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FATIGUE EVALUATION OF WING AND ASSOCIATED STRUCTURE ON SMALL AIRPLANES

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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 $\div$ 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. <u>MEAN FATIGUE LIFE DETERMINATION</u>. 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. <u>Full scale spectrum testing</u>. 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. <u>Component testing</u>. 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 eonstruction 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 matèrials 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. <u>ANALYSIS OF PRINCIPAL STRUCTURAL ELEMENTS</u>.

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.

a. "Metal Fatigue, Theory and Design" - Angel F. Madayag, Wiley and Sons, New York

b. "NASA General Aviation VGH Data" NASA SP 270 Volume 1 of NASA Safety Operating Problems Conference - 4-5 May 1971

c. "Analysis of VGH Records From Aircraft Engaged in Aerial Survey Over Mountainous Terrain," Canadian National Aeronautical Establishment Report LR-449, March 1966.

d. "Stress Concentration Design Factors" - R. E. Peterson, Wiley and Sons, New York.

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.

Appendix

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 × 3 FREQ./HR.	5 2.5 × 1A An	6 5500 × 5 Sa	7 Fig 11 $\times 10^6$ N	8 4 ÷ 7 $\times 10^{-6}$ /hr. n/N
.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 $\times 10^{-6}$	$\times 10^{-6}$ n/N
	+.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 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.

one g

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	N $\times 10^6$	n/N $\times 10^{-6}$
.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.	n FREQ./HR.	Sa	N_1 x 10^6	n/N_1 x 10^{-6}	.6Sa	N_2 x 10^6	n/N_2 x 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 x 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	$N \times 10^6$	$n/N \times 10^{-6}$
4375	5275	1.54	.75	2.0533

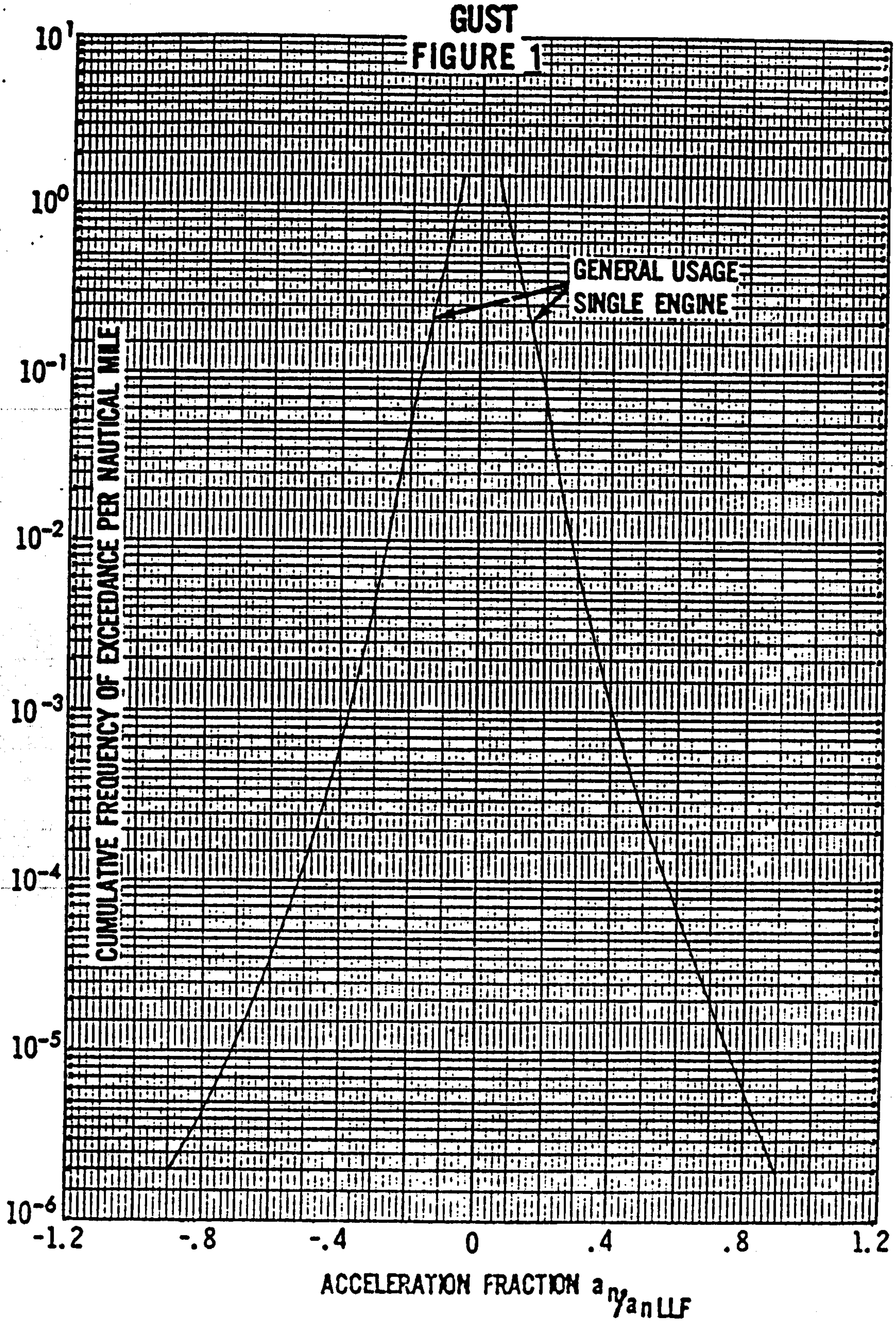
DAMAGE = 2.0533×10^{-6}/HR.

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
10^1
10^0
10^-1
10^-2
10^-3
10^-4
10^-5
10^-6
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
GENERAL USAGE
SINGLE ENGINE
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION a n/a n LLF

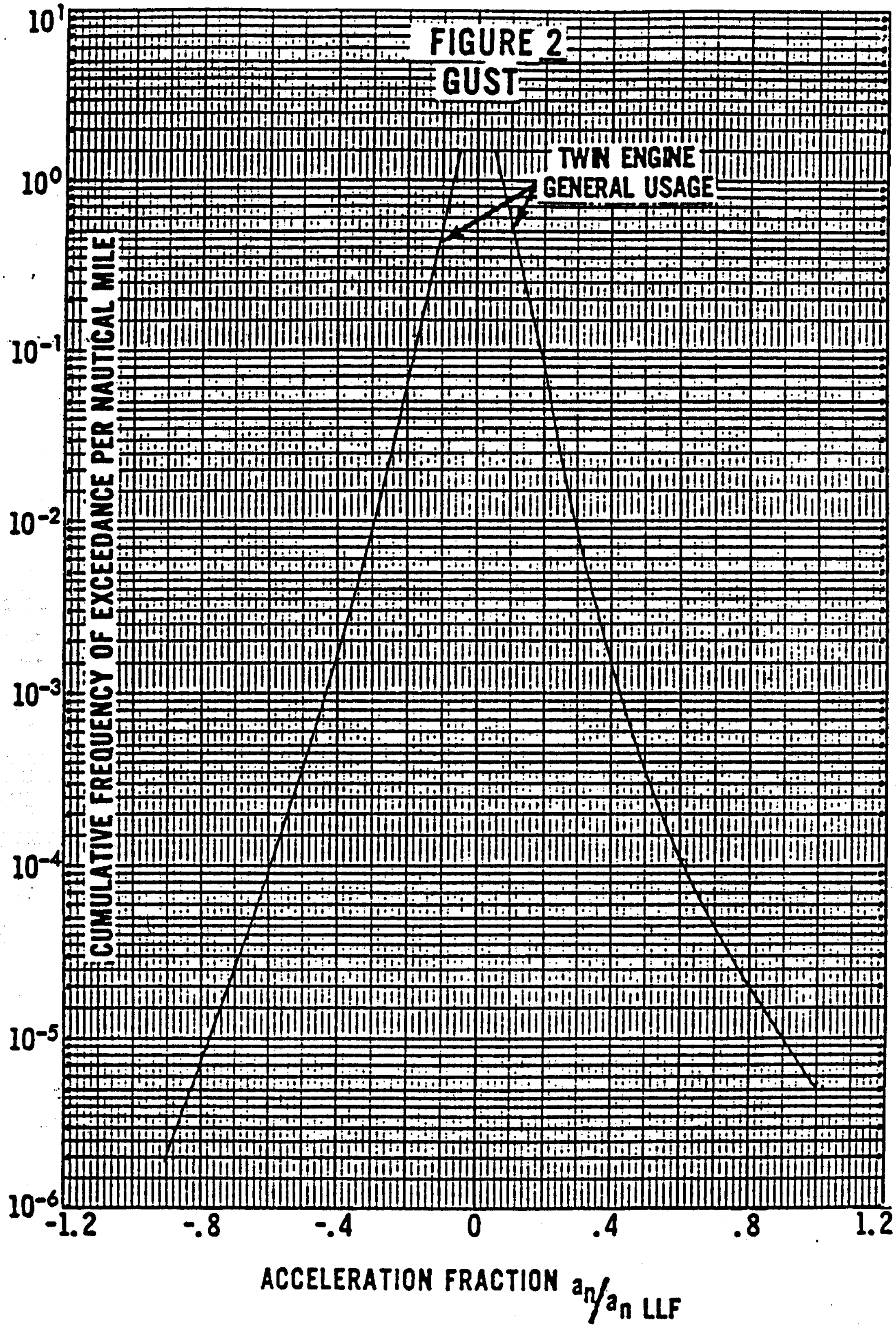
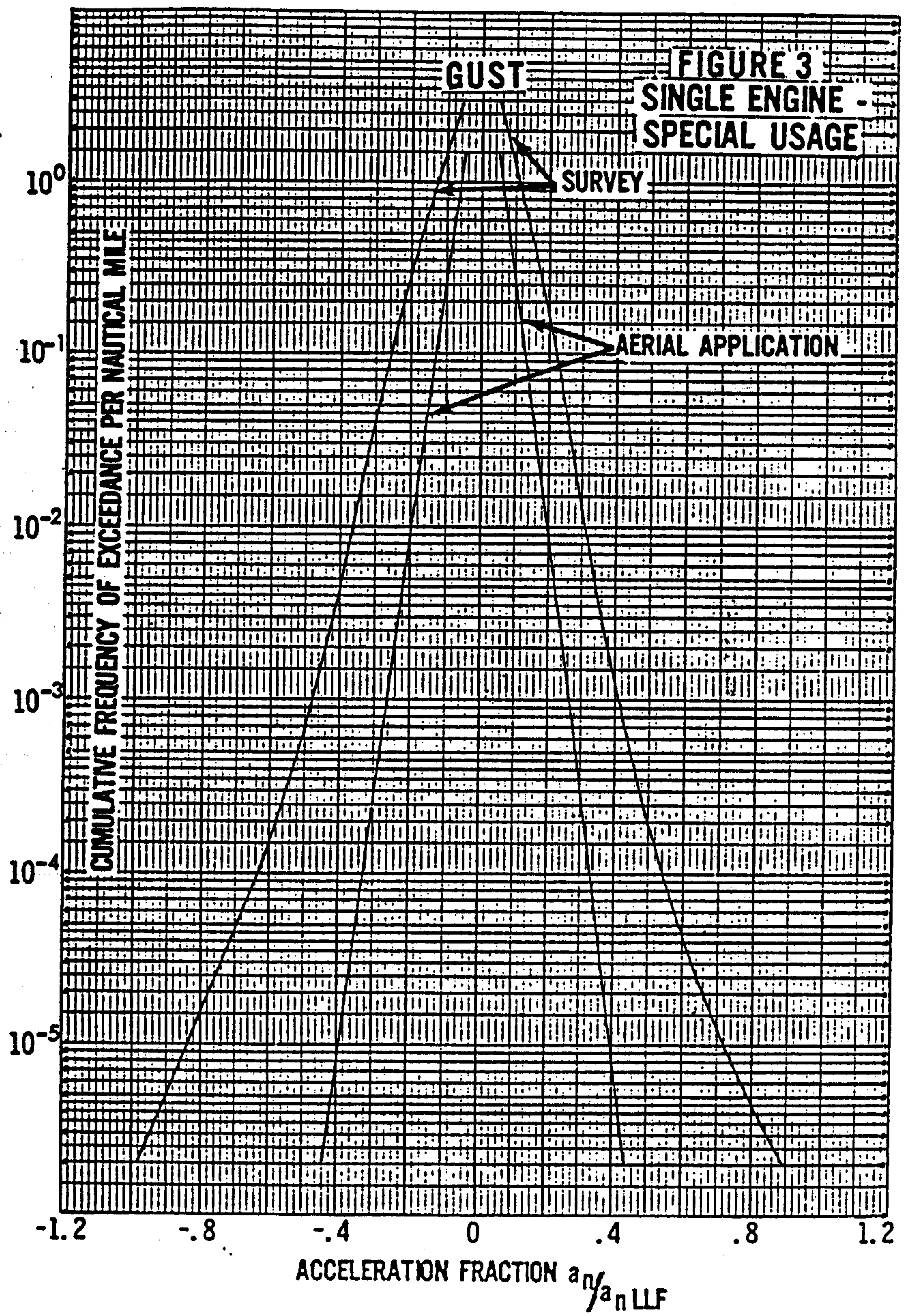


