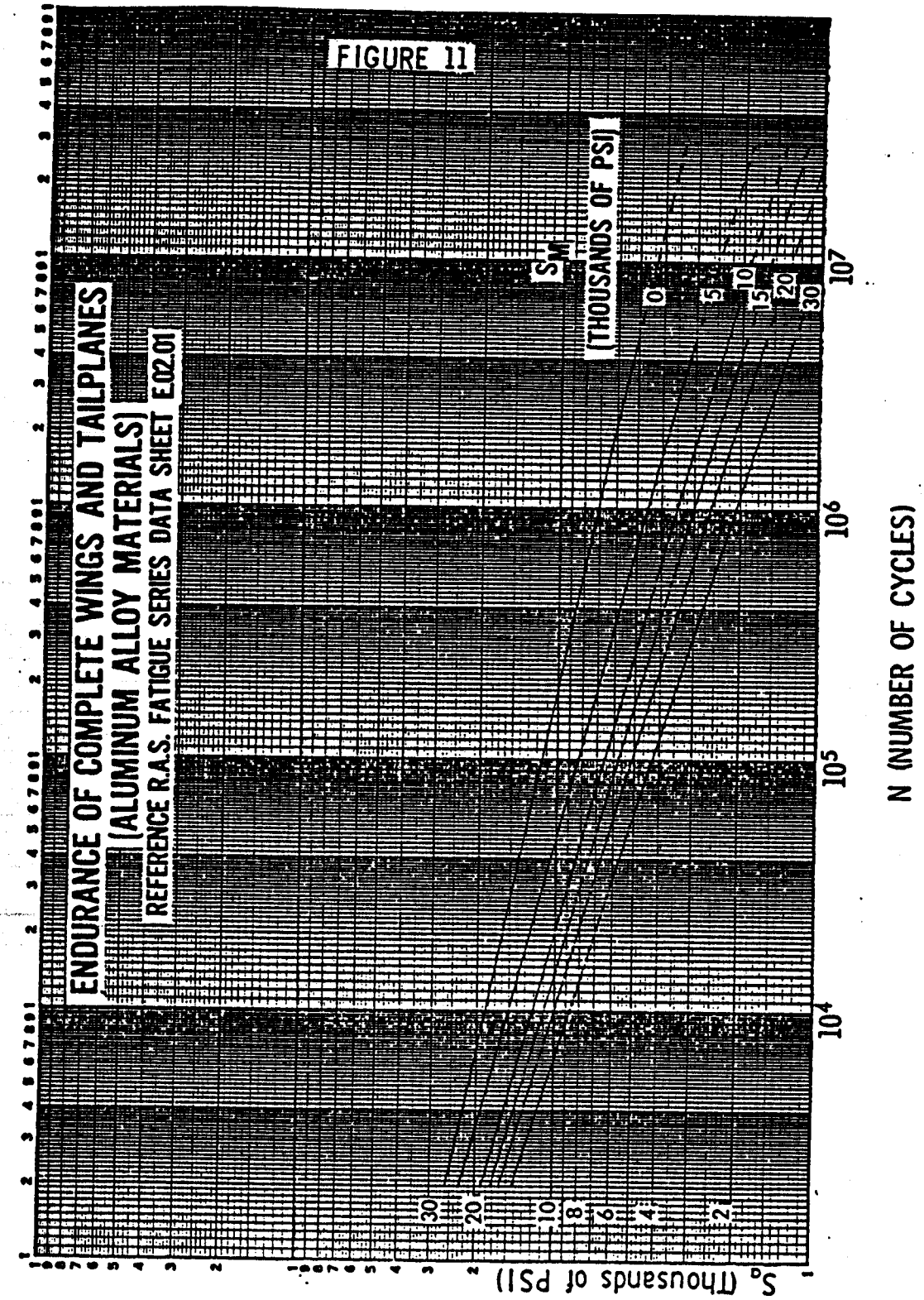

ENDURANCE OF COMPLETE WINGS AND TAILPLANES
(ALUMINUM ALLOY MATERIALS)
REFERENCE R.A.S. FATIGUE SERIES DATA SHEET E.02.01
S_M (THOUSANDS OF PSI)
0
5
10
15
20
30
S_a (Thousands of PSI)
30
20
10
8
6
4
2
N (NUMBER OF CYCLES)
10^4
10^5
10^6
10^7

