U.S. Department
of Transportation

Federal Aviation Administration

Advisory Circular

AC 23-13A

FATIGUE, FAIL-SAFE, AND DAMAGE TOLERANCE EVALUATION OF METALLIC STRUCTURE FOR PART 23 AIRPLANES

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FOREWORD

This Advisory Circular (AC) sets forth an acceptable means of showing compliance with Title 14 Code of Federal Regulations (14 CFR), part 23. This guidance is applicable to fatigue, fail-safe, and damage tolerance evaluations of metallic structure in normal, utility, aerobatic, and commuter category airplanes. This AC also provides information on approval of continued operational flight with known cracks in the structure of small airplanes. Finally, this AC consolidates existing policy documents and certain technical reports into one document.

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1. What is the purpose of this AC?

a. This advisory circular (AC) provides information and guidance concerning acceptable means, but not the only means, of compliance with Title 14 of the Code of Federal Regulations (14 CFR) part 23. This guidance is applicable to fatigue, fail-safe, and damage tolerance evaluations of metallic structure in normal, utility, aerobatic, and commuter category airplanes. This AC also provides information on approval of continued operational flight with known cracks in the structure of small airplanes. This AC consolidates existing policy documents, and certain technical reports, into a single document. This AC incorporates, and supersedes, FAA Report No. AFS-120-73-2, Fatigue Evaluation of Wing and Associates Structure on Small Airplanes, dated May 1973. Material in this AC is neither mandatory nor regulatory in nature and does not constitute a regulation.

2. What is incorporated in this AC and, thereby, superseded?

The following AC has been incorporated into this revision in whole or in part and is hereby superseded:

AC 23-13, Fatigue and Fail-Safe Evaluation of Flight Structure and Pressurized Cabin for Part 23 Airplanes, dated April 15, 1993.

3. What is the background for this AC and what time period does this AC cover?

The AC is current through Amendment 23-55, effective March 1, 2002. This material spans approximately 38 years of Federal Aviation Administration (FAA) aviation history.

4. How should the information contained in this AC be applied?

a. This material does not have any legal status and should be treated accordingly. However, to ensure standardization in the certification process, these procedures should be considered during all small airplane Type Certification (TC) and Supplemental Type Certification (STC) activities. The user of this AC should check the latest amendments to the applicable 14 CFR part to verify that the regulations referenced in this AC have not been superseded or that additional regulations have not been added.

b. New approaches to fatigue evaluation, design features, methods of fabrication, and new airplane configurations may require variations and deviations from the procedures described in this AC. Engineering judgment, guided by the extensive literature on the subject, should be exercised for each particular application. An applicant should evaluate the primary structure of the pressurized cabin, wing, empennage, and associated structures for any original design and for any design changes that affect the loading spectra, internal stresses, or stress concentrations or that change the construction methods or materials. Changes to the design that may be minor from a static strength standpoint can have a major effect on fatigue characteristics.

5. What regulations are discussed in this AC?

a. The requirements for fatigue evaluation included in 14 CFR, part 23, §§ 23.571, 23.572, 23.573, 23.574, and 23.627.

b. The requirements for inspections and Instructions for Continued Airworthiness included in 14 CFR, part 23, §§ 23.575, 23.611, and 23.1529.

c. The requirements for minimization of rotorburst hazards included in 14 CFR, part 23, § 23.903.

6. What other publications are related to this AC?

a. AC 23-19, Airframe Guide for Certification of Part 23 Airplanes, dated January 27, 2003.

b. AC 25.571-1C, Damage Tolerance and Fatigue Evaluation of Structure, dated April 29, 1998.

c. AC 25.613-1, Material Strength Properties and Material Design Values, dated August 6, 2003.

d. AC 20-128A, Design Considerations for Minimizing Hazards caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor Failure, dated March 25, 1997.

e. AC 20-107A, Composite Aircraft Structure, dated April 25, 1984.

f. AC 91-56A, Continuing Structural Integrity Program for Large Transport Category Airplanes, dated April 29, 1998.

g. AC 91-60, The Continued Airworthiness of Older Airplanes, dated June 13, 1983.

h. Aging Airplane Inspection and Maintenance Baseline Checklist for Airplanes Without a Type Specific Checklist (Best Practices Guide for Maintaining Aging General Aviation Airplanes). Electronic version available at: *www.faa.gov/certification/aircraft/aceagingchecklist.doc*

7. What are the definitions for the common terms used in this AC?

This Advisory Circular uses the definitions listed below. Some of these terms are used throughout and will only be defined here. Sometimes, the term definitions may be repeated in the relevant document section as an aid to the reader.

a. Principal Structural Element (PSE). A PSE is an element that contributes significantly to the carrying of flight, ground, or pressurization loads, and whose integrity is essential in maintaining the overall structural integrity of the airplane.

b. Fatigue. The process of progressive localized permanent structural change occurring in a material subjected to conditions that produce fluctuating stresses and strains at some point or points, which may result in cracks or complete fracture after a sufficient number of fluctuations.

c. Safe-Life. The safe-life of a structure is that number of events, such as flights, landings, or flight hours, during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue cracking.

d. S-N or ε–N. Stress–Life (S-N) or Strain–Life (ε -N) curves depict the magnitude of applied stress (S) or strain (ε) necessary to develop a fatigue crack in a specimen at a given life (N), where N is expressed in the number of cyclic applications of stress or strain.

e. Scatter Factor. The scatter factor, or life reduction factor, is a statistically derived divisor applied to fatigue test results to account for the variation in fatigue performance of built-up or monolithic structures. A scatter factor can also be used in a fatigue analysis to address the uncertainties inherent in a fatigue analysis.

f. Fail-Safe. Fail-safe is the attribute of the structure that permits it to retain its required residual strength for a period of unrepaired use after the failure or partial failure of a principal structural element.

g. Damage Tolerance. Damage tolerance is the attribute of the structure that permits it to retain its required residual strength for a period of use after the structure has sustained a given level of fatigue, corrosion, accidental, or discrete source damage.

h. Discrete Source Damage. Damage to the structure caused by a discrete source, such as fragments from uncontained engine failures.

i. Residual Strength. The strength capability of a structure after the structure has been damaged due to fatigue, corrosion, or discrete source damage. The residual strength capability includes consideration of static strength, fracture, and stiffness.

CHAPTER 1. SMALL AIRPLANE FATIGUE REGULATIONS

1-1. What is the goal of the small airplane fatigue requirements?

The goal of the fatigue requirements is to maintain the design strength capability, both limit and ultimate load, throughout the operational life of the airplane.

1-2. How have the fatigue requirements changed over time?

a. Before 1957. The only fatigue requirements for small airplanes appeared in the Civil Air Regulations (CAR), § 3.307. The CAR 3 paragraph required the airplane designer to choose design details that avoided stress concentrations. This airworthiness requirement applied to all structural design details on the whole airplane. When the CAR requirements were recodified to the 14 CFR parts, this requirement was carried over as § 23.627.

b. 1957. Amendment 3-2 added a specific requirement to CAR 3 for the fatigue assessment of pressurized fuselage structure. The requirement gave applicants the option of a safe-life evaluation or incorporating fail-safe design features. CAR § 3.270 was recodified as 14 CFR, part 23, § 23.571.

c. 1969. Amendment 23-7, effective September 14, 1969, added fatigue requirements for the wing, wing carry-through, and attaching structure in § 23.572.

d. 1979. Special Federal Aviation Regulation No. 41 (SFAR 41), effective September 17, 1979, required a fatigue substantiation of “those parts of the wing, wing carry-through, vertical fin, horizontal stabilizer, and attaching structure whose failure would be catastrophic.”

e. 1989. Amendment 23-38, effective October 26, 1989, added the empennage to the § 23.572 fatigue requirements.

f. 1993. Amendment 23-45 introduced § 23.573 requiring damage tolerance and fatigue evaluation for composite structures. This amendment also added damage tolerance as an option for metallic pressurized cabin in § 23.571 and for wing, empennage, and associated structure in § 23.572. Amendment 23-45 also expanded the fatigue assessment of wing structure to include canards, tandem wings, and winglets/tip fins.

g. 1996.

(1) Amendment 23-48 introduced § 23.574, which requires damage tolerance for metallic structure on commuter category airplanes. This amendment became effective on March 11, 1996.

(2) Amendment 23-48 also introduced § 23.575, which clarifies that inspections and any other procedures necessary to prevent catastrophic failure must be included in the Limitations Section of the Instructions for Continued Airworthiness (ICA).