FIGURE 2
GUST
TWIN ENGINE
GENERAL USAGE
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10^1
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
10^{-6}
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION $a_n/a_{n\ LLF}$


GUST
FIGURE 3
SINGLE ENGINE -
SPECIAL USAGE
SURVEY
AERIAL APPLICATION
10⁰
10⁻¹
10⁻²
10⁻³
10⁻⁴
10⁻⁵
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION an/an LLF

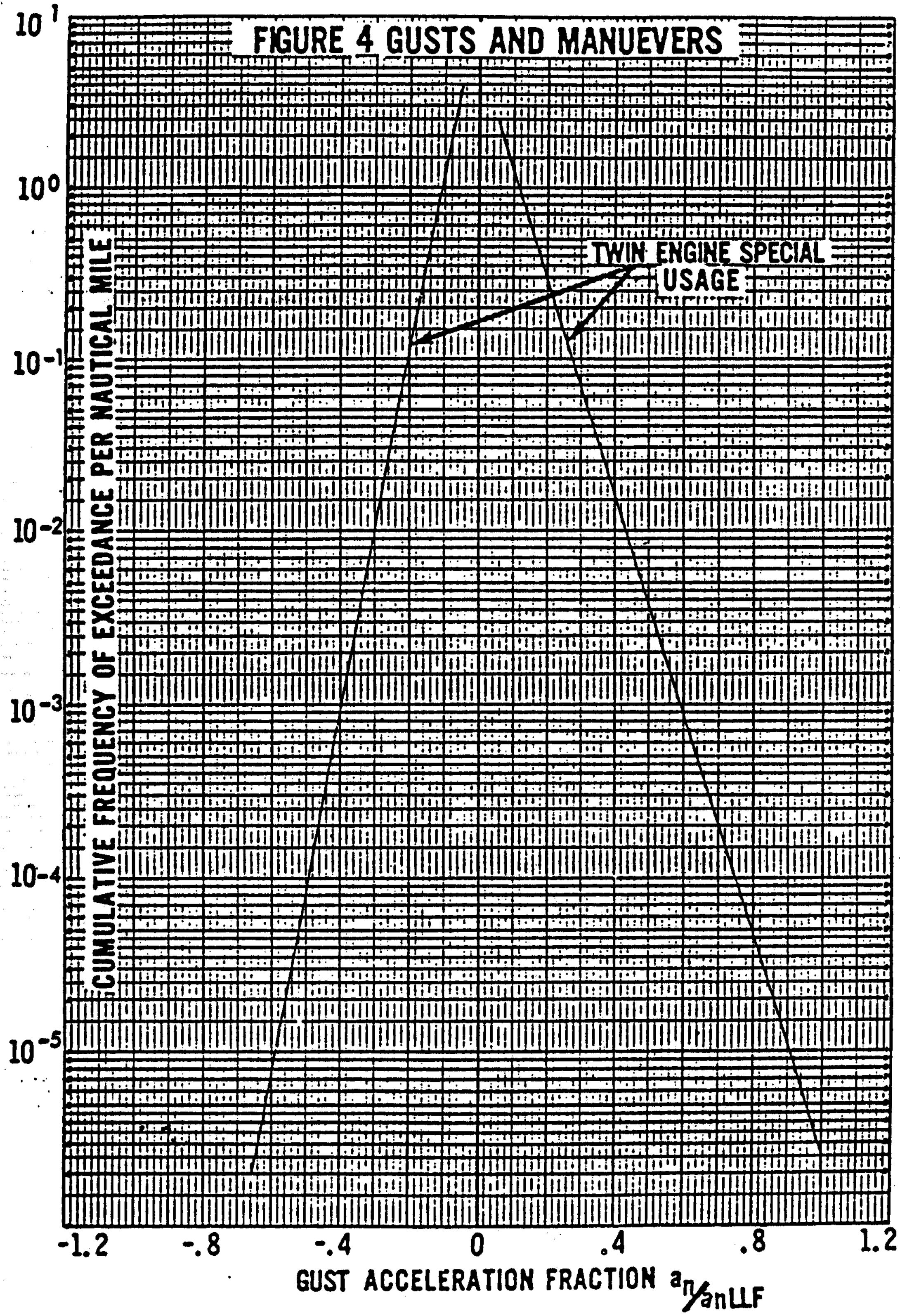
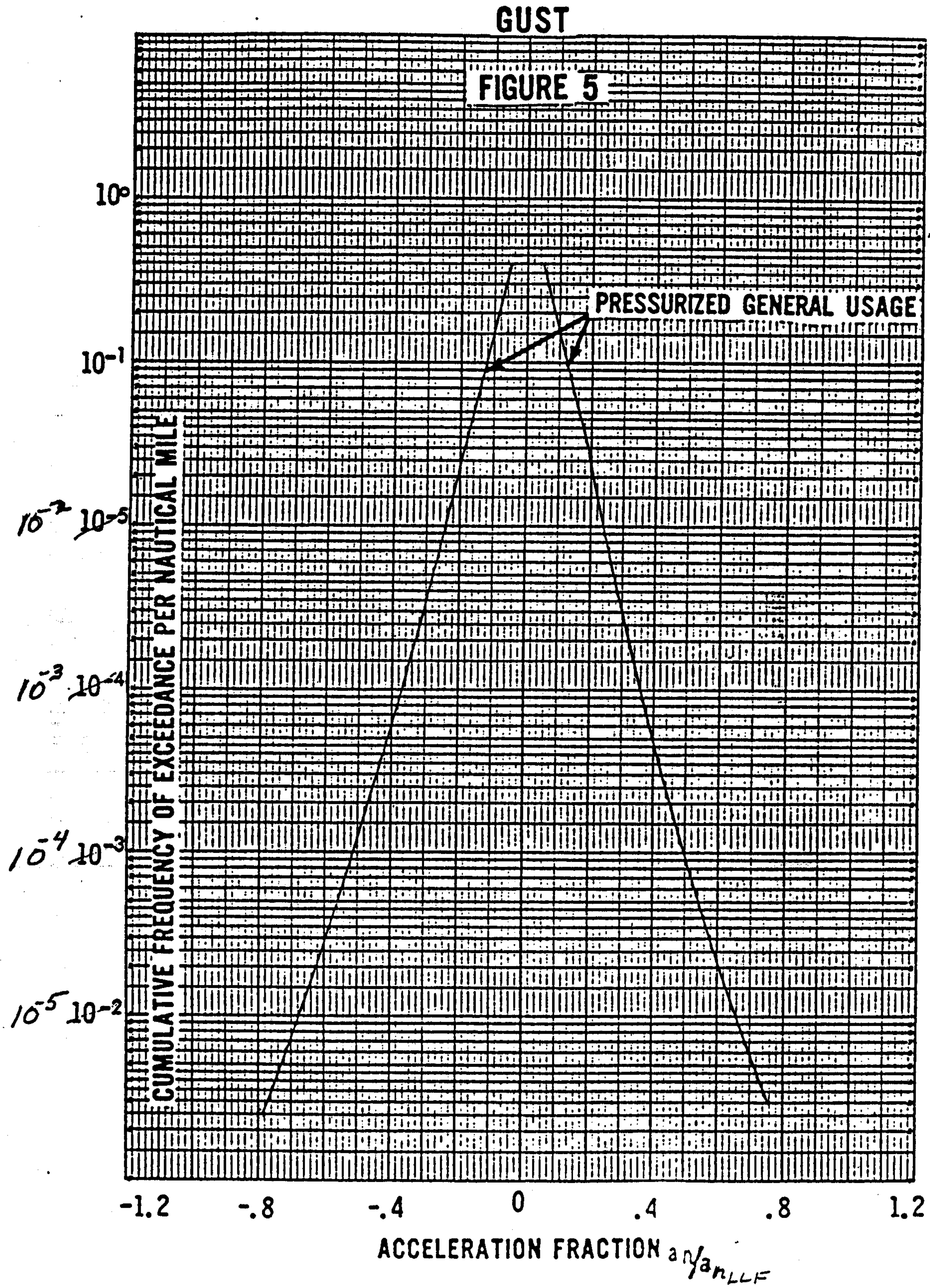


FIGURE 4 GUSTS AND MANUEVERS
TWIN ENGINE SPECIAL
USAGE
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10 1
10 0
10-1
10-2
10-3
10-4
10-5
-1.2
-.8
-.4
0
.4
.8
1.2
GUST ACCELERATION FRACTION an/anLLF


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

MANEUVERS

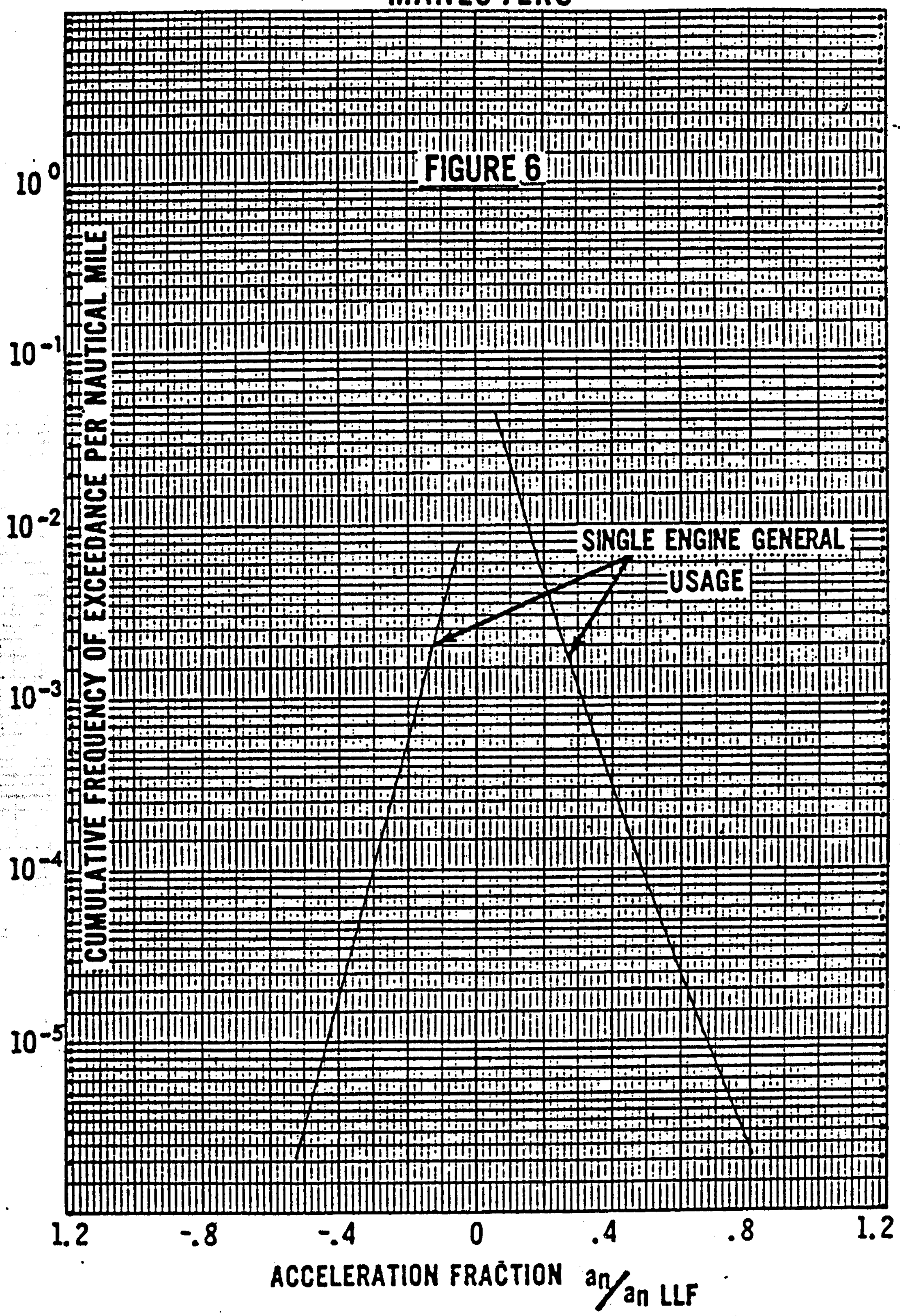


FIGURE 6
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
SINGLE ENGINE GENERAL USAGE
1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION a_n/a_n LLF