FIGURE 11

FIGURE 12

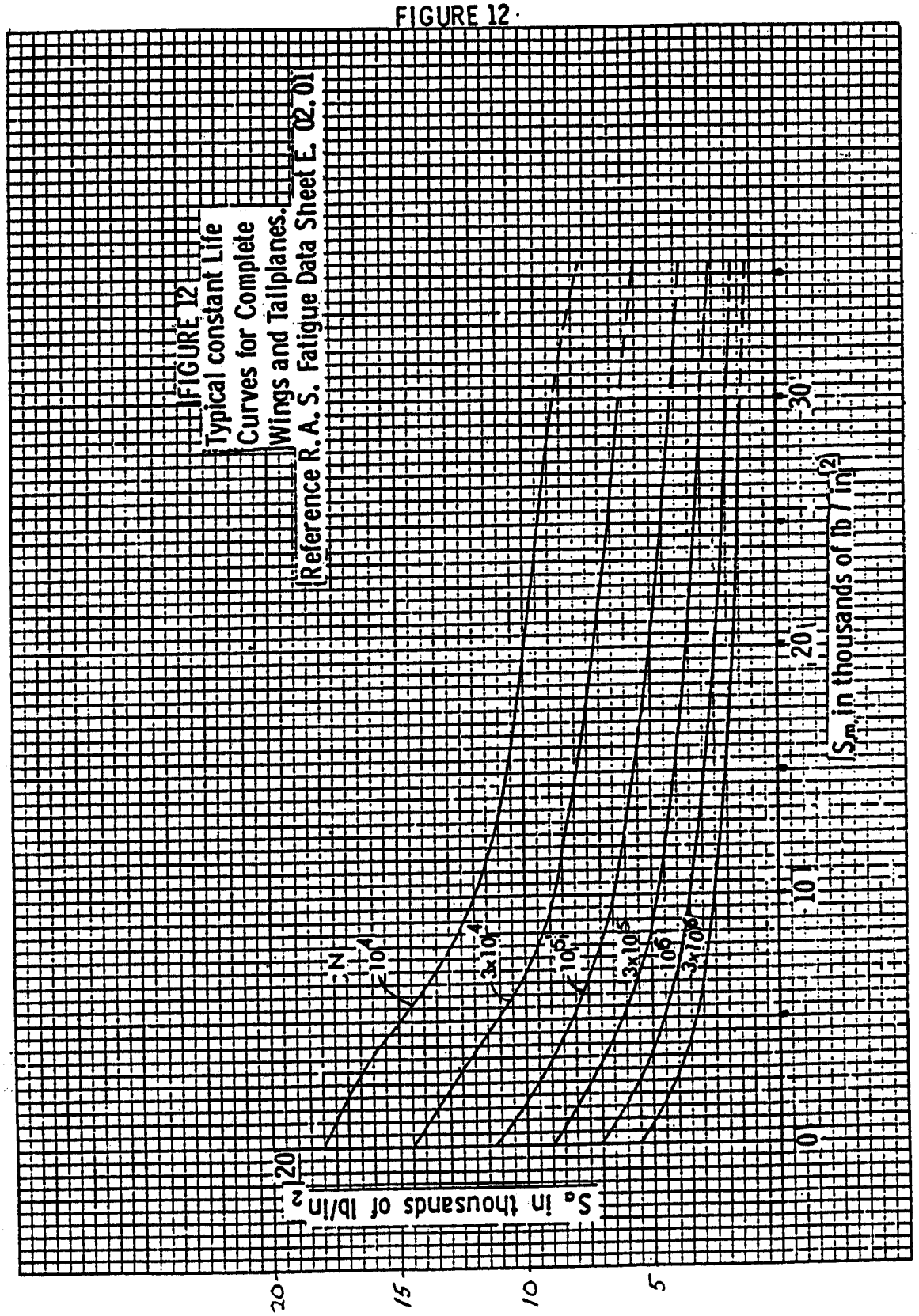


FIGURE 12
Typical constant Life
Curves for Complete
Wings and Tailplanes.
(Reference R.A.S. Fatigue Data Sheet E. 02.01)
N
10^4
3x10^4
10^5
3x10^5
10^6
3x10^6
S_a in thousands of lb/in^2
S_m in thousands of lb / in^2
20
15
10
5
0
10
20
30