1-3. Can you summarize the current fatigue regulations for small airplanes?

a. For metallic pressurized cabin, wing, empennage, and associated structure in normal, utility, and aerobatic category airplanes, you have the option of the following fatigue evaluation methods:

(1) A fatigue strength evaluation, more commonly known as a safe-life evaluation. See § 23.571(a) for pressurized cabin and § 23.572(a)(1) for wing, empennage, and associated structure.

(2) A fail-safe strength evaluation. See § 23.571(b) for pressurized cabin and § 23.572(a)(2) for wing, empennage, and associated structure.

(3) A damage tolerance evaluation. See § 23.573(b) and § 23.571(c) for pressurized cabin and § 23.572(a)(3) for wing, empennage, and associated structure.

b. For wings, empennage, and associated structure, you may be able to show compliance to the fatigue requirements by comparing your design to an existing design. This method of substantiation includes showing that the structure, operating stress level, materials, and expected uses are comparable, from a fatigue standpoint, to a similar design that has had extensive satisfactory service experience. See § 23.572(a).

c. For metallic commuter category airplanes, you must use a damage tolerance evaluation. If you can show that damage tolerance is impractical for a particular structure, you may use a fatigue strength, or safe-life evaluation. See § 23.574.

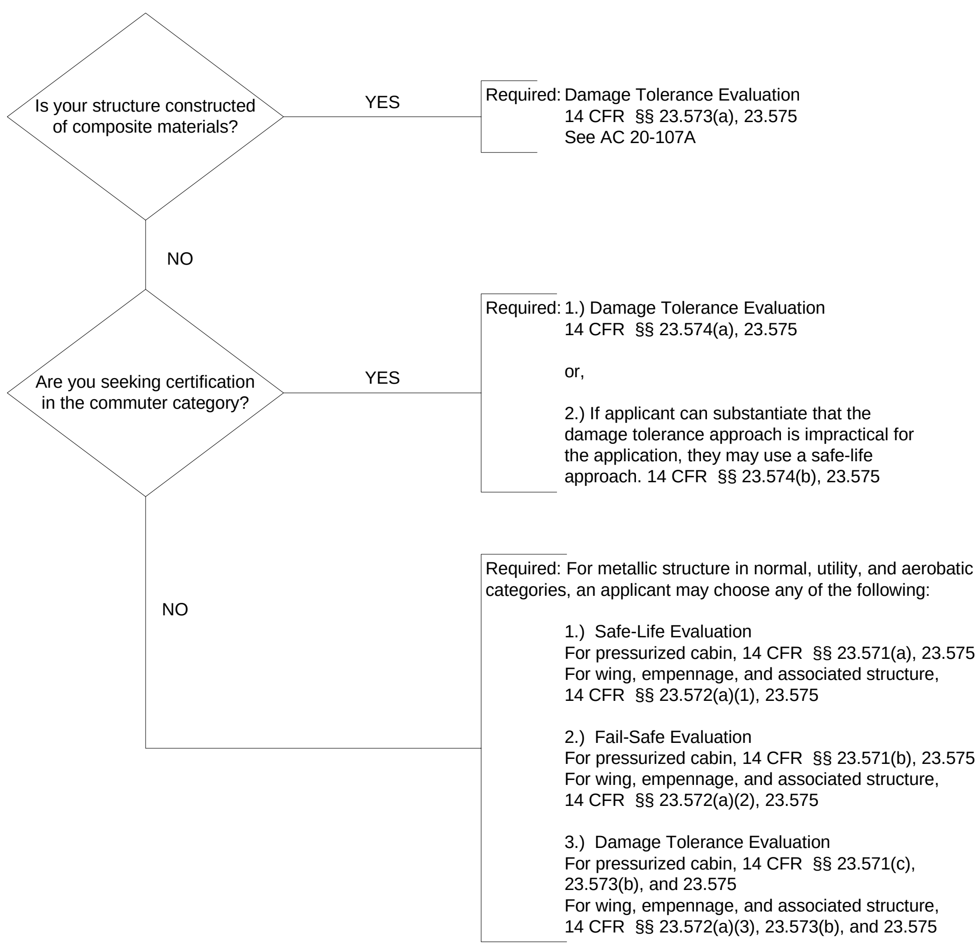
d. For all categories of small airplanes constructed with composite materials, you must use a damage tolerance evaluation. AC 20-107A provides guidance for composite structures.

e. The following flowchart may help clarify which method of fatigue evaluation is required for your application.

Figure 1

Which Fatigue Regulations Apply to My New Type Design?

(Amendment 23-48, effective February 9, 1996.)



f. The most recent amendment to part 23 that revised the fatigue requirements was Amendment 23-48, effective February 9, 1996. Always check the latest amendment to part 23 to verify that the regulations referenced in this AC have not been superseded or that additional regulations have not been added.

1-4. Which metallic components should I include in the fatigue evaluation?

Although the fatigue regulations discuss “those parts of the airframe structure whose failure would be catastrophic,” a 1998 policy letter clarifies that you are required to include only certain sections of metallic airplanes in your fatigue evaluation. For airplanes certificated in any small airplane category, these sections include the pressurized cabin, wing (includes tandem wings, canards, winglets and tip fins), empennage, and associated structures. You do not need to include other portions of the airplane, including the landing gear and engine mounts.

1-5. Are the small airplane fatigue regulations different in other countries?

a. The FAA fatigue regulations for small airplanes are the same as those of many foreign countries. The FAA and the CAA (Civil Aviation Authority) of these countries have worked together to harmonize the part 23 regulatory wording.

b. Even though the regulations may be identical, some differences in policy, guidance, and acceptable means of compliance may exist between the FAA and the CAA of foreign countries. These differences include:

(1) The CAA of some countries may require fatigue evaluation of the engine mounts and landing gear.

(2) The CAA of some countries may require different scatter factors in a safe-life evaluation.

1-6. How do I show compliance to the fatigue requirements for alterations, modifications, or changes to the design?

a. The fatigue performance of the structure should be evaluated for any alteration, modification, or change to the type design. Physical changes to the design that may effect fatigue performance include changes that increase the external or internal loads, alter the distribution of internal loads, or change the materials or method of construction. You should also evaluate the effect of any change to the airplane’s operational characteristics that invalidate the assumptions used in the original fatigue substantiation. Changes to the operational characteristics that may be important for fatigue include higher design airspeeds or higher average speed. They also include changes to the maximum allowable weight and center of gravity envelope, changes to the average weight and center of gravity location, and engine or propeller changes.

b. Modifiers and applicants for supplemental type certificates (STCs) often do not have access to the Type Certificate holder’s original fatigue substantiation data. In these cases, applicants often complete a “comparative fatigue analysis” in which the fatigue performance of the modified and unmodified airplane are compared. Applicants use the results of the comparative analysis to determine the appropriate changes to the safe-life or inspection program of the modified airplane.

CHAPTER 2. SAFE-LIFE FATIGUE EVALUATION

2-1. What is a safe-life fatigue evaluation?

A safe-life fatigue evaluation is the assessment of a structure to ensure that the structure is able to withstand, without catastrophic failure, the repeated loads of variable magnitude expected in service throughout its operational life. The "safe-life" is a point in the airplane's operational life, expressed in hours or flight cycles, at which the structure must be taken out of service or replaced. The structure is retired or replaced at the safe-life to prevent the structure from developing fatigue cracks. An applicant may determine the safe-life of a structure using the following methods:

a. full-scale testing,

b. component testing, or

c. analysis supported by test evidence.

2-2. What is the definition of a fatigue failure in a safe-life structure?

For a safe-life structure, fatigue failure is the development of a detectable crack. A detectable crack is one that can be detected by common inspection methods, or the inspection methods required in the maintenance instructions. Since a safe-life evaluation usually does not include demonstration of crack growth rates or residual strength capability, we assume that the development of a detectable crack may result in catastrophic failure of the structure.

2-3. What are the steps in evaluating the safe-life of a structure?