MANEUVERS

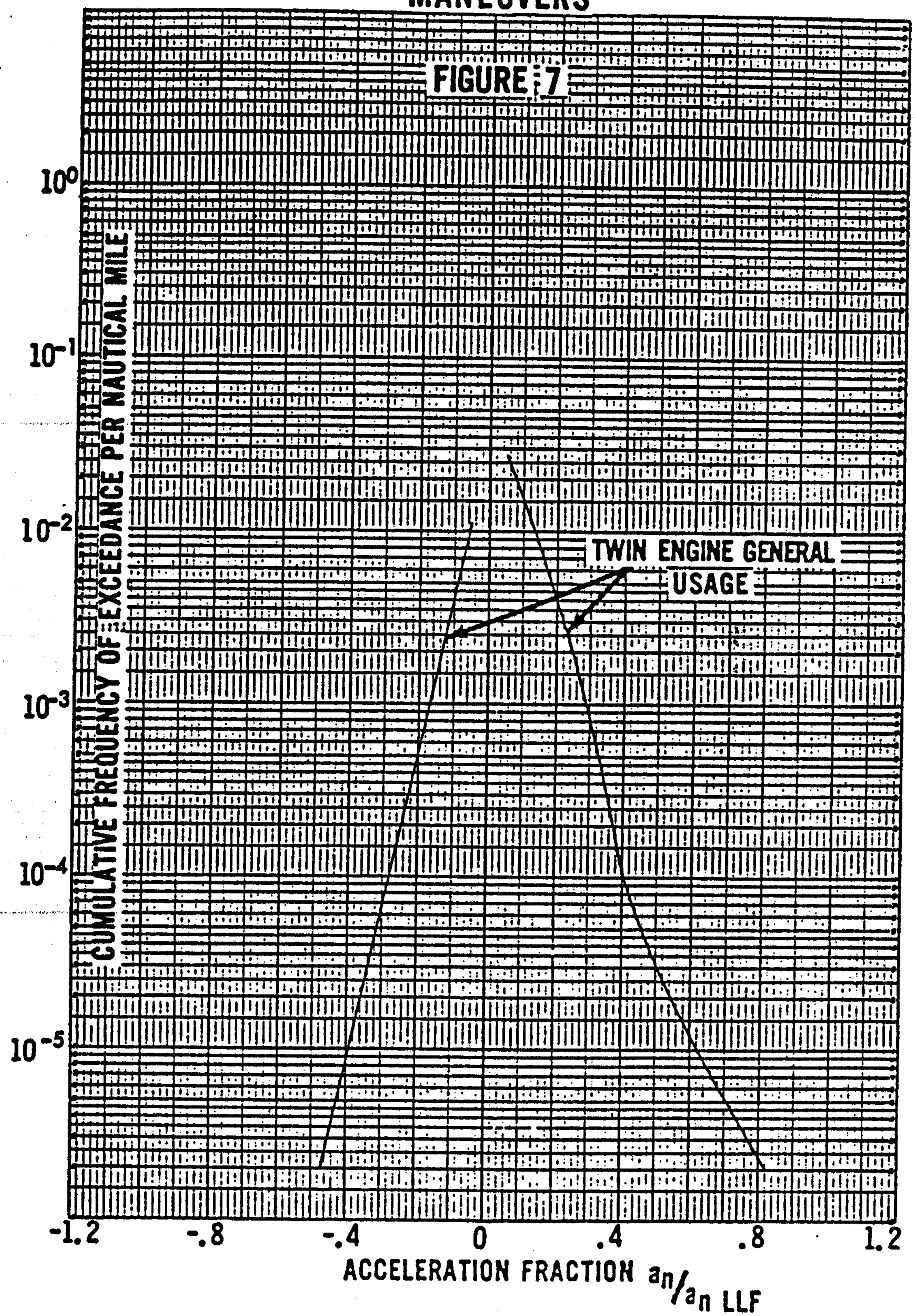
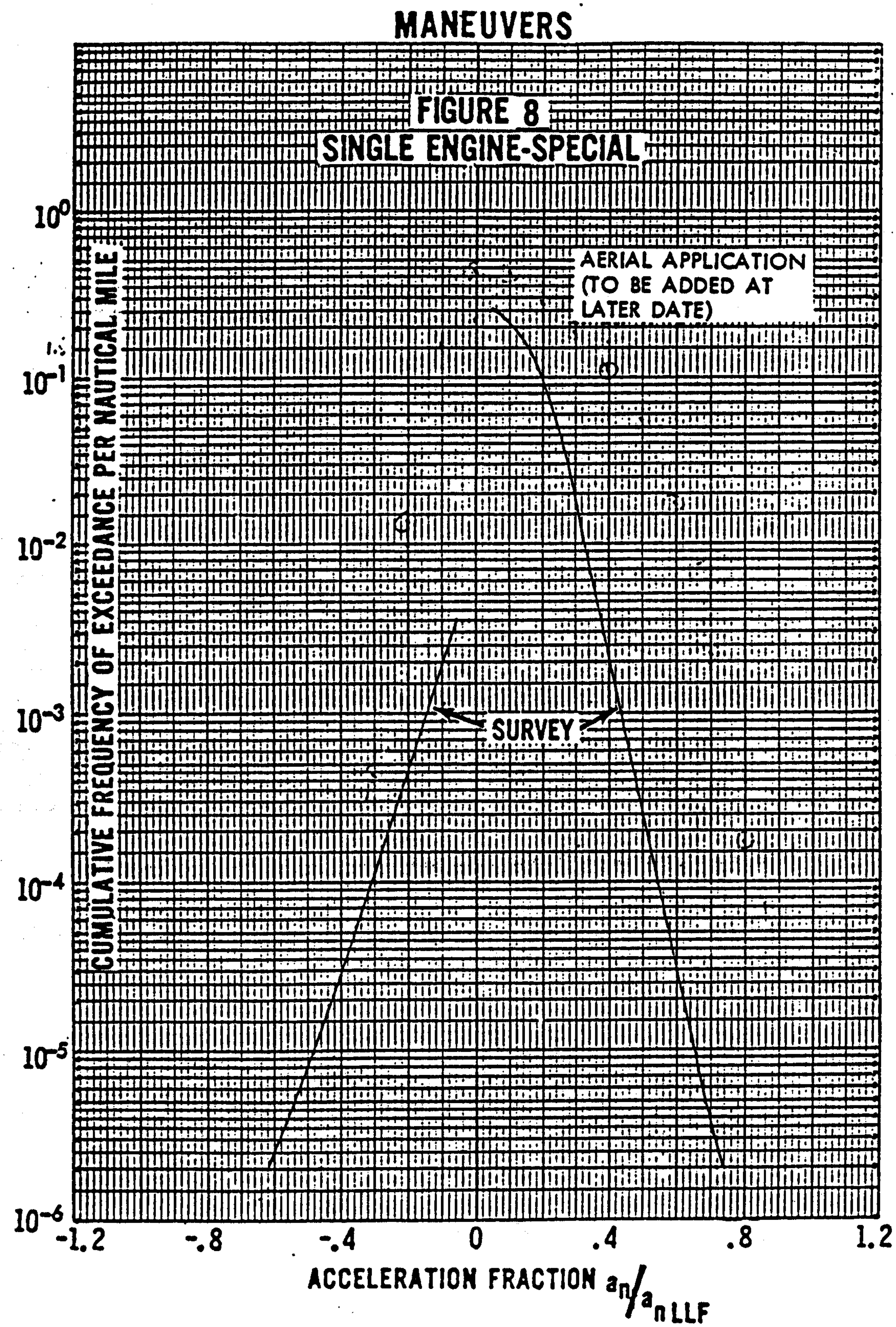


FIGURE 7
CUMULATIVE FREQUENCY OF EXCEEDANCE PER NAUTICAL MILE
10^0
10^-1
10^-2
10^-3
10^-4
10^-5
TWIN ENGINE GENERAL USAGE
-1.2
-.8
-.4
0
.4
.8
1.2
ACCELERATION FRACTION an/an LLF