We can summarize the essential elements of any safe-life evaluation as follows:

a. Estimate or measure the expected loading spectra for the structure. The loading spectra should include all loads you expect the structure to experience in normal operation. This includes gust loads, maneuver loads, and pressurization loads. Although you are not required to include the landing gear and engine mounts in your fatigue evaluation, you need to include appropriate taxi loads, landing impact loads, engine loads, vibration, buffet, and propeller slipstream effects in your loading spectra.

b. Conduct a structural analysis, which includes consideration of the effects of stress concentrations. Your structural analysis should include determination of gross stress levels and determination of the stress concentration factors, K_t, for all details in principal structural elements. Your analysis should consider, in detail, structural joints and fittings, paying particular attention to eccentrically loaded joints and the bearing and bypass stresses in the joint. The structural analysis should identify locations where fretting may occur. Identify areas where corrosion may develop or extreme thermal environments may affect fatigue performance.

c. Conduct a fatigue test of any structure that cannot be related to a test background to establish the structure's response to the typical loading spectrum expected in service. Full-scale fatigue testing is the most reliable method of determining the response of the structure to expected operational loadings. Fatigue substantiation by analysis is also acceptable. However, the stress-life (S-N) or strain-life (ε–N) data used in the analysis must be based on fatigue tests of similar structures or adjusted to account for differences between the actual structure and the tested structure.

d. Determine reliable replacement times by interpreting the loading history, variable load analysis, fatigue test data, service experience, and fatigue analysis. The reliability of the replacement times is usually achieved through the use of fatigue scatter factors or other statistical adjustments.

e. Provide data for inspection and maintenance instructions and guidance information to the operators. This data should include inspection methods, inspection thresholds and intervals.

2-4. What loading spectra should I use?

a. Appendix 1 presents gust and maneuver flight load spectra in both graphic and tabulated form for the following types of airplanes and usage:

(1) Single-Engine Executive Usage (non-pressurized, engine size greater than 185 hp)

(2) Single-Engine Personal Usage (non-pressurized, engine size less than or equal to 185 hp)

(3) Single-Engine Instructional Usage (non-pressurized)

(4) Single-Engine Aerobatic Usage (non-pressurized)

(5) Twin-Engine General Usage (non-pressurized)

(6) Twin-Engine Instructional Usage (non-pressurized)

(7) Pressurized Usage

(8) Special Usage (including survey and aerial application)

b. For other types of loads:

(1) Appendix 1 also provides ground load spectra, including landing impact and taxi loads.

(2) The loading spectrum for pressurized cabins should include the repeated application of the normal operating differential pressure, and the superimposed effects of the gust and maneuver flight loads and external aerodynamic pressure.

(3) If the wing center section skin panels are affected by cabin pressurization, the wing loading spectrum should include the effects of pressurization.

(4) Reference 1 provides additional guidance on load spectra development and fatigue evaluation for the empennage, and for configurations with canards (or forward wings) and winglets (or tip fins).

c. Appendix 1 also discusses the gust load formula you should use to develop the gust load spectra for your airplane.

d. Your loading spectra should include the ground-air-ground cycle (GAG). The GAG cycle represents the range of the maximum and minimum loads expected to occur on a per flight basis. Typically, the minimum load results from landing or taxi conditions and the maximum load from the gust or maneuver spectra. For typical airplanes certificated in the normal category, two-thirds to three-fourths of the total fatigue damage on the wing may be caused by the GAG cycle.

e. While positive and negative load cycles are considered to occur randomly in service, the high positive and negative loads of a given type of repeated loading tend to occur at the same time. For testing and analysis purposes, the high positive load cycles are typically combined with the high negative load cycles of the same frequency.

2-5. How were the load spectra in Appendix 1 developed?

a. The FAA developed the flight load spectra in Appendix 1 based on a statistical analysis of the data presented in Reference 2. We used the ground load spectra directly from Reference 3 for the Appendix 1 ground load spectra.

b. The Appendix 1 flight load spectra include an increment (1.5 standard deviations) added to the average measured load frequency. The increment accounts for the variability in loading spectra experienced from individual airplane to airplane. The magnitude of the increment was selected to maintain the probability that a component will reach its safe-life without a detectable fatigue crack established by the scatter factor (see paragraph 2-15). Reference 4 provides additional details on the development of the Appendix 1 flight load spectra.

2-6. Can I use other loading spectra in place of the Appendix 1 spectra?

The Appendix 1 spectra are the FAA's approved load spectra for use in small airplane fatigue evaluations. Applicants may propose alternate spectra in certain situations. If an applicant proposes alternate spectra, the Small Airplane Directorate Standards Office must approve these spectra. Any spectrum proposed by an applicant should maintain the probability that a component will reach its safe-life without a detectable fatigue crack established by the scatter factor (see paragraph 2-15 and Reference 4). Situations in which applicants might consider alternate spectra include the following:

a. The applicant may develop gust spectra for pressurized usage based on power spectral density (PSD) methods. It is acceptable in this case to use PSD mission analysis techniques to account for typical operating mission profiles.

b. The applicant has access to a large database of measured spectra data on a similar configuration or on an airplane with similar usage. This type of data is particularly useful in situations involving unique usage, such as agricultural usage.

2-7. What should I consider in developing a mission profile?

In developing the mission profile for your airplane, you should use the following assumptions:

a. Flight time

(1)	Single-Engine and Twin-Engine (non-pressurized)	0.65 hours
(2)	Pressurized	1.1 hours
(3)	Single-Engine special usage (low level survey)	2.0 hours
(4)	Twin-Engine special usage	3.0 hours

b. Airplane Speed

(1) The speed for determining miles flown should not be less than 0.9 V_{NO} or 0.9 V_{MO}.

(2) For special usage, the speed for determining miles flown may be 100 knots or 0.9 V_A, whichever is less.

c. Gross weight and load distribution

(1) You should base estimates of the gross weight and distribution of disposable load on conservative estimates of typical operating conditions.

(2) One acceptable means of compliance, used by some small airplane applicants, is to assume the weight condition that gives the highest 1-g stress.

d. The mission profiles of certain airplanes should consider the unique aspects of the usage. Some examples follow:

(1) Instructional usage. Takeoff and landing training, or "touch-and-go" training, is a significant portion of the student pilot training curriculum. Incorporate "touch-and-go" training into the mission profiles of any airplane used for instructional purposes. Each "touch-and-go" should be treated as a short duration flight, 6 to 10 minutes in length.

(2) Mixed usage. Some airplanes are certificated in more than one category. A common example is an airplane certificated in both the aerobatic and utility/normal categories. In general, the safe-life should be based on the usage that results in the shorter life. Some applicants have estimated a mix of missions, the percent of time spent operating in either category, as a way of accounting for the different types of usage. In using this approach, it is not acceptable to use a mix of missions within a single flight. For purposes of fatigue evaluation, a single flight is operated within a single category and within a single type of usage.

(3) Severe usage. Some airplanes may be used more severely than you had assumed in the fatigue evaluation. Common examples of this include airplanes considered to be in the single-engine personal or executive usage category used for pipeline and utility patrol, for instruction, or for short duration commuter and air taxi flights. Some applicants address this issue by placing a statement in the limitations section of the Airplane Flight Manual (AFM) that certain types of usage require a re-evaluation of the structure by the Type Certificate (TC) holder. The AFM lists these types of usage and requests the owner to contact the TC holder regarding the usage.

2-8. Does the FAA require a flight test to validate the loads I use in my fatigue evaluation?

In most cases, the FAA will not require flight test validation of external loads or the internal stresses used in a fatigue evaluation. Typically, it is acceptable for an applicant to substantiate that the predicted external loads conservatively represent actual flight load conditions. However, it is in the applicant's best interest to accurately know the internal loads, or stresses, at critical fatigue details. A one percent increase in stress level can shorten the expected fatigue life by more than five percent. Due to the sensitivity of fatigue life to stress level, an applicant should consider flight testing to measure strains at or near fatigue critical structural details of the principal structural elements.

2-9. Does the FAA require validation of the stresses I use in my fatigue evaluation?

In determining whether validation or substantiation of the stresses used in a fatigue evaluation is necessary, the applicant and the FAA should consider the following:

a. For full-scale fatigue testing, an applicant should substantiate that the loads applied in the test either accurately or conservatively represent actual flight load conditions. Substantiation may be shown with comparisons of applied and predicted external loads. Substantiation should include demonstration that the applied external loads produce conservative stresses in the fatigue critical components.

b. Component testing is different from full-scale testing in that the component test omits substantial portions of adjacent structure that may impart local stresses on the tested component. Because of the omitted structure and other simplifications inherent in component testing, an applicant should validate that the stresses in the component test accurately or conservatively represent the stresses in the actual structure. One acceptable method of validating the internal loads is with strain gages on the airframe static test article. Consider applying unique conditions on the static test article to better represent fatigue loading conditions. Loads applied in the static test should conservatively represent actual flight load conditions.

c. For fatigue evaluation by analysis, an applicant should validate the internal loads used in the analysis. In validating the internal loads, the applicant should validate the component's total stress field, including magnitude and orientation. One acceptable method of validating the internal loads is with strain gages on the airframe static test article. Consider applying unique conditions on the static test article to better represent fatigue loading conditions. Loads applied in the static test should conservatively represent actual flight load conditions.

d. Most modifiers and applicants for supplemental type certificates (STCs) evaluate the effect of their modification using a comparative fatigue analysis. In this case, it is usually sufficient for an applicant to substantiate that the loads and stresses used in their analysis conservatively account for the effects of their modification. The substantiation should incorporate the validation of stress analysis methods used for static strength substantiation.

2-10. How do I substantiate a fatigue safe-life with full-scale fatigue tests?

Full-scale fatigue tests are the most reliable method for establishing the fatigue performance of a structure. A full-scale test may use a complete, or nearly complete, airframe with flight, ground, and pressurization loads applied to the entire structure. A full-scale test may also be a series of separate tests of the individual major subassemblies of the airframe, such as the complete wing assembly, the empennage assembly, and the pressurized cabin. In testing major subassemblies, you should simulate the loads applied from any omitted structure and any fretting that would have been present with the omitted structure. For example, in testing the pressurized cabin, accurately simulate the loads and deflections, including assembly preload, imparted on the cabin structure by the wing or empennage.

2-11. What should I include in the test plan for my full-scale fatigue tests?

Your test plan should include at least the following items:

a. Description of the structure you will test. Describe whether you will test a complete airframe or separate major assemblies. Your test plan should include an analysis of the loading effects of any omitted structure and how you will simulate the loads imparted on the test article from these omitted structures. You may omit secondary structures from the full-scale test article provided the omitted structure does not contribute to the fatigue performance of the primary structure. Pay particular attention to the effects of fretting of any secondary structure you plan to omit.

b. Description of the test apparatus. Describe how the test article will be constrained and what effects the attachment to the test fixture will have on distributing internal loads in the test article. Describe how the test loads will be applied. The test set-up should be capable of accurately controlling the application of test loads. The set-up should record the time history of applied test loads.

c. Validation of internal loads. See paragraphs 2-8 and 2-9 of this AC.

d. Comparison of predicted internal loads and applied fatigue test loads. Make sure that the applied fatigue test loads produce internal loads that conservatively represent the predicted internal loads. You could show this with calibrated strain gages on the fatigue test article.

e. Description of loading sequences. The most realistic method of applying fatigue test loads is a random application of loads on a flight-by-flight basis. However, you may apply test loads in ordered loading blocks. The block length should be no greater than the number of flight hours (unfactored) that can be repeated ten times during the expected life. The sequence of loads should be from low to high to low within each block and use at least six load levels to describe each block. You should apply the ground-air-ground cycles either individually or in frequent blocks.

f. Description of truncated and clipped loads.

(1) The highest load level applied in the test should not exceed limit load or the load level that will be equaled or exceeded ten times in the life of the part. The magnitude of these high loads should be reduced, or clipped, down to the load level that will not be equaled or exceeded ten times in the life of the part. The intent of this precaution is to prevent the development of localized yielding.

(2) Your test loads should extend to the lowest level that causes significant fatigue damage unless you otherwise account for such loads. To expedite testing, you may omit, or truncate, lower level loads and simulate the fatigue damage resulting from these loads by applying a limited number of loads at higher load levels. However, enough cycles must be applied to account for the effects of fretting.

(3) Validate the load truncation level. You may do this with either coupon tests or analysis using data obtained from fatigue tests of similar structures.

g. How, when, and where you will inspect the structure.

(1) You should periodically inspect the test article to check for the development of detectable cracks. Use an analytical fatigue analysis of your structure to help determine the locations where cracks are likely to develop. Start inspections well before the predicted life of the structure. Later inspections should be frequent enough to allow an accurate estimate of the time of crack development. Your test plan should define the type of inspection and equipment required to perform the inspection. This will insure familiarity and proper calibration of the inspection equipment before the start of the test.

(2) Consider a teardown inspection of the structure after completion of the test. A teardown inspection often reveals unpredicted fatigue problems and can be valuable in addressing continuing airworthiness issues in the future.

h. Describe your definition of a completed test. The detection of a fatigue crack is not necessarily the end of a fatigue test.

(1) When testing a complete wing assembly, if a crack develops on one side, you may want to continue the test to determine when a crack develops on the other side of the wing. You would then compute the safe-life based on two test results. Continuation of the test in this circumstance is acceptable provided the failed structure does not affect performing the test on the other wing.

(2) Some applicants choose to repair the cracked structure and continue the test. Consider the effects of any repair on the distribution of internal loads and the associated effects on fatigue performance in the repaired area. We use the initial crack detection to determine the safe-life. However, applicants use the information gained from continuing the test to provide a greater understanding of the fatigue performance of the structure and as a database in addressing continuing airworthiness issues. You should obtain concurrence from the responsible Aircraft Certification Office (ACO) before installing any repair on the test article.

(3) Decide before you start the fatigue test when you will stop if no detectable crack is found. Some applicants set a "design life goal." Once the fatigue test article reaches this goal ("design life goal" multiplied by the appropriate scatter factor) without a detectable crack, they choose to stop the test at that point. Other applicants choose to let the test continue and set the safe-life as high as the test results will allow.

i. Approval of test plan and conformity inspection. The responsible ACO should review and approve your test plan. The FAA must also perform a conformity inspection of the test article and test setup.

2-12. Do I need to complete the test before the FAA will issue the type certificate (TC)?

a. The FAA recognizes the significant time often required to complete a full-scale fatigue test; therefore, the FAA may issue a type certificate before the applicant completes the full-scale fatigue test. In this instance, the safe-life specified in the limitations section of the Airplane Flight Manual (AFM) should be based on one of the following:

(1) The safe-life shown by an on-going test at the time of TC issuance. The safe-life should be computed as shown in the following paragraphs.

(2) The safe-life substantiated by a fatigue analysis. Your analysis should follow the guidance on fatigue analysis included in paragraphs 2-24 through 2-26 of this AC. Your analysis may consider information gained from the on-going fatigue test but should not be a prediction or extrapolation based solely on the on-going test.

b. Your certification plan, compliance checklist, and methods of compliance document should discuss how to determine the safe-life for TC issuance. In developing your certification plan, consider that a revision to the limitations section of the AFM may be all that is required to document a safe-life greater than that specified at the time of TC issuance. However, for a shorter safe-life, an airworthiness directive (AD) may be necessary.

2-13. How do I calculate the fatigue safe-life using full-scale fatigue test results?

a. You can calculate the safe-life from the results of a full-scale fatigue test using the following equation:

$$\text{The Safe-Life} = \text{The mean test life} \div \text{The scatter factor} \qquad \textit{Equation 1}$$

b. The following two paragraphs discuss the calculation of the mean test life and the purpose and derivation of the scatter factor.

2-14. How do I calculate the mean test life?

a. The calculation of the mean test life might seem a simple issue. However, the statistical distribution of fatigue life follows a log-normal statistical distribution; therefore, we should not use the arithmetic mean. We should instead calculate the mean based on the logarithms of the test lives. You may use the following two equations to calculate the mean test life based on the logarithms of the individual specimen test lives.

$$\log_{10}(mean\,test\,life) = \frac{1}{n}\sum_{i=1}^{n}\log_{10}(test\,life_i) \qquad \textit{Equation 2}$$

$$mean\,test\,life = 10^{\log_{10}(mean\,test\,life)} \qquad \textit{Equation 3}$$

b. The following example calculation shows the use of Equations 2 and 3.

Given: Specimen 1 test life: 100,000 hours

Specimen 2 test life: 200,000 hours

Using Equation 2:

$$\log_{10}(mean\,test\,life) = \frac{1}{2}[\log_{10}(100{,}000) + \log_{10}(200{,}000)]$$

$$= \frac{1}{2}[5.0 + 5.30103] = 5.150515$$

Using Equation 3:

$$mean\,test\,life = 10^{5.150515} = 141{,}421\ hours$$

c. The mean test life using the logarithmic mean (141,421 hours) is less than you would calculate using the arithmetic mean (150,000 hours).

2-15. What are the scatter factors for full-scale fatigue tests?

a. The purpose of the scatter factor is to account for the statistical variability inherent in the fatigue performance of built-up structures. The equation for the minimum scatter factor, given below, assumes a log-normal statistical distribution of fatigue life.

$$Scatter\ Factor_{FST} = 10^{Z_p \sigma \sqrt{\frac{n_s+1}{n_s}}} \qquad \textit{Equation 4}$$

where: FST denotes full-scale test.

Z_p is the normal distribution variate for the specified probability of a detectable crack-free safe-life.