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 $a_n/a_{n\,LLF}$

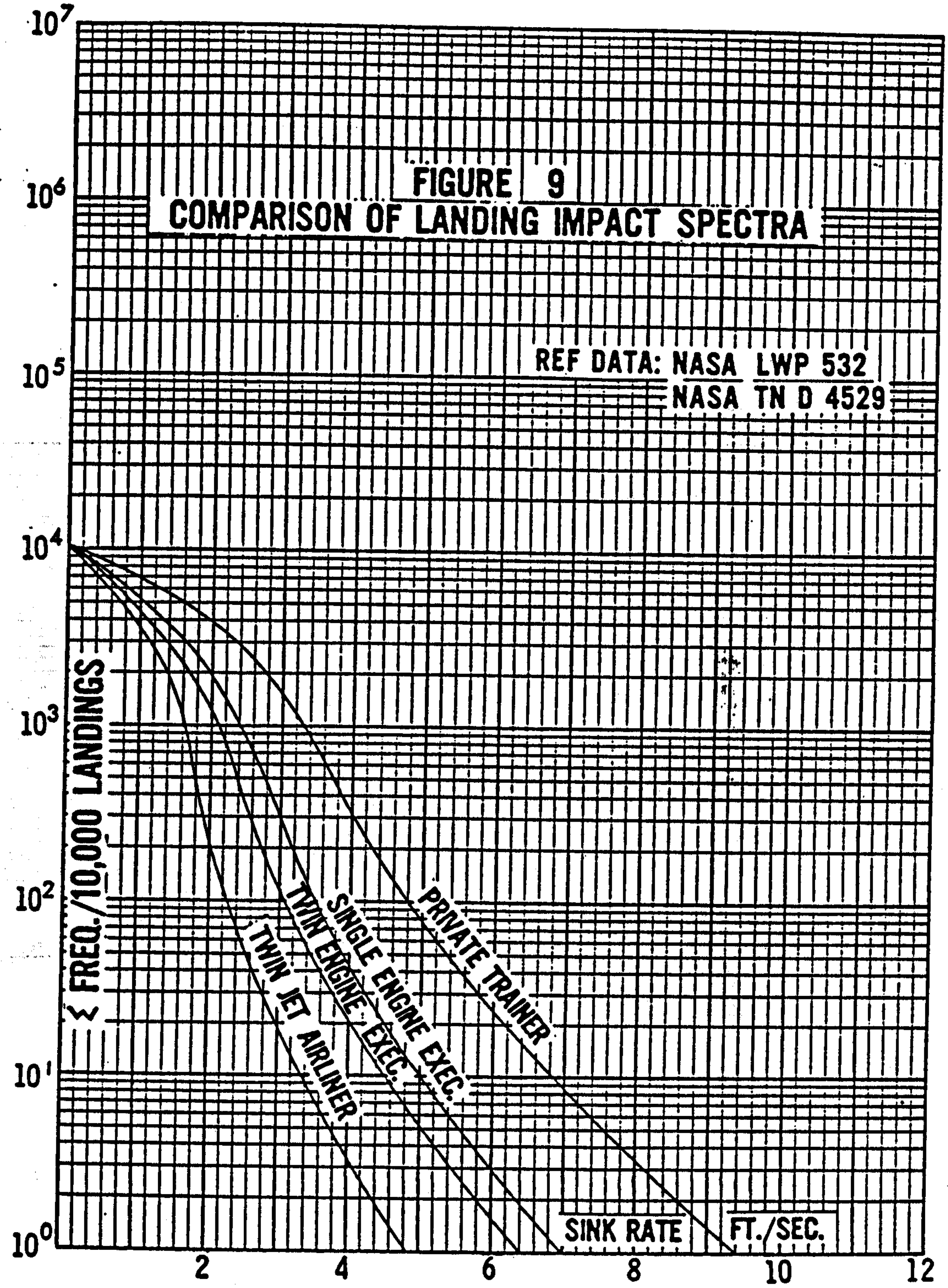


FIGURE 9
COMPARISON OF LANDING IMPACT SPECTRA
REF DATA: NASA LWP 532
NASA TN D 4529
Σ FREQ./10,000 LANDINGS
10^7
10^6
10^5
10^4
10^3
10^2
10^1
10^0
PRIVATE TRAINER
SINGLE ENGINE EXEC.
TWIN ENGINE EXEC.
TWIN JET AIRLINER
SINK RATE FT./SEC.
2
4
6
8
10
12

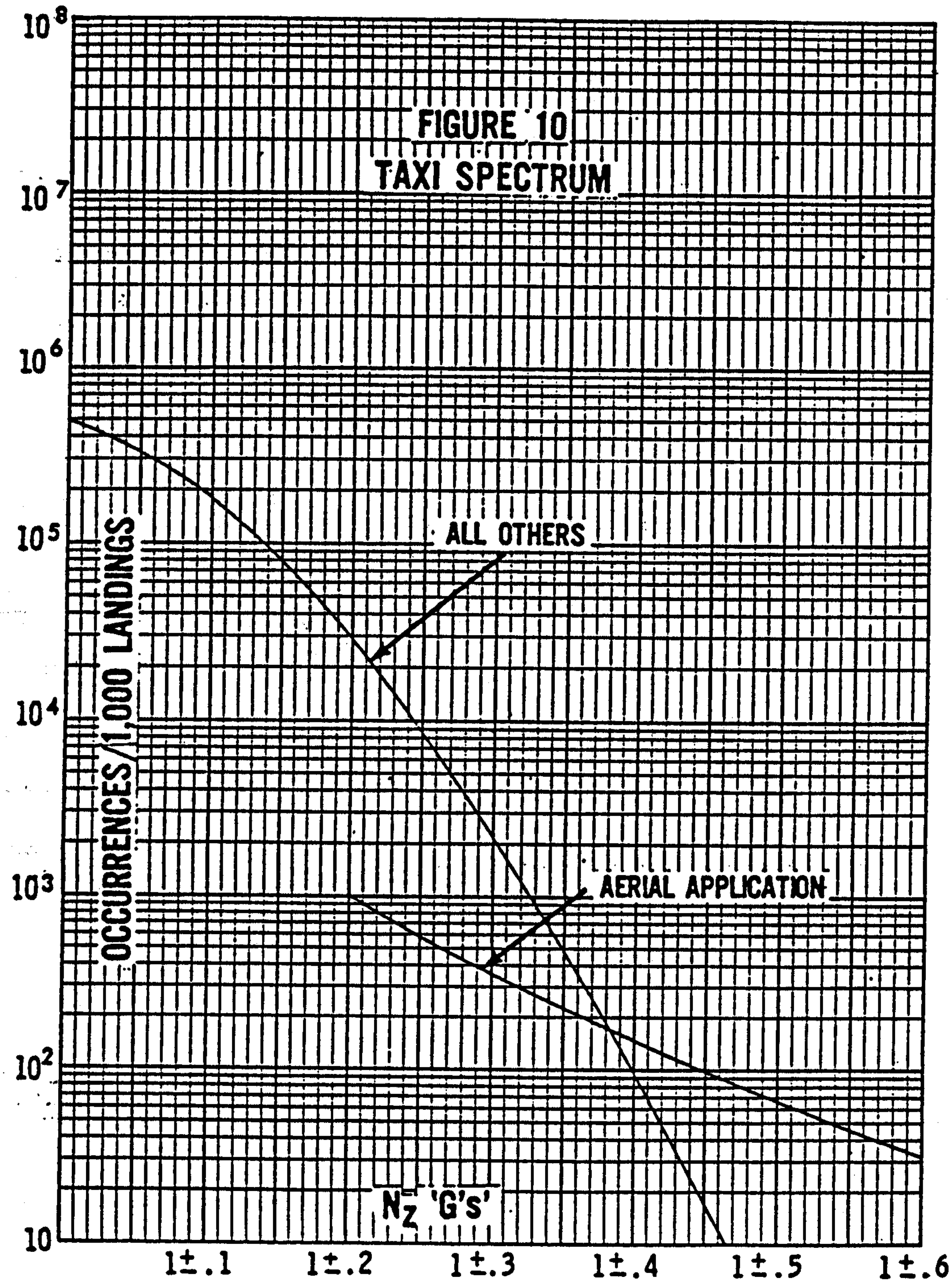


FIGURE 10
TAXI SPECTRUM
OCCURRENCES/1,000 LANDINGS
ALL OTHERS
AERIAL APPLICATION
N_Z 'G's'
10^8
10^7
10^6
10^5
10^4
10^3
10^2
10
1±.1
1±.2
1±.3
1±.4
1±.5
1±.6

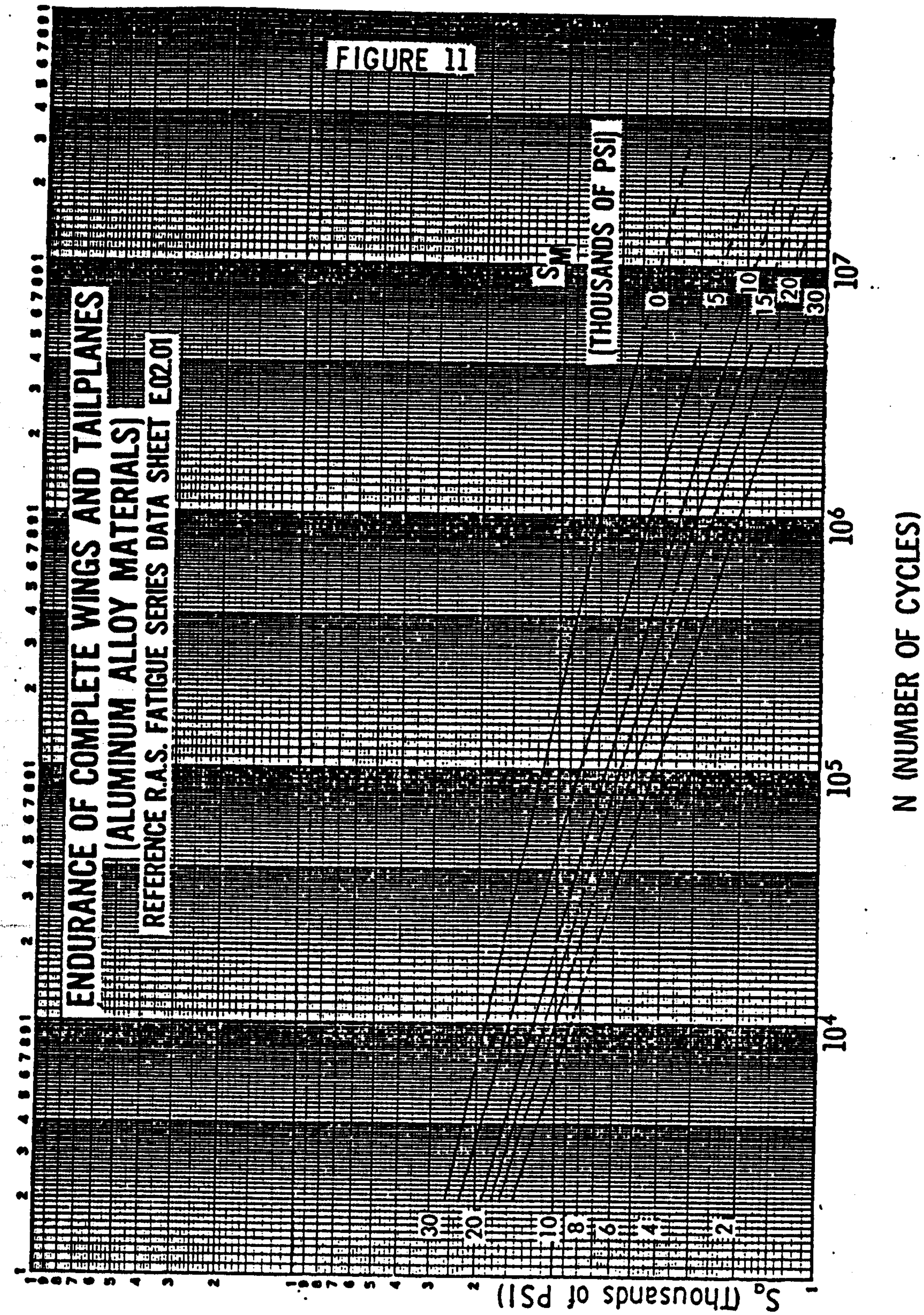


FIGURE 11
ENDURANCE OF COMPLETE WINGS AND TAILPLANES
(ALUMINUM ALLOY MATERIALS)
REFERENCE R.A.S. FATIGUE SERIES DATA SHEET E.02.01
S_M
(THOUSANDS OF PSI)
0
5
10
15
20
30
30
20
10
8
6
4
2
10^4
10^5
10^6
10^7
N (NUMBER OF CYCLES)
S_a (Thousands of PSI)