σ is the standard deviation of the population of fatigue test lives. For use in this equation, σ , is a percentage of the log of the mean test life for the built-up structure.

n_s is the number of fatigue specimens tested.

b. Reference 3 specified a value for $Z_p = 3.511$. This value for the normal distribution variate equates to a probability of 99.9777 percent. For any individual structural component in a small airplane, the FAA's specified probability that the component will reach its safe-life without a detectable fatigue crack is 99.9777 percent.

c. The value for the standard deviation of the fatigue test life is usually taken from historic data. This is due to the high cost of testing sufficient numbers of full-scale test articles to develop an accurate value for the standard deviation. For built-up aluminum structures, Reference 3 specified a value for $\sigma = 0.14$. The following table summarizes the commonly accepted values for the standard deviation for various structural metals.

Table 1. Standard Deviation of Log Fatigue Life for Built-up Structures Constructed of Common Structural Metals (Reference 5)	
Base Material of Built-up Structure	σ, Standard Deviation (as percent of log life)
Aluminum alloys	14%
Steels (f_{tu} = 100 – 200 ksi)	17%
Steels (f_{tu} = 200 – 300 ksi)	25%
Titanium alloys	20%

d. These values for the standard deviation are for complete, built-up structures. Values for the standard deviation given in material handbooks, such as Reference 6, are from simple coupon tests and should not be used to determine full-scale fatigue test scatter factors.

e. The table below summarizes the required full-scale test scatter factors for aluminum structures. You may recognize the commonly known scatter factor of 4.0 from Reference 3. If your structure is made of a metal other than aluminum, you need to use Equation 4 to calculate the correct scatter factor.

Table 2. Full-Scale Fatigue Test Scatter Factors for Aluminum Structures For Use with Safe-Life Structures		
	Number of Test Specimens	Required Scatter Factor
Probability of Detectable Crack-Free Safe-Life 99.9777% (Z_p = 3.511)	1	4.96
	2	4.00
Standard Deviation of Log Fatigue Life, σ 0.14 (14%) for Aluminum Structures	3	3.70
	4	3.54

2-16. What about statistical confidence intervals on full-scale fatigue test results?

a. Certification authorities, military specifications, and industry standards often require the use of confidence intervals in the analysis of test results. A common example is the requirement for material strength allowables to be given as an "A-basis," defined as 99 percent probability of survival with 95 percent confidence. A common requirement for fatigue is 99.9 percent probability of survival with 95 percent confidence. The confidence, or confidence interval, is a statistical method of estimating the possible range for the true mean of a population based on the test results of only a few specimens. The confidence interval gives a lower and upper boundary for the likely value of the population mean. In establishing design allowables, or design lives, we are interested in the lower boundary for the mean.

b. In the statistical model we have used to determine the scatter factor, each individual fatigue critical component has a probability of reaching its safe-life without a detectable crack of 99.9777 percent (with no confidence interval on the mean). This probability is equivalent to the following:

(1) With two test results (scatter factor = 4.0): 99.9 percent probability with 91.3 percent confidence (or 99.822 percent probability with 95 percent confidence).

(2) With one test result (scatter factor = 4.96): 99.9 percent probability with 93.95 percent confidence (or 99.868 percent probability with 95 percent confidence).

c. The scatter factor model we are using provides probabilities nearly equivalent to alternate fatigue requirements that use confidence intervals. However, be aware that the CAAs of other countries often require the use of a somewhat larger scatter factor that provides 99.9 percent probability with 95 percent confidence.

2-17. What counts as one test specimen, and what counts as two test specimens?

a. Structural components or details that occur more than once in a test article may count as two or more test specimens provided the details have the following:

(1) Same structural function.

(2) Same geometry.

(3) Same materials.

(4) Same stress level, both flight and ground.

(5) Same stress concentration factor.

(6) Same flight and ground load spectrum.

b. Testing of a complete wing, both the left and right side, generally counts as two test specimens. Similarly, certain frame, stringer, cutout, and other details that occur symmetrically in a complete pressurized cabin may count as two or more test specimens.

c. Examples of testing that might count as only one test specimen include details in the vertical tail, the keel beam in a pressurized cabin, and non-symmetric details in the pressurized cabin, such as entrance and emergency exit doors.

d. A 1996 policy letter further clarifies when a test article counts as only one test specimen. An applicant asked, “Does the fact that each fuselage fitting assembly consists of two parts, back to back, mean that the time to initial failure of the one assembly counts as 2 (two) results?” Our answer was, “If each part can independently sustain ultimate load for the assembly, then our answer is yes. If each part cannot sustain ultimate load, and safe-life certification is being done, then we consider the fitting assembly as having failed when one part has failed, and our answer is no.”

2-18. Can I use a full-scale test scatter factor lower than that obtained from Equation 4?

You should not use a full-scale test scatter factor less than that obtained from Equation 4 unless you substantiate compliance to fail-safe design requirements. Reference 1 allowed reduction of the scatter factor from 4.0 down to 3.0 with an inspection program based on the results of testing that today we recognize as crack growth testing. Industry practice and FAA regulation, policy, and guidance have evolved considerably since the guidance provided in Reference 3. Today, we do not allow a type certificate (TC), amended type certificate, or supplemental type certificate

(STC) program to rely on inspections alone, without demonstrated residual strength, as a means of maintaining continued airworthiness. Use of scatter factors less than those obtained from Equation 4 without the compensating features of fail-safe design features or significant residual strength capability does not provide adequate structural reliability.

2-19. What about the '3-sigma' method of computing the safe-life?

a. Some industries and manufacturers use a method of safe-life computation known as the "3-sigma" method, wherein the safe-life is determined by subtracting three standard deviations from the mean fatigue test life. While this method is in common use, it will not provide the same probability of a detectable fatigue crack at the safe-life as our scatter factor model.

(1) Our scatter factor model uses 3.511 standard deviations to obtain the required probability, compared to 3.0 standard deviations in the '3-sigma' method.

(2) Our scatter factor model incorporates an adjustment to account for the number of fatigue specimens tested.

(3) Our scatter factor model incorporates values for the standard deviation of fatigue life based on historically accepted values for the standard deviation.

b. To ensure standardization of fatigue safe-life evaluations, applicants should use the scatter factor model approach to compute the safe-life based on full-scale test results.

2-20. How do you substantiate a fatigue safe-life with component tests?

a. You can also substantiate the safe-life of a structure with component tests that use less than the complete structure or major subassembly. Examples of component tests include testing of the wing spar and testing of the wing-center section joint. When testing at the component level, take great care to ensure that the test stresses are valid and that you have included all critical details in your test.

b. Review the design, stress analysis, static test, strain surveys, tests of similar structural configurations, and service experience to make sure that you have identified all structural details that are significant for fatigue. It is acceptable to use component specimens that 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. Special attention should be given to areas of stress concentration such as holes, joints, changes in section, sharp corners, and rough surface finish.

c. Component testing is usually limited to simple and determinate structure that is free of stresses due to eccentricities, assembly preload, and so forth, unless enough adjacent structure is included to ensure valid test stresses.

d. You should validate the test stresses, including the peak local stresses, by strain survey comparisons with complete structures for the test loading conditions. Validation of the test stresses is necessary to account for omitted structure and to gain an accurate knowledge of the stress state in the full-scale component. Additional guidance on validation of the internal loads for component testing is given in paragraphs 2-8 and 2-9 of this AC.

2-21. What loads do I apply in a component test?

There are two types of loadings used in component tests, as follows:

a. Spectrum loading, similar to full-scale testing, intended to accurately simulate operational loadings and stress distributions.

b. Constant amplitude, or S-N testing. This testing is intended to define the mean S-N curve for use in a cumulative damage analysis. Although the loads are constant amplitude, the stress distributions in the test specimen and the magnitudes of the applied loads should accurately simulate the stress distributions in the full-scale structure and duplicate the range of expected operational loadings.

2-22. For component testing, how many specimens do I need to test?

a. If you test using spectrum loading, you should test at least three specimens representative of the most critical location of each type of detail.

b. If you are testing using constant amplitude, or S-N testing, test a sufficient number of specimens to define the mean S-N curve. Chapter 9 in Reference 6 provides guidance on methods of defining the mean S-N curve.

2-23. For component testing, what scatter factor should I use?

a. If your test uses spectrum loading, the scatter factor and safe-life are calculated similarly as for a full-scale test. The following equation defines the minimum scatter factor for component testing with spectrum loading.

$$Scatter\ Factor_{CT} = 1.5 \times ScatterFactor_{FST} \qquad \textit{Equation 5}$$

where: CT stands for component test
FST stands for full-scale test

b. The following example calculation shows the use of Equation 5.