29360

FIGURE 12

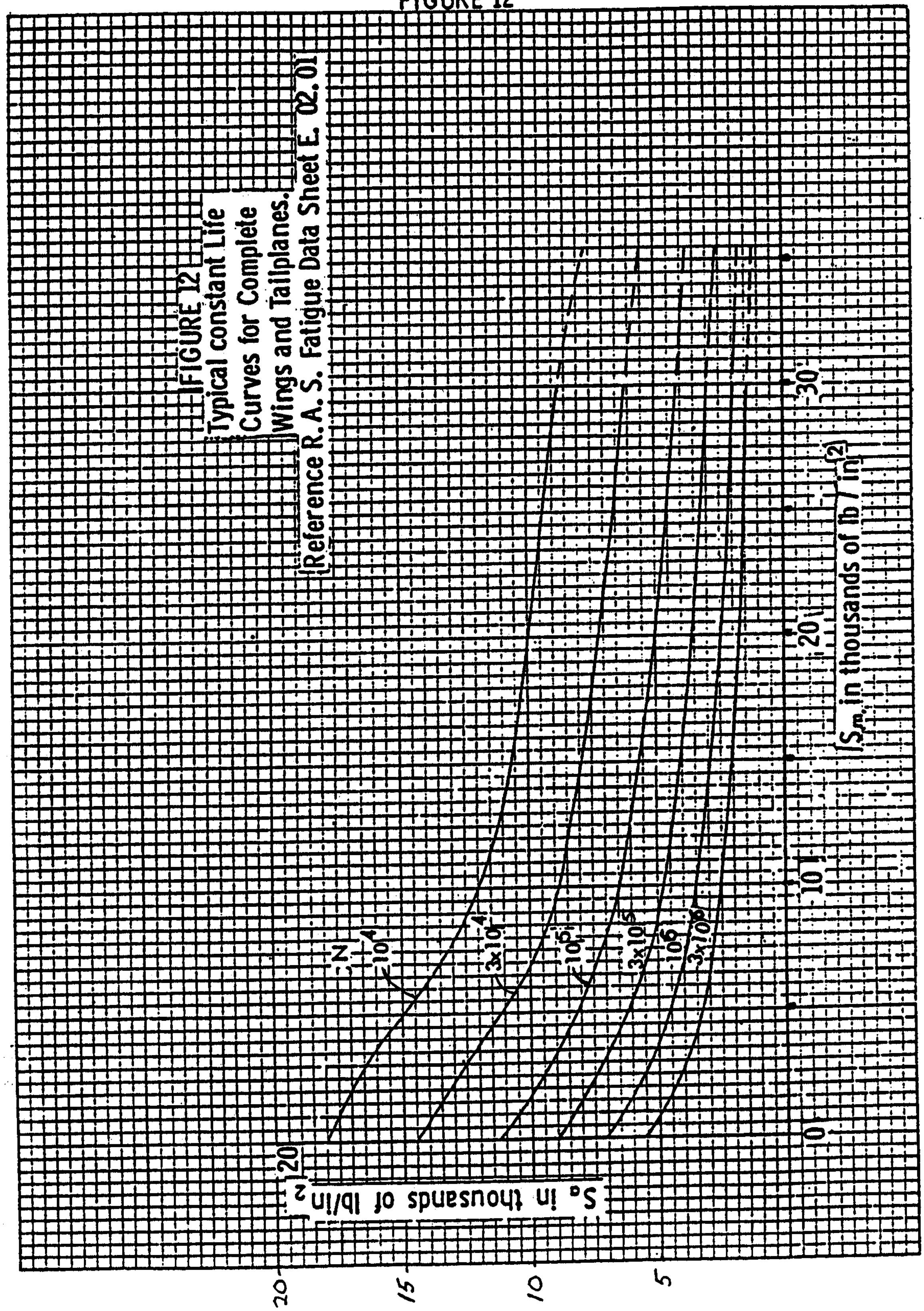


FIGURE 12
Typical constant Life
Curves for Complete
Wings and Tailplanes.
Reference R. A. S. Fatigue Data Sheet E. 02.01
N
10^4
3x10^4
10^5
3x10^5
10^6
3x10^6
0
10
20
30
Sm in thousands of lb / in2
Sa in thousands of lb/in2
20
15
10
5

SAMPLE CALCULATIONS SHOWING USE OF METHOD IN AFS-120-73-2 FATIGUE REPORT.

by Hank Nauert (RETIRED) & Don CAMPBELL (RETIRED)

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DRAFT

GW 4300 Lb

V_c 165 Kts

V_a 132 Kts

± gust limit load factor ± 2.155 ~ a_{mLLF} (gust)

± maneuver limit load factor $^{+2.8}_{-2.52}$ ~ a_{mLLF} (maneuve

$(0 \rightarrow 1g)$ stress 7410 psi

ground stress -4520 psi

$S_{MAX_{Landing}} = (2/3) \times (0 \rightarrow 1g)$ Stress 4943 psi

assume 2/3 wing lift
FAR 23.473(e)

DRAFT

REV. C
7-20-93

EXAMPLE PROBLEM FOR FATIGUE CALCULATION
IN REPORT AFS-120-73-2

REVISIONS

Revision	Date	Page	Description
Rev. A	2-15-90	11	Definition of static ground stress (Sm) was modified.
Rev. B	3-91	16	Item 4 changed
		17	Landing impact curve replotted lower. Word "total" removed from "TAXI" curve.
Rev. C	7-20-93	2,4, 18	Gust and maneuver calculations revised.

Gust (T.E. Gen. Usage), Nonpressurized; $V_c = 165$ kts ; $.9V_c = 148.5$ kts

Flight Stress/g = 7410 psi (@ critical component, wing strut forging)

Gust $a_{n_{LLF}} = \pm 2.155$g (from a 1-g condition)

(A)	(1)	(2)	(3)	(4)	(5)		(6)		(7)	(8)	(9)	(10)
(−) $a_n / a_{n_{LLF}}$	(+) $a_n / a_{n_{LLF}}$	Cum. Freq. per n.m. Fig 2 of AFS-120-73	Freq per n.m.	n ~ Freq/Hr. (3) × .9V_c	a_n −	a_n +	Δ Max Stress −	Δ Max Stress +	S_m psi	S_a psi	N × 10^{-6}	n/N × 10^6 (4)/(9)
−.12	.10 .13	.57	.39	57.9	.259	.28	1920	2070	7485	1995	14.0	4.13
−.18	.16 .19	.18	.128	19	.388	.409	2870	3030	7490	2950	3.1	6.13
−.25	.22 .25	.052	.039	5.79	.539	.539	4000	4000	7410	4000	1.05	5.52
−.32	.28 .31	.013	.0087	1.292	.690	.667	5980	4940	7325	5025	.50	2.59
−.37	.34 .37	.0043	.0028	.416	.797	.797	5830	5830	7410	5830	.20	2.08
−.43	.40 .43	.0015	.00089	.1322	.927	.927	6870	6870	7410	6870	.135	.98
−.49	.46 .49	.00061	.00043	.0638	1.055	1.055	7810	7810	7410	7810	.086	.74
−.55	.52 .55	.00028	.00015	.0253	1.185	1.185	8770	8770	7410	8770	.055	.40
−.59	.58 .61	.00013	.000055	.00877	1.27	1.315	9400	9740	7570	9580	.036	.23
−.63	.64 .67	.000075	.000027	.00402	1.36	1.44	10070	10650	7700	10360	.028	.14
−.67	.70 .73	.000048	.000021	.00312	1.44	1.57	10660	11630	7890	11140	.019	.16
−.72	.76 .79	.000027	.000010	.00148	1.55	1.70	11490	12600	7970	12040	.014	.11
−.75	.82 .85	.000017	.000017	.00253	1.61	1.83	11920	13550	8225	12735	.013	.19

$\Sigma = 23.40 \times 10^{-6}$ Per Hour

Fatigue Damage due to Gust = 2.34×10^{-2} per 1000 Hours

Rev C
7/20/93

COLUMN — INSTRUCTIONS PERTINENT TO PAGE 2

(1). Arbitrarily chosen increments of $a_n/a_{n_{LLF}}$; a_n = incremental load factor due to a gust as related to 1-g flt. condition. $a_{n_{LLF}}$ = incremental design limit load factor = ± 2.155. In this example, (+) $a_n/a_{n_{LLF}}$ = .10 was chosen to start and increased by increments of .06. The "in-between" values denote averages between successive (+) $a_n/a_{n_{LLF}}$ value

(2) Read from Figure 2 of AFS-120-73-2 for "Twin Eng. General" at the various values of $a_n/a_{n_{LLF}}$ = "Cum. Occurences Per N. Mile

(A) The neg. values represent neg. $a_n/a_{n_{LLF}}$ that will result in the same numerical value of "Cum. Occur./n.m." as the ave. pos. value of $a_n/a_{n_{LLF}}$ (ie., the "in-between values of (+) $a_n/a_{n_{LLF}}$.

(3) Freq. increment ; ie., col (2) is a cumulative value and we need incremental values. (3) = difference between successive values in col (2).

(4) No. of gust cycles accumulated per hour = col (3) × Velocity where the velocity is $.9V_c$ = 148.5 knots. Hence (4) = (3) × 148.5.

(5) $a_{n_{LLF}}$ and $a_n/a_{n_{LLF}}$ values have already been selected. Hence $a_n = (a_n/a_{n_{LLF}}) \times a_{n_{LLF}}$. In this example, col. (5) = (A) × 2.155 or (5) = (1) × 2.155 as appropriate.

(6) Δ Max Stress = (5) × (Stress/g) = (5) × 7410.

(7) $$S_m = \frac{[(0\text{-}1g)\text{Stress} + (\text{pos.}\Delta\text{ max stress})] + [(0\text{-}1g)\text{stress} + (\text{neg.}\Delta\text{ max stress})^*]}{2}$$

= Mean Stress

* Added algebraically (see example below).

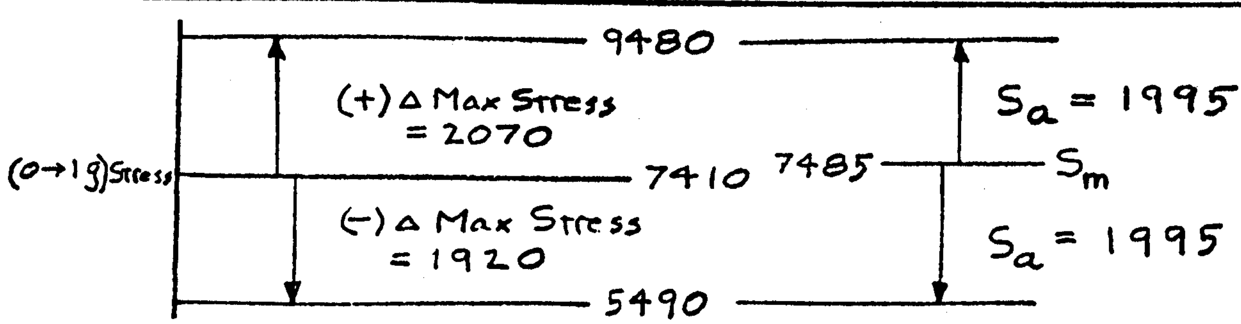
(8) S_a = [(0-1g) stress) + (pos. Δ max stress) − S_m]

= Alternating Stress (see example below)

(9) Endurance cycles, N, read from the Raithby S-N data. (Fig. 11 of AFS-120-73-2)

(10) Damage per hour = (4)/(9)

TOTAL DAMAGE DUE TO GUST IS SUMMATION OF COL. (10)



Example for (+) $a_n/a_{n_{LLF}}$ = .13

Maneuver (T.E. Gen. Usage); V_c = 165 kts ; .9V_c = 148.5 kts

Flight Stress/g = 7410 psi (@ wing strut forging)

Maneuver $a_{n_{LLF}}$ = (+) 2.8 g , (−) 2.52 g (from a 1-g condition)

A	1	2	3	4	5		6		7	8	9	10
(−) $a_n/a_{n_{LLF}}$	(+) $a_n/a_{n_{LLF}}$	Cum. Freq. per n.m. Fig 7 of AFS-120-73	Freq Per n.m.	n ~ Freq/Hr 3 × .9V_c	a_n −	a_n +	Δ Max Stress −	Δ Max Stress +	S_m psi	S_a psi	N ×10^{-6}	n/N ×10^6 4/9
−.05	.10 .13	.016	.0088	1.31	.126	.364	930	2690	8290	1810	17.0	.077
−.10	.16 .19	.0072	.0040	.594	.252	.532	1870	3940	8450	2910	2.9	.204
−.14	.22 .25	.0032	.0019	.282	.353	.700	2610	5180	8700	3900	.85	.322
−.19	.28 .31	.0013	.0009	.134	.478	.868	3540	6420	8850	4980	.35	.383
−.25	.34 .37	.0004	.00028	.0417	.630	1.036	4670	7680	8920	6180	.15	.278
−.29	.40 .43	.00012	.000072	.0107	.731	1.205	5410	8930	9170	7170	.082	.131
−.33	.46 .49	.000048	.000021	.00312	.832	1.370	6160	10150	9410	8160	.048	.065
−.36	.52 .55	.000027	.000011	.00163	.907	1.540	6720	11410	9760	9070	.030	.054
−.38	.58 .61	.000016	.0000076	.00113	.958	1.71	7100	12670	10200	9890	.021	.054
−.41	.64 .67	.0000084	.0000030	.000446	1.032	1.88	7650	13930	10550	10790	.0145	.031
−.43	.70 .73	.0000054	.0000018	.000267	1.083	2.04	8020	15120	10960	11570	.0113	.023
−.46	.76 .79	.0000036	.0000016	.000238	1.158	2.21	8580	16380	11310	12480	.0080	.030
−.48	.82 .85	.0000020	.0000020	.000297	1.210	2.38	8960	17640	11750	13300	.0064	.046

Σ = 1.698 × 10^{-6} Per Hour

Fatigue Damage due to Maneuvering = .170 × 10^{-2} per 1000 Hours

REV. C
7-20-93

MANEUVER

COLUMN	INSTRUCTIONS PERTINENT TO PAGE 4
1	Arbitrarily chosen values of (+) $a_n/a_{n_{LLF}}$. The first value is usually chosen very small so that the value of N (ie. cycles to failure) $\cong$ the endurance limit (ie. 30 million cycles).
2	Obtained from maneuver spectrum for "T.E. Gen. Usage", Figure 7 of AFS-120-73-2.
3	Difference between successive values in the 2 column; ie., it changes the cumulative freq. distribution values to incremental frequency distribution values; eg. .016 − .0072 = .0088
4	No. of cycles per hour experienced by the aircraft due to maneuvering = (Freq/Hr) × (.9V_c) eg. .0088 × 148.5 = 1.31
A	Negative values of $a_n/a_{n_{LLP}}$ read from the maneuver spectrum that correspond to the same value of Cum. Freq/N.Mile as the AVERAGE value of positive $a_n/a_{n_{LLF}}$ (ie., the "in-between" values of col. 1).
5	$a_n = (a_n/a_{n_{LLF}}) \times a_{n_{LLF}}$ = col. A × (−2.52) and col. 1 × 2.8 as appropriate.
6	Δ Max Stress = a_n × Stress/g = col. 5 × 7410 psi
7	$S_m = \dfrac{[\text{Stress/g} + (\text{pos.}\Delta\text{ max stress})] + [\text{Stress/g} + (\text{neg.}\Delta\text{ max stress})]}{2}$ = Mean Stress (see example, page 3)
8	$S_a = [\text{Stress/g} + (\text{pos.}\Delta\text{ max stress}) - S_m]$ = Alternating Stress (see example, page 3)
9	Endurance cycles, N, read from the S-N curves, Figure 11 of AFS-120-73-2, for each pair of S_m and S_a calculated in col's. 7 & 8.
10	Damage per hour = 4/9

TOTAL DAMAGE DUE TO MANEUVERING IS SUMMATION OF COL. 10

LANDING DAMAGE ; (0→1g) Stress = 7410, si ; $S_{max} = \frac{2}{3} \times 7410 = 4943$ psi ;
See Instructions, Page 8 ; S_{min} = Ground Stress = −4520 psi

Descent Velocity ~fps	n ~g's	n (Ave)	$S_{(min)}$	S_a	S_m	ΣFreq/1000 Flts	Freq/1000 Flts	N (Impact) ~cycles	Damage per 1000 Flts. Impact	$.65 S_a$	N (Rebound) ~Cycles	n/N per 1000 Flts Rebound
0	1.33					$1.0 \cdot 10^3$						
		1.35	-6100	5520	-580		$1.3 \cdot 10^2$	$3.47 \cdot 10^6$	$3.74 \cdot 10^{-5}$	3315	$4.30 \cdot 10^7$	$3.03 \cdot 10^{-6}$
.2	1.37					$8.7 \cdot 10^2$						
		1.39	-6280	5615	-670		$8 \cdot 10^1$	3.25	2.96	3370	4.02	1.99
.4	1.41					7.9						
		1.43	-6465	5705	-760		9	3.04	2.96	3420	3.77	2.38
.6	1.45					7.0						
		1.47	-6645	5795	-850		9	2.84	3.16	3475	3.55	2.54
.8	1.49					6.1						
		1.51	-6830	5885	-940		9.5	2.66	3.56	3530	3.34	2.85
1.0	1.53					5.1						
		1.55	-7010	5975	-1030		9.5	2.50	3.79	3585	3.14	3.03
1.2	1.57					4.2						
		1.59	-7190	6065	-1120		9.5	2.35	4.04	3640	2.95	3.22
1.4	1.61					3.2						
		1.63	-7370	6155	-1215		8	2.21	3.62	3695	2.79	2.87
1.6	1.65					2.4						
		1.67	-7550	6245	-1305		6.5	2.08	3.12	3750	2.63	2.47
1.8	1.67					$1.8 \cdot 10^2$						
		1.71	-7730	6335	-1395		5	1.96	2.55	3800	2.48	2.01
2.0	1.73					$1.3 \cdot 10^2$						
		1.75	-7910	6425	-1485		3.8	1.85	2.06	3860	2.35	1.62
2.2	1.77					$9.2 \cdot 10^1$						
		1.79	-8090	6520	-1575		2.9	1.74	1.66	3910	2.22	$1.31 \cdot 10^{-6}$
2.4	1.81					6.3						
		1.83	-8270	6610	-1665		2.1	1.64	$1.28 \cdot 10^{-5}$	3965	2.11	$9.98 \cdot 10^{-7}$
2.6	1.85					4.2						
		1.87	-8450	6700	-1775		$1.4 \cdot 10^1$	1.55	$9.01 \cdot 10^{-6}$	4020	2.00	7.02
2.8	1.89					2.8						
		1.91	-8635	6790	-1845		$8.5 \cdot 10^0$	1.47	5.78	4075	1.89	4.50
3.0	1.93					2.0						
		1.95	-8815	6880	-1940		6.5	1.39	4.67	4130	1.79	3.62
3.2	1.97					$1.3 \cdot 10^1$						
		1.99	-8995	6970	-2025		3.9	1.32	2.96	4180	1.71	2.29
3.4	2.01					$9.1 \cdot 10^0$						
		2.03	-9175	7060	-2120		2.7	1.25	2.16	4235	1.62	1.67
3.6	2.05					6.4						
		2.07	-9355	7150	-2210		1.9	1.18	1.60	4290	1.54	$1.23 \cdot 10^{-7}$
3.8	2.09					4.5						
		2.11	-9540	7240	-2300		$1.2 \cdot 10^0$	1.12	$1.11 \cdot 10^{-6}$	4345	1.47	$8.53 \cdot 10^{-8}$
4.0	2.13					3.2						
		2.15	-9720	7330	-2390		$9.5 \cdot 10^{-1}$	1.07	$8.96 \cdot 10^{-7}$	4400	1.40	6.80
4.2	2.17					2.3						
		2.19	-9900	7420	-2480		6.0	$1.01 \cdot 10^6$	5.92	4455	1.33	4.51
4.4	2.21					1.7						
		2.23	-10080	7510	-2570		4.5	$9.63 \cdot 10^5$	4.67	4510	1.27	3.55
4.6	2.25					$1.2 \cdot 10^0$						
		2.27	-10260	7600	-2660		$3.3 \cdot 10^{-1}$	$9.17 \cdot 10^5$	$2.60 \cdot 10^{-7}$	4560	$1.21 \cdot 10^{-7}$	$2.73 \cdot 10^{-8}$
4.8	2.29					$9.2 \cdot 10^{-1}$						

CONTINUED ON PAGE 7

LANDING DAMAGE; continued from page 6

Descent Velocity ~fps	n ~g's	n (Ave)	$S_{(min)}$	S_a	S_m	$\frac{\Sigma \text{ Freq}}{1000 \text{ Flts}}$	$\frac{\text{Freq}}{1000 \text{ Flts}}$	N (Impact) ~cycles	Damage per 1000 Flts.	$.6S_a$	N (Rebound) ~Cycles	n/N per 1000 Flts.
									Impact			Rebound
4.8	2.29					$9.2 \cdot 10^{-1}$						
		2.31	-10440	7690	-2750		$2.4 \cdot 10^{-1}$	$8.72 \cdot 10^{5}$	$7.75 \cdot 10^{-7}$	4615	$1.16 \cdot 10^{7}$	$2.06 \cdot 10^{-8}$
5.0	2.33					6.8						
		2.35	-10620	7780	-2840		$1.7 \cdot 10^{-1}$	8.31	2.05	4670	1.10	$1.54 \cdot 10^{-8}$
5.2	2.37					5.1						
		2.39	-10800	7875	-2930		$1.2 \cdot 10^{-1}$	7.91	1.58	4725	1.05	$1.18 \cdot 10^{-8}$
5.4	2.41					3.8						
		2.42	-10960	7950	-3010		$9.5 \cdot 10^{-2}$	7.59	$1.25 \cdot 10^{-7}$	4770	$1.01 \cdot 10^{7}$	$9.37 \cdot 10^{-9}$
5.6	2.44					2.9						
		2.46	-11120	8031	-3090		6.5	7.28	$8.92 \cdot 10^{-8}$	4820	$9.7 \cdot 10^{6}$	6.67
5.8	2.48					2.3						
		2.50	-11300	8120	-3180		5.0	6.95	7.19	4875	9.33	5.36
6.0	2.52					1.8						
		2.54	-11480	8210	-3270		3.5	6.64	5.27	4930	8.94	3.92
6.2	2.56					1.4						
		2.57	-11640	8290	-3350		3.0	6.38	4.70	4975	8.60	3.49
6.4	2.59					$1.1 \cdot 10^{-1}$						
		2.61	-11800	8370	-3430		$2.8 \cdot 10^{-2}$	$6.13 \cdot 10^{5}$	$4.57 \cdot 10^{-8}$	5020	$8.29 \cdot 10^{6}$	$3.38 \cdot 10^{-9}$
6.6	2.63					$8.8 \cdot 10^{-2}$						