Given: Three (3) component aluminum specimens

From Equation 4: Scatter Factor_{FST} = 3.70

Using Equation 5: Scatter Factor_{CT} = 1.5 x 3.70 = 5.55

c. A scatter factor larger than that given by Equation 5 may be necessary if the component test specimen and test loading do not accurately simulate operational loadings, stress distributions, and the full-scale structure.

d. For constant amplitude, or S-N testing, the component test results are used to develop the mean S-N curve for use in a cumulative damage analysis. The scatter factors, or statistical adjustments, applied to cumulative damage analysis are discussed in the following paragraphs.

2-24. How do you substantiate a fatigue safe-life with analysis supported by test evidence?

a. You may substantiate the fatigue safe-life of a structure by fatigue analysis supported by test evidence. The analysis method most commonly used is the Palmgren-Miner hypothesis, also known as Miner’s Linear Cumulative Damage Theory.

b. The basic philosophy of Miner’s Theory is 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 that stress level. This ratio is called a cycle ratio and is used to measure damage. If you apply various levels of stress amplitude to the structure, the total damage is the sum of the cycle ratios. Failure is predicted to occur when the sum of the cycle ratios is equal to one, or the following:

$$\sum_{i=1}^{k} \frac{n_i}{N_i} = 1.0 = \frac{n_1}{N_1} + \frac{n_2}{N_2} + \ldots\ldots + \frac{n_k}{N_k} \qquad \textit{Equation 6}$$

where: n_i is the fatigue damage due to the applied number of cycles at a certain stress level

N_i is the number of cycles to failure at that stress level.

k is the number of intervals the stress spectrum is divided into. Typically, the stress spectrum is divided into increments corresponding to 0.2g.

c. Reference 7 provides a sample calculation using Miner's Theory. The Small Airplane Directorate has developed a computer program based on Miner's Theory. The Directorate makes this software available to airframe Designated Engineering Representatives (DERs). For additional information on the software, please see Reference 8.

2-25. What are some concerns with substantiation by fatigue analysis?

a. Accuracy of the cumulative damage hypothesis. Several references, including Reference 9, compare spectrum test results to cumulative damage predictions. These references show that fatigue failure may occur between damage summations between 0.5 to 2.0, depending on various loading parameters. In particular, severe usage, including aerobatic and agricultural usage, may significantly reduce the accuracy of the cumulative damage calculation.

b. Adequacy of S-N data.

(1) Reference 6, formerly MIL-HDBK-5, includes the following disclaimer regarding handbook S-N data:

"The S-N or ε–N curves may not apply directly to the design of structures because they may not take into account the effect of specific concentration associated with reentrant corners, notches, holes, joints, rough surfaces, and other similar conditions, which are present in fabricated parts."

(2) Handbook S-N data may also not be directly applicable to your structure due to small differences in the material used to develop the handbook curves and the material used in your structure. Also, data for the exact stress concentration factors in your structure may not be available in the handbook data.

(3) Applicants may address these issues by adjusting the S-N curves to account for differences between the actual structure and the structure, or coupon, used to develop the S-N curves. Applicants should adjust the S-N curves for differences in geometry and stress, including eccentric loads and net stress field. You may use component testing or previous testing experience with similar structural details to select and adjust the most applicable S-N curves. In determining the applicability of S-N data, similar structures are those:

(a) Constructed of the same material (same form, same heat treat).

(b) Constructed with the same or similar manufacturing methods.

(c) With the same or similar structural function and geometry.

(d) With the same or similar stress concentration factor.

(e) With the same or similar stress level and loading spectrum.

(f) With the same or similar principal stresses, for example axial and bending loads.

For complex details, including joints and fittings, you should not use S-N data without first adjusting the data for applicable component level or full-scale test results.

c. Accuracy in determining the stress concentration factor, K_t. Identify all areas of stress concentration in the structure and determine the magnitude of the stress concentration factor, K_t, associated with each of these details. Accurate determination of K_t is necessary to make sure that you are using the correct S-N data in your fatigue analysis. You may determine stress concentration factors for some details, including open holes, radii, filets, and so forth, from handbook references. Stress concentration factors for more complex details, including fittings and joints, require detail analysis. In this case, the determination of K_t should include consideration of eccentric loadings and the bearing and bypass stresses at each bolt or rivet in a joint.

d. Accuracy in determining the stress level. Since you will not conduct full-scale fatigue tests or spectrum loading component level tests on your structure, the quality, detail, and completeness of your stress analysis is important. In fatigue substantiation by analysis, we rely completely on the stress analysis to identify all fatigue critical details. Applicants should follow the guidance in paragraphs 2-8 and 2-9 of this AC to validate the stresses used in a fatigue analysis.

2-26. What scatter factors do I use in a fatigue analysis?

The scatter factor you use in a fatigue analysis is larger than the scatter factors you would apply to full-scale and component level test results. This is due to the uncertainties in a fatigue analysis outlined in the previous paragraph. The scatter factor used in an analysis depends on the type of metal, statistical basis, and applicability of the S-N data used in the analysis.

a. For aluminum structures, you may use the following:

(1) You may use the S-N data provided in Appendix 2. This S-N data is applicable to conventional built-up aluminum structure with no fittings (other than continuous splice fittings), no parts with high residual stresses, no unique structural features, and no stress concentrations greater than $K_t = 4$. This data is carried over from the guidance provided in Reference 3. Use of this S-N data, combined with a scatter factor of 8.0, has provided satisfactory service experience.

(2) You may use proprietary, handbook, or material supplier mean S-N curves adjusted for any differences between the actual structure and the structure or coupon used to develop the base S-N curves. Test adjusted data is recommended for lugs, joints, and fittings. In this case, you may use a minimum scatter factor of 8.0.

b. Adjustments to the typical scatter factor of 8.0 must be made for the following:

(1) Structures made out of metals other than aluminum.

(2) Structures, including aluminum, whose S-N curves have a statistical basis other than the mean (for example, curves based on 99 percent probability of survival). *(Note: The S-N data provided in Reference 6 represent the mean S-N curve, with no adjustments for confidence intervals.)*

(3) Any adjustments to the scatter factor must be coordinated with the Small Airplane Directorate Standards Office.

2-27. How can I extend the safe-life, or retirement life, of my airplane?

a. Extensions of the safe-life are accomplished either as an amended type certificate (TC) by the TC holder or as a supplemental type certificate (STC) either by the TC holder or an STC applicant. In formulating a life extension program, applicants should consider the following:

(1) Inspection programs alone are not sufficient to extend the safe-life of an airplane.

(2) The safe-life of a structural component is usually set to the detail that has the shortest fatigue life. An applicant should address the possibility that other structural details or components may have nearly as short a fatigue life as the detail you are extending the life of. To avoid the development of unforeseen fatigue cracks in other parts of the structure, you should complete a fatigue evaluation of the entire structure.

b. One method of extending the safe-life is a thorough re-evaluation of the fatigue performance of the structure.

(1) A re-evaluation of the fatigue performance requires detailed knowledge of the data, including the loading spectra, stress levels, stress concentration factors, and S/N curves, used to substantiate the original type design. STC applicants are at a particular disadvantage in fatigue life extensions since they typically do not own or possess the type design data.

(2) It may be difficult for you to substantiate a life extension through analysis alone. An analytical approach would require you to show that the type design analysis was overly conservative or that your analysis, including assumptions and data, is more accurate than that used in the type design substantiation.

(3) A full-scale fatigue test of the airplane's structure is the preferred method of extending the original safe-life, especially if you know that the original design was substantiated by analysis. In this case, follow the guidance provided in this AC for full-scale fatigue testing, including mission profile development, selection of load spectra, and validation of internal loads. Consider in-flight measurements of loads and stresses to further justify the life extension.

(4) In your fatigue test, you should use an airframe with zero flight time. This is necessary since we do not know the usage history of any structural member that has accumulated flight time. It is also possible that an airframe that has accumulated flight time may have experienced an overload event that may have altered the fatigue performance of the structure.

c. One life extension method that applicants have used successfully has been to provide fail-safe design features to an existing safe-life design. In this approach, an applicant completes all of the following:

(1) Incorporates redundant structure into the existing design. For example, an external spar strap that provides the required residual strength capability.

(2) Inspects the structure for cracks. Any parts with existing cracks are replaced.

(3) Oversizes existing bolt and rivet holes to remove any small cracks not detected during inspection.

(4) Develops an inspection program for the fail-safe and redundant structure.

d. Life extensions must be approved by the responsible Aircraft Certification Office (ACO) and coordinated with the Small Airplane Directorate Standards Office.

CHAPTER 3. FAIL-SAFE DESIGN

3-1. What is a fail-safe design?