$\Sigma = 4.11 \cdot 10^{-4}$
IMPACT DAMAGE Per 1000 Flights

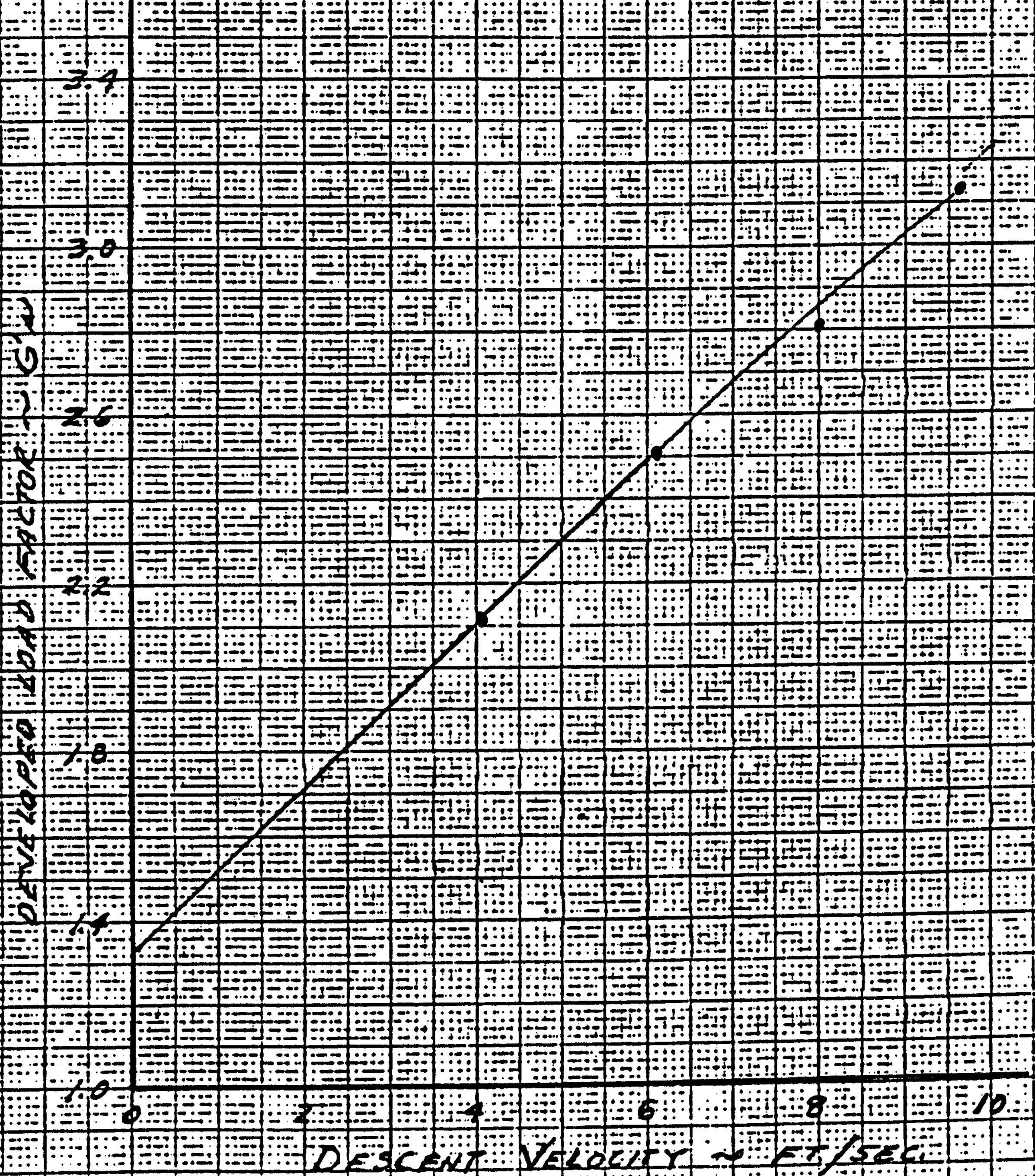
$\Sigma = 3.27 \cdot 10^{-5}$
REBOUND DAMAGE Per 1000 Flights

COLUMN	INSTRUCTIONS PERTINENT TO PAGES 6 & 7
S_{MAX}	Based on the (0→1g) stress × 2/3 lift acting on the airplane (ie., same as used in ldg. gear drop test equation) =(7410)(.667) = 4942 psi
S_{MIN}	Ground stress = −4520 psi (see page 11)
Descent Velocity	Arbitrarily chosen increments of descent velocity, ft/sec.
n	Read from plot of load factor ('g' units) vs. descent velocity (see page 9).
n (Ave)	Average of successive values in the "n" column
$S_{(MIN)}$	$= n_{(Ave)} \times S_{MIN}$ (eg., $1.35 \times (-4520) = -6100$)
S_a	$= \dfrac{S_{MAX} - S_{(MIN)}}{2}$ (alternating stress)
S_m	$= S_{(MIN)} + S_a$ (for a symmetric spectrum such as this) (mean stress)
$\frac{\Sigma \text{ Freq}}{1000 \text{ Flts.}}$	Read from Figure 9 of AFS-120-73-2 Report. Use the curve for T.E. Exec. NOTE: The table shows ΣFreq per 10^3 flights, and the curve is plotted as ΣFreq per 10^4 Flights.
$\frac{\text{Freq}}{1000 \text{ Flts.}}$	Difference between successive values in the "Σ Freq per 1000 Flts" column.
N (Impact)	Endurance cycles read from the S-N curves on Figure 11 of AFS-120-73-2 at S_a & S_m.
$\frac{\text{Damage}}{1000 \text{ Flts.}}$	$= \dfrac{\text{Freq. per 1000 Flts}}{N_{(impact)}}$, Initial Impact Damage
$.6\,S_a$	Second contact or rebound during landing $= .6\,S_a$ (Reduced alternating stress)
N (Rebound)	Endurance cycles read from the S-N curves on Figure 11 of AFS-120-73-2 at $.6\,S_a$ & S_m
$\frac{\text{Damage}}{1000 \text{ Flts}}$	$= \dfrac{\text{Freq per 1000 Flts}}{N_{(Rebound)}}$, Landing Rebound Damage

FINALLY, TOTAL LANDING DAMAGE IS SUMMATION OF BOTH "Damage/1000 Flts" COLUMNS

LANDING GEAR DROP TEST DATA

(ASSUMED 2/3 LIFT ACTING ON AIRCRAFT)



TAXI DAMAGE CALCULATIONS

Stress/g = Ground Stress = −4520 psi = S_m

$\pm\Delta g$	$\pm\Delta g$ (Ave)	S_a psi	Cum Freq / 1000 Flts	Δ Freq / 1000 Flts	N cycles	Damage Per 1000 Flts.
0			$6.4\cdot10^{5}$			
	.025	113		$1.46\cdot10^{5}$	∞	0
.05			$4.94\cdot10^{5}$			
	.075	339		$1.59\cdot10^{5}$		
.10			$3.35\cdot10^{5}$			
	.125	565		$1.41\cdot10^{5}$		
.15			$1.94\cdot10^{5}$			
	.175	791		$1.03\cdot10^{5}$		
.20			$9.10\cdot10^{4}$			
	.225	1017		$6.19\cdot10^{4}$		
.25			$2.91\cdot10^{4}$			
	.275	1245		$2.04\cdot10^{4}$		
.30			$8.70\cdot10^{3}$			
	.325	1470		$6.61\cdot10^{3}$		
.35			$2.09\cdot10^{3}$			
	.375	1695		$1.62\cdot10^{3}$		
.40			$4.65\cdot10^{2}$			
	.425	1920		$3.71\cdot10^{2}$		
.45			$9.42\cdot10^{1}$			
	.475	2150		$7.37\cdot10^{1}$		
.50			$2.05\cdot10^{1}$			
	.525	2375		$1.63\cdot10^{1}$		
.55			$4.16\cdot10^{0}$			
	.575	2600		$3.33\cdot10^{0}$		
.60			$8.30\cdot10^{-1}$			
	.625	2825		$6.75\cdot10^{-1}$		
.65			$1.55\cdot10^{-1}$			
	.675	3051		$1.27\cdot10^{-1}$		
.70			$2.85\cdot10^{-2}$			
	.725	3277		$2.35\cdot10^{-2}$	8.78×10^{7}	2.68×10^{-8}
.75			$5.00\cdot10^{-3}$			

$\Sigma = 2.68\cdot10^{-8}$ Per 1000 Flights

COLUMN	INSTRUCTIONS PERTINENT TO PAGE 10
S_m	Stress for static on-ground condition obtained from ground measurements with full fuel and at "zero-g" condition in flight (a difficult task to obtain reliable "zero-g" data). Change in stress due to off-loading fuel is obtained from strain data. The stress level is then corrected for fuel loading & operating weight.
$\pm \Delta g$	Arbitrarily chosen incremental load factor.
$\pm \Delta g$ (Ave)	Average between successive values of $\pm \Delta g$.
S_a	$= \pm \Delta g_{(Ave)} \times S_m$, Alternating Stress.
$\frac{\text{Cum. Freq}}{\text{1000 Flts}}$	Read from Taxi Spectrum, Figure 10 of AFS-120-73-2 (OCCURENCES/1000 LANDINGS versus $N_z \sim g$'s).
$\frac{\Delta\text{ Freq}}{\text{1000 Flts}}$	Difference between two successive values of $\frac{\text{Cum. Freq}}{\text{1000 Flts.}}$
N	Endurance, obtained from S-N curves, Figure 11 of AFS-120-73-2 at S_m & S_a.
$\frac{\text{Damage}}{\text{1000 Flts}}$	$= \frac{\Delta\text{ Freq}}{\text{1000 Flts}} \Big/ N$

FINALLY, SUM THE COLUMN OF "Damage per 1000 Flts." TO OBTAIN THE TOTAL DAMAGE PER 1000 FLIGHTS

G.A.G. CYCLE MAX. STRESS

See Instructions, Page 13

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)
$\frac{a_n}{a_{n_{LLF}}}$	Cum Freq / N.Mile	Cum Freq / 1000 Flts	Max Stress ~ psi $S_m + S_a$	$\frac{a_n}{a_{n_{LLF}}}$	Cum Freq / N.Mile	Cum Freq / 1000 Flts	Max Stress ~ psi $S_m + S_a$	S_{MAX} From Plot	GUST Cum Freq / 1000 Flts From Plot	MANEUVER Cum Freq / 1000 Flts From Plot	TOTAL Cum Freq / 1000 Flts
.13	.33	31800	9480	.13	.011	1062	10100	14000	120	65	185
.19	.11	10620	10440	.19	.0045	435	11350	13000	330	130	460
.25	.025	2410	11410	.25	.0019	184	12590	12000	1080	270	1350
.31	.007	676	12350	.31	.00075	72	13830	11000	4300	550	4850
.37	.0026	251	13240	.37	.00022	21	16250	10000	16100	1120	17220
.43	.00096	93	14280	.43							
GUST				MANEUVER							

COLUMN	INSTRUCTIONS PERTINENT TO PAGE 12
1	$\frac{a_n}{a_{n_{LLF}}}$ taken from the table used in calculating gust fatigue damage (page 2).
2	Read from gust spectrum, Fig. 2 of AFS-120 Rep't.
3	$\frac{\text{Cum Freq}}{1000\text{ Flts}} = \frac{\text{Cum Freq}}{\text{N. Mile}} \times .9V_c \times \frac{\text{Hours}}{\text{Flt}} \times 1000\text{ Flts}$ $= ② \times 148.5 \times .65 \times 10^3 = ② \times 96,500$ (gust)
4	Max. gust stress $= S_m + S_a$ taken from page 2.
5	$\frac{a_n}{a_{n_{LLF}}}$ taken from maneuver damage table (page 4).
6	Read from maneuver spectrum, Fig 7 of AFS-120 Rep't
7	$\frac{\text{Cum Freq}}{1000\text{ Flts}} = ⑥ \times 96,500$
8	Max. maneuver stress $= S_m + S_a$ taken from page 4.
	At this point plot ④ vs. ③ and ⑧ vs. ⑦ (see page 14)
9	Choose uniformly-spaced values of Max Stress over the range covered by the plot of ③ vs. ④.
10	For each value of Max Stress chosen in col. ⑨, read (gust) $\frac{\text{Cum Freq}}{1000\text{ Flts}}$ and enter in col. ⑩
11	For each value of Max Stress chosen in col. ⑨, read (maneuver) $\frac{\text{Cum Freq}}{1000\text{ Flts}}$ and enter in col. ⑪
12	= ⑩ + ⑪, total max. stresses occuring in the vicinity of once per flight. Landing and taxi stresses do not ordinarily contribute to the max. once-per-flight stress.

FINALLY, PLOT ⑨ versus ⑫. AT $\frac{\text{Cum Freq}}{1000\text{ Flts}} = 10^3$,

READ THE MAX. STRESS FOR THE GAG CYCLE.

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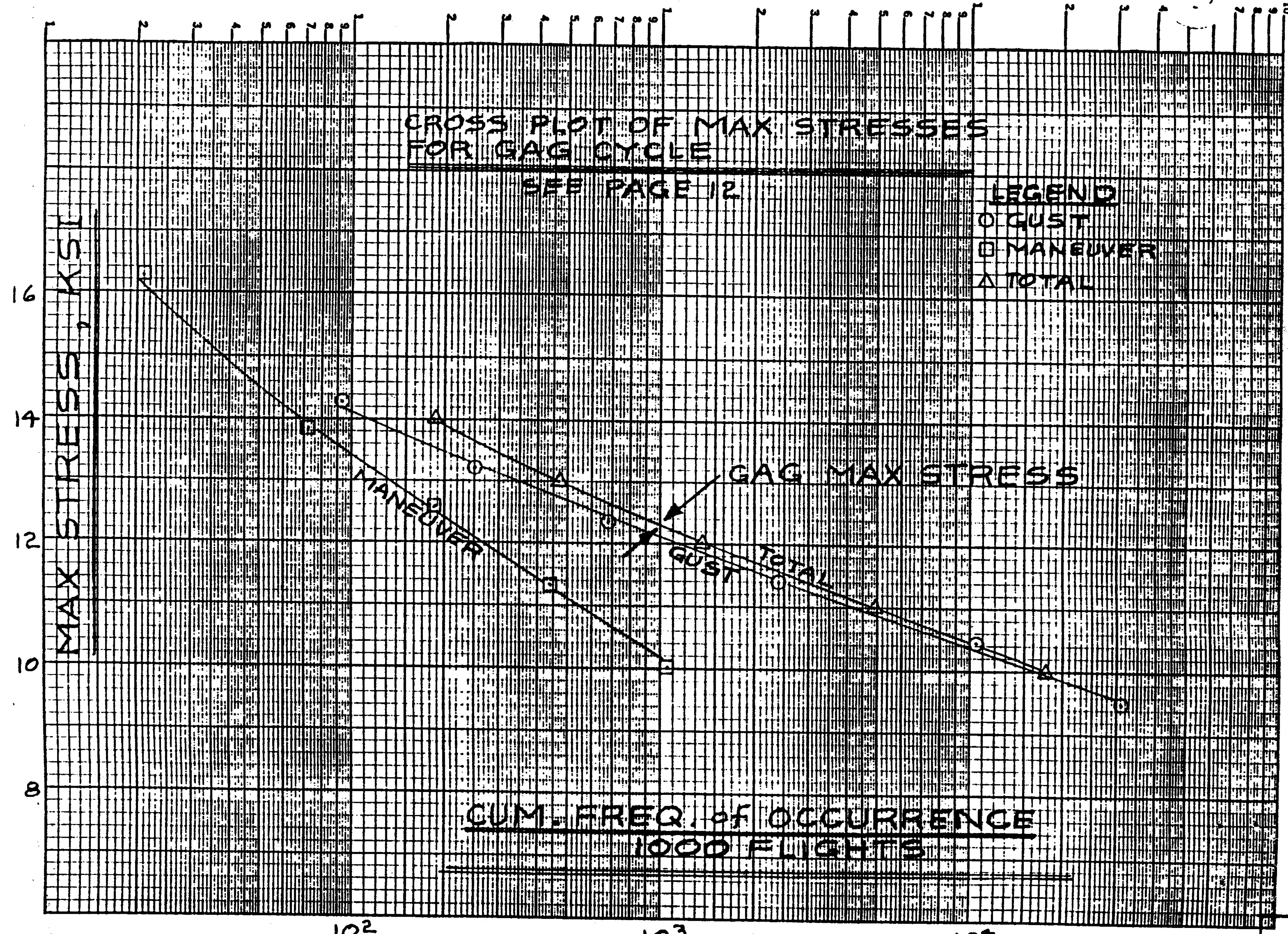


G.A.G. CYCLE MIN. STRESS

See Instructions, Page 16

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩	⑪	⑫
$\pm\Delta g$	Min. Stress TAXI ~psi	$\frac{\Sigma Freq}{1000 Flts}$ TAXI	n	Min. Stress Impact ~ psi	Min Stress Rebound ~ psi	$\frac{\Sigma Freq}{1000 Flts}$ LANDING	S_{MIN} From Plot ~ psi	$\frac{\Sigma Freq}{1000 Flts}$ TAXI	LDG Impact	LDG Rebound	TOTAL
.30	-5875	8700	1.33	-6012	-3821	1000	-6000	4000	1000	18	5018
.35	-6100	2090	1.41	-6373	-4110	790	-6100	2010	960	14	2984
.40	-6330	465	1.53	-6916	-4544	515	-6300	520	850	8	1378
.45	-6555	94	1.61	-7277	-4833	325	-6600	70	690	4	764
.50	-6870	20.5	1.81	-8181	-5556	63	-6900	8.5	520	1.6	530
.55	-7005	4.2	2.01	-9085	-6280	9.1					
TAXI			LANDING								

L

GAG CYCLE MIN. STRESS

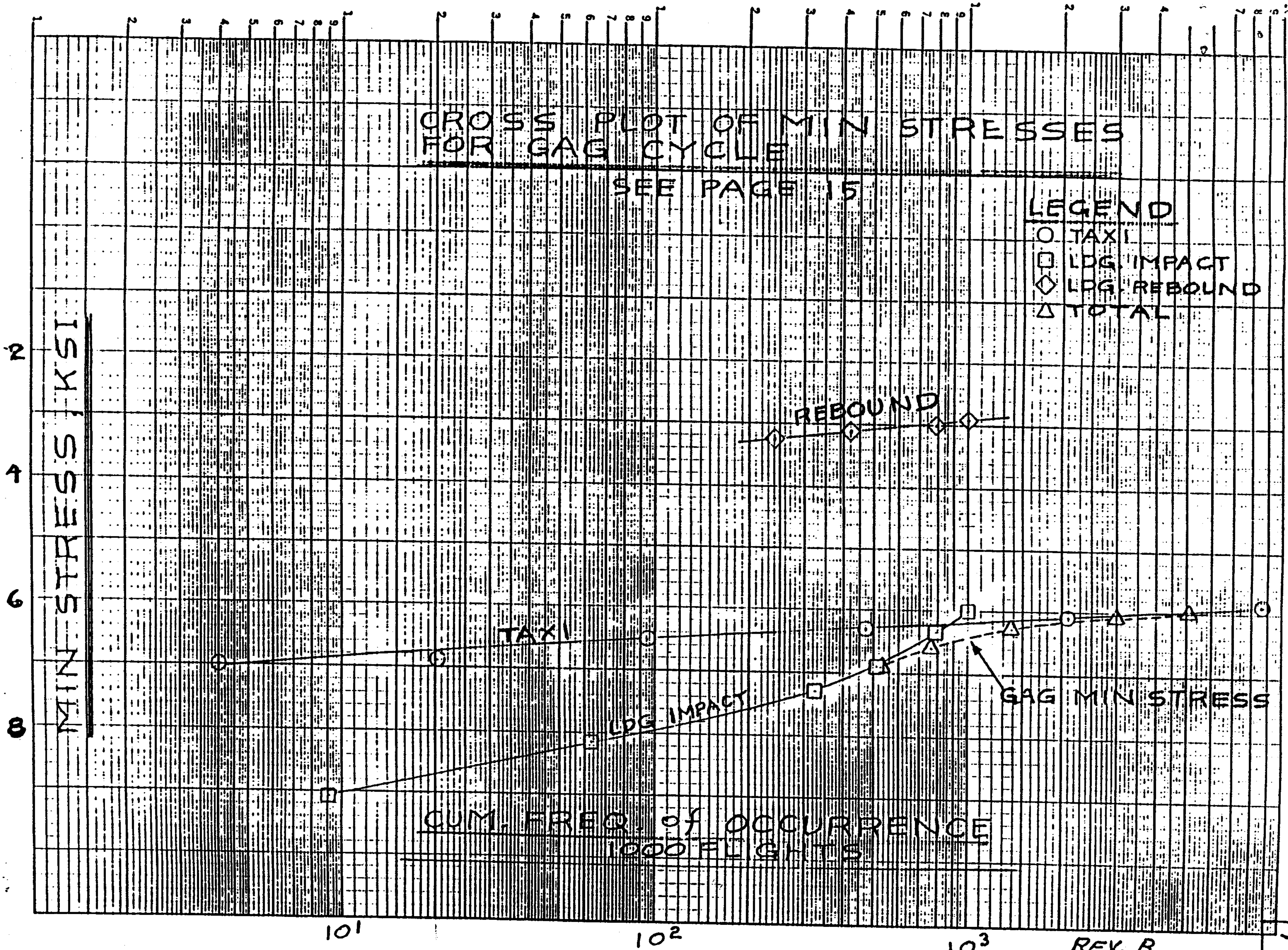
COLUMN	INSTRUCTIONS PERTINENT TO PAGE 15
1	$\pm \Delta g$ values selected from Taxi Damage calc's, page 10, over a range surrounding $\frac{\Sigma \text{Freq}}{1000 \text{ Flts}} = 10^3$.
2	Min. Stress (Taxi) = (stress/g) × ($\pm \Delta g$ + 1) = −4520 × (1 + 1)
3	$\frac{\Sigma \text{Freq}}{1000 \text{ Flts}}$ taken from Taxi Damage calc's, page 10.
4	n values selected from Landing Damage calc's, page 6, over a range near $\frac{\Sigma \text{Freq}}{1000 \text{ Flts}} = 10^3$.
5	Min. Stress (Landing) = (stress/g) × (n) = −4520 × 4.
6	Min. Stress (Rebound) = $S_m - .6 S_a$ where S_m and S_a can be interpolated off the table on page 6, or calculated per page 8. This assumes S_m for rebound is equal to S_m for initial impact.
7	$\frac{\Sigma \text{Freq}}{1000 \text{ Flts}}$ taken from Landing Damage calc's, page 6.

AT THIS POINT PLOT 2 vs. 3, 5 vs. 7 and 6 vs. 7 AS SHOWN ON PAGE 17.

8	Choose uniformly-spaced values of Min Stress over a range near the intercept at $\frac{\Sigma \text{Freq}}{1000 \text{ Flts}} = 10^3$.
9	For each value of Min Stress chosen in col. 8, read (taxi) $\frac{\Sigma \text{Freq}}{1000 \text{ Flts}}$ and enter in col. 9.
10	Likewise read (Landing Impact) $\frac{\Sigma \text{Freq}}{1000 \text{ Flts}}$ & enter in col. 10.
11	Likewise read (Landing Rebound) $\frac{\Sigma \text{Freq}}{1000 \text{ Flts}}$ & enter in col. 11.
12	= 9 + 10 + 11, total min. stresses occuring in the vicinity of once per flight. Gust and maneuver do not ordinarily contribute to the min. stresses.

FINALLY, PLOT 9 versus 12. AT $\frac{\Sigma \text{Freq}}{1000 \text{ Flts}} = 10^3$, READ THE MIN. STRESS FOR THE GAG CYCLE

REV. B MARCH 91


CROSS PLOT OF MIN STRESSES
FOR GAG CYCLE
SEE PAGE 15
LEGEND
○ TAXI
□ LDG. IMPACT
◇ LDG. REBOUND
△ TOTAL
REBOUND
TAXI
LDG IMPACT
GAG MIN STRESS
MIN STRESS, KSI
2
4
6
8
CUM. FREQ. of OCCURRENCE
1000 FLIGHTS
10^1
10^2
10^3

REV. B

SAFE LIFE

Max and Min stress for the Ground-Air-Ground cycle can be read from the "TOTAL" curves on pages 14 and 17, respectively.

$$S_{MAX} = 12,300 \text{ psi} \ \& \ S_{MIN} = -6450 \text{ psi}$$

$$S_a = \frac{12,300-(-6450)}{2} = 9380 \text{ psi}$$

$$S_m = 12,300 - 9380 = 2920 \text{ psi}$$

From the S-N curves of Figure 11 in AFS-120-73-2 the endurance is $N = 1.0 \times 10^5$ cycles. Fatigue damage due to the GAG cycle is $\frac{1}{N} = 1.0 \cdot 10^{-5}$ per landing, which is also 1.0×10^{-5} per flight. GAG damage per 1000 flights is 1.0×10^{-2}. Flight duration is .65 hours (per AFS-120-73-2), so GAG damage per 1000 hours is $1.0 \times 10^{-2}/.65 = 1.54 \cdot 10^{-2}$.

SUMMARY

CONTRIBUTION	PAGE	DAMAGE Per 1000 Hrs
Gust	2	.0234
Maneuver	4	.0017
Landing:		
impact	7	.0006
rebound	7	—
Taxi	10	—
GAG	18	.0154
	TOTAL DAMAGE =	.0411

$$\text{Unfactored Safe Life} = \frac{1000}{.0411} = 24,300 \text{ Hours}$$

REV. C
7-20-93