A fail-safe design is a design that retains its required residual strength after the failure or partial failure of a principal structural element. A fail-safe design typically consists of the fail-safe component or primary structural element, and a redundant or backup structural element. A fail-safe design is, therefore, often said to be a redundant design or a multi-load path design. Failure in a multi-load path design is the inability to support the required residual strength loads or the inability to prevent excessive deformation that interferes with continued safe flight and landing. Follow the general guidance in sections 2-4, 2-7, 2-8, 2-9, and 2-11 of this AC for developing the loading spectra, mission profile, and test plan for evaluating your fail-safe design.

3-2. What is the history of fail-safe design?

a. Fail-safe design has been an option available to applicants since the first specific fatigue requirement in Amendment 3-2 to CAR 3. A fail-safe evaluation generally includes the following:

(1) Establish the components you will make fail-safe. This includes identifying which primary structural elements will have redundant load paths.

(2) Define the loading conditions, including the magnitude of external loads. This step includes determining the internal loads in the primary and redundant structure in both the pristine and damaged conditions.

(3) Determine the extent of damage for which you will design the fail-safe structure.

(4) Conduct structural tests and analyses to substantiate that you have achieved the fail-safe design objective.

(5) Establish inspection programs aimed at early detection of fatigue damage.

b. In the more than 45 years since CAR Amendment 3-2 first introduced the fail-safe option for addressing fatigue requirements, the redundant structure provided by fail-safe design has prevented otherwise catastrophic failures. However, there are well-documented cases in which a design thought to be fail-safe was not, and the design failed to prevent a catastrophic failure. Several references, including References 10 and 11, identify the following potential problems of designs mistakenly thought to be fail-safe:

(1) Potential loss of fail-safe attributes with time due to normal fatigue wear-out.

(2) Difficulty in making an accurate prediction and validation of failure modes.

(3) Incorrect assumptions that a design is sufficiently and consistently self-annunciating.

(4) Loss of type design strength due to inadequate crack detection. If a crack or other damage is not found by inspections or is not so obvious as to be detected before further flight, the capability of the structure may be below the design limit and ultimate load capability.

(5) Inadequate residual life with obvious damage present. Redundant structure may not have sufficient safe-life to ensure that damaged structure will be found by inspections.

3-3. How can I assure that my design complies with the fail-safe design requirements?

a. The historical guidance provided in Reference 3 described the importance of repeated inspections in maintaining the continued airworthiness of a fail-safe design. Detecting fatigue cracks before they become dangerous is the ultimate control to ensure the fail-safe characteristics of flight structure and pressurized cabin. For a fail-safe design, applicants should develop an inspection program capable of detecting fatigue cracks and other partial failures. Applicants should provide enough guidance information to aid operators in establishing the frequency and extent of the repeated inspections of the critical structures or critical areas. Include instructions for these inspections, including inspection schedule and inspection methods, in the information required for § 23.1529.

b. You could develop an inspection program based on:

(1) A service history based approach. For guidance on service history based inspections, see AC 91-60.

(2) A damage tolerance approach.

(3) A hybrid approach in which the inspection threshold equals the safe-life determined using Equation 1 and repeat inspection intervals based on crack growth analysis.

3-4. What are some design considerations in achieving a fail-safe design?

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

b. Use of composite (more than one element) duplicate structures so a fatigue failure occurring in one-half of the composite member will be confined to the failed half. Also, the remaining structure will still possess the load carrying ability required by § 23.571 and § 23.572.

c. Use of backup structure wherein one member carries the entire load, with a second member available that can assume the extra load if the primary member fails.

d. Selection of stress levels and materials with low notch sensitivity (particularly for components with high stress concentration) that provide a controlled slow rate of crack propagation combined with a high residual strength after initiation of cracks.

e. Arrangement of design details to allow easy detection of failures in all critical structural elements before the failures can become dangerous or result in a loss of strength below that required by § 23.571 and § 23.572. You should also arrange design details to allow replacement or repair.

Note: The last two items are examples of good design practice and improve fail-safe design concepts, but they cannot be used alone to achieve fail-safe design.

3-5. What is a Principal Structural Element?

A principal structural element (PSE) is an element that contributes significantly to carrying flight, ground, or cabin pressurization loads, and whose integrity is essential in maintaining the overall structural integrity of the airplane. Typical examples of PSEs are as follows:

a. Wing, horizontal stabilizer, vertical fin, canard, forward wing, winglets/tip fins:

(1) Fixed surface, stabilator, or trimmable stabilizer attachment fittings;

(2) Integrally stiffened plates;

(3) Primary fittings;

(4) Principal splices;

(5) Skin or reinforcement around cutouts or discontinuities;

(6) Skin-stringer combinations;

(7) Spar caps; and

(8) Spar webs.

b. Pressurized cabin:

(1) Circumferential frames and adjacent skin;

(2) Pressure bulkheads;

(3) Cockpit window posts;

(4) Skin and any single frame or stiffener element around a cutout;

(5) Skin or skin splices, or both, under circumferential loads;

(6) Skin or skin splices, or both, under fore and aft loads;

(7) Skin around a cutout;

(8) Skin and stiffener combinations under fore and aft loads;

(9) Door frames, skins, and latches; and

(10) Window frames.

3-6. How much fatigue damage should I consider in my fail-safe evaluation?

Each particular design should be carefully assessed to establish appropriate damage criteria. As discussed in Reference 10, you should not rely on the detection of an "obvious partial failure." Instead, you should consider fatigue damage in terms of the complete failure of the single element involved. Typical examples of the fatigue damage you should consider are outlined below:

a. Skin cracks in splice joints and those emanating from the edge of structural openings or cutouts that can be readily detected by visual inspection of the area.

b. Failure of one element where dual construction is used in components such as spar caps, window posts, window or door frames, and skin structure.

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

d. Failure of one element of primary attachments, such as wing and empennage fixed surface, or stabilator attach fittings.

e. Excessive loss of stiffness under load as evidenced by excessive deformation.

f. Failure of skin, frames, stringers, and pressure bulkheads in pressurized cabins.

3-7. How do I address inaccessible areas?

Designers should make every reasonable effort to ensure all principal structural elements can be inspected, as required by § 23.611. If inaccessible or blind areas are unavoidable, place emphasis on determining crack propagation and residual strength of the particular fatigue-damaged structure to ensure continued airworthiness of the structure with reasonable inspection methods and controls by the operator. Also consider providing 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. Pay particular attention to corrosion prevention in inaccessible areas.

3-8. How do I account for dynamic effects?

Apply the dynamic magnification factor of 1.15, required by §§ 23.571 and 23.572, to all loads, including pressure cabin loads. You may also make fail-safe cuts under load to show a different value for the factor, or you may show that the dynamic effects are negligible by comparison to dynamic test data from a similar structure.

3-9. How do I test principal structural elements?

a. The nature and extent of tests on either a complete structure or portions of the primary structure, or both, will depend upon previous experience with similar types of structures 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. If 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. The required residual strength of the remaining structure is a static ultimate load factor of 75 percent of the critical limit load factor at V_C, per §§ 23.571 and 23.572.

b. You may apply the fail-safe loads to the structure before severing and omit the 1.15 factor. In this case, the test specimen and test fixture must be carefully designed to ensure that the correct dynamic effects are obtained. If you are testing distributed members, such as a sheet-stringer combination or an integrally stiffened tension skin, make a cut to represent an initial crack in the element under test. If there is no failure, you may increase the length of the cut with the fail-safe load applied until one of the following occurs:

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

(2) The crack propagation rate decreases due to redistribution of load paths; or

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

c. The simulated cracks should be as representative as possible of actual fatigue damage. If 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. If saw-cuts in primary structure are used to simulate sharp fatigue cracks, sufficient evidence should be available from element tests to indicate equivalent residual strength. If it is necessary to simulate damage at joints or fittings, bolts may be removed to simulate the failure if this condition represents an actual failure.

d. Failure is the inability of the redundant structure to support the required residual strength loads or the inability to prevent excessive deformation that would interfere with safe flight and landing.

3-10. Can I use analysis to substantiate a fail-safe design?

a. Sometimes you can substantiate fail-safe characteristics analytically. You may use an analytical approach when the structural configuration involved is similar to one already verified by fail-safe tests, whether conducted on a previously approved type design or on other similar areas of the design currently being evaluated.

b. You may also use the analytical approach when conservative failures are assumed. However, the failure must be detected long before the critical crack length is approached, and the margins of safety resulting from the analysis must be considerably more than the fail-safe residual static strength level. In any such analysis, include the 1.15 dynamic magnification factor unless it can be shown that this factor is not required.

3-11. How do I select the critical areas?

Paragraph 3-5 of this AC lists typical single principal structural elements and detail design points that require fail-safe evaluation. The process of determining where you should simulate fail-safe damage in an element, such as a wing spar cap or fuselage frame, requires use of sound engineering judgment that considers various factors, such as the following:

a. Conducting an analysis to locate areas of maximum stress and a 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 location in an element (such as a spar cap) where the stresses in adjacent elements (such as the spar web or wing skin) would be the maximum with the spar cap failed.

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

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

h. Locating areas susceptible to operational damage, such as foreign object damage, corrosion, and so forth.

CHAPTER 4. DAMAGE TOLERANCE EVALUATION

The FAA has not developed guidance unique to part 23 damage tolerance evaluations. For damage tolerance evaluations, use the guidance provided in AC 25.571C, Damage Tolerance and Fatigue Evaluation of Structure. Follow the general guidance in sections 2-4, 2-7, 2-8, 2-9, and 2-11 of this AC for developing the loading spectra, mission profile, and test plan for evaluating your damage tolerant design.

4-1. Are there any differences between the part 23 and part 25 damage tolerance regulations?

There is one significant regulatory difference between part 23 and part 25. Part 25 includes a discrete source damage requirement. There is no requirement for substantiation of discrete source damage capability in part 23. However, be aware that a discrete source damage analysis might be necessary to show compliance to rotorburst requirements.

CHAPTER 5. ROTORBURST REQUIREMENTS

5-1. What are the requirements for turbine or turbopropeller installations in small airplanes?

a. Compliance with 14 CFR, part 23, § 23.903(b)(1) requires the applicant to take design precautions to minimize the hazards to the airplane, including the airframe, due to an engine rotor failure. A Small Airplane Directorate policy letter dated April 7, 1989, defines "minimize" in the context of § 23.903 as "to reduce to a minimum, decrease to the least practical amount. The 'least practical amount' obviously has to include consideration of what is attainable with current technology and materials."

b. Industry practice for part 23 turbine-powered airplanes is to use discrete source damage analysis to evaluate design precaution options. Applicants have used the discrete source damage evaluations to select the most effective, appropriate design precautions and to substantiate that the selected design precautions minimize the rotorburst hazard to the least practical amount. The design precautions selected by these applicants to show compliance to § 23.903(b)(1) have provided sufficient residual strength capability to demonstrate compliance with the discrete source damage requirements in 14 CFR, part 25, § 25.571(e)(3).

5-2. I thought there was not a discrete source damage requirement in part 23?

a. Part 23 does not include a discrete source damage requirement in the fatigue regulations. However, small airplane applicants have used, and the FAA has accepted, discrete source damage analysis as a quantitative method of substantiating that selected design precautions minimize the rotorburst hazard to the least practical amount.

b. AC 20-128A, Design Considerations for Minimizing Hazards caused by Uncontained Turbine Engine and Auxiliary Power Unit Rotor Failure, provides a means, but not the only acceptable means, of showing compliance to § 23.903(b)(1). The AC uses discrete source damage criteria to include airframe hazards in a safety assessment and risk analysis. The correct interpretation of the AC is that the discrete source damage evaluation is required for all categories of airplanes. If you choose to show compliance using AC 20-128A, include airframe hazards, via discrete source damage analysis, in the rotorburst safety assessment and risk analysis.

CHAPTER 6. FLIGHT WITH KNOWN CRACKS

6-1. What is the policy for operational flight with known cracks?

It is the Small Airplane Directorate's policy to not allow continued operational flight with known cracks in primary structure. An airplane with a known crack in the primary structure no longer meets its type design and may no longer possess its type design strength.

6-2. Are there any exceptions to this policy?

a. The exceptions to this policy, to be taken on a case-by-case basis, are the following:

(1) Substantiation that the cracks are not in primary structure.

(2) Substantiation of the ability of single load path structure with known cracks to carry ultimate loads. Only in unusual circumstances, such as the difficulty of an operator in obtaining replacement parts, will this be allowed. The time allowed for replacement or repair must be as short as practical.

(3) Substantiation that the cracks are in fail-safe structure. The ability to sustain ultimate load with the maximum permissible crack must be substantiated. Temporary repairs, such as "stop drilling," should be specified. The conditions of a temporary repair, and the time allowed before a permanent repair is required, may be less restrictive than for single load path structure.

b. The Aircraft Certification Office (ACO) responsible for the airplane's type certificate should make each specific approval for flight with known cracks. The ACO issuing the approval should coordinate the approval with the Small Airplane Directorate Standards Office.

6-3. What will the FAA consider before allowing operational flight with known cracks?

Continued operational flight with known cracks in a structure whose failure may be catastrophic, in general, is not permissible. However, there are certain unusual circumstances under which flight is permissible for a limited period of time with known cracks in the structure. Any approval for flight with known cracks should meet **all the following criteria**.

a. There must be an unusual need for allowing flight with known cracks. Unusual needs are those that arise from such things as the scarcity of parts or operator hardship. For example, unavailability of parts or tooling to support an immediate part replacement would qualify as an unusual need. Another would be the need for operational flight to a repair facility from an area where insufficient technical or maintenance resources are available to perform a repair. Deferral of corrective action should not be approached as either business as usual or as a matter of convenience to operators.

b. The level of safety provided by the certification basis must not be compromised by the allowance to operate with a known crack. At all times during the expected crack growth phase, the airplane must meet its limit and ultimate load requirements with the crack present. Part modification, replacement, or repair must be implemented if the structure's strength, with the crack present, is anticipated to fall below these load requirements before the next inspection of the structure. Specifically, this includes the following:

(1) Cracked structure must support limit load without detrimental permanent deformation (§ 23.305(a)).

(2) Cracked structure must support ultimate load for three seconds without failure (§ 23.305(b)).

(3) Airplanes certified as fail-safe should remain fail-safe with the damage present. This means that the cracked structure must be able to bear ultimate load and meet the fail-safe criterion specified in the airplane's certification basis.

(4) Any other certification requirement identified by the ACO as being potentially violated by repair deferral should be addressed.

(5) Single failure requirements remain in effect with the damage present (e.g., § 23.629 (g)(h)).

c. There should be a high degree of confidence in the evaluation used to justify the particular case. Regardless of the certification basis of the airplane, a fracture mechanics assessment must be performed for the cracked structure before further flight. The evaluation should determine that the material, location, and type of damage is such that a reliable fracture mechanics assessment can be performed. The following items are especially critical to a thorough evaluation:

(1) An investigation into the cause of the crack. This investigation should include a complete characterization of the crack location, size, and shape. The investigation should estimate the time of crack initiation and growth rate using microscopic mapping techniques. Identify the presence of any corrosion or manufacturing or material defect. Review the results of any previous inspections of the cracked part to find any indications of abnormalities. The investigation should also review the assumptions used in the cracked structure's design. This should include a comparison of the load spectra used in the type certification fatigue evaluation and actual usage, validation of internal loads, and verification of material properties.

(2) Static strength and fracture mechanics analysis. The fracture mechanics analysis must provide a high degree of confidence that the crack will not grow faster than the predicted rate with the expected usage until the next inspection, or until it is repaired. All residual strength substantiations should be shown by a conservative analysis, by test, or by test combined with analysis. Analysis is allowed only where experience shows that analysis is reliable.

(3) Inspection intervals.

(a) This assessment must establish a viable inspection program for the cracked structure. The inspection program should provide a high level of confidence that the crack will not be allowed to grow to a size that would compromise the safety provided by the certification basis. The probability of detection and reliability of the inspection must be high. The inspection method should be suitable for the situation, and the cracked structure should be adequately accessible. The inspection program, along with any other limitations, must be placed in the mandatory section of the airplane's maintenance manual for the period of time in which the cracked structure remains in operation.

(b) Inspection intervals should be conservatively determined based on approved crack growth data and a fracture mechanics based evaluation. The inspection interval is established by calculating the crack growth time from the detected crack size (or maximum of detected size range for non-visual) for the Non-Destructive Inspection (NDI) method used to detect critical crack size and dividing that number of flight hours or flights by a conservative value. Inspection intervals should provide for at least two opportunities to detect crack growth before the crack size reduces the capability of the structure below that provided by the certification basis.

(4) Temporary intervention. If possible, some intervention, such as stop drilling of the crack tip, should be implemented to slow crack growth (stop drilling is not considered a repair).

(5) Single-Load path structures. Flight with known cracks is strongly discouraged for structure where applied loads are eventually distributed through a single member, the failure of which would result in the loss of the structural capability to carry the applied loads. If cracking is found in this type of structure, the part must be replaced or repaired before further flight. Exceptions may be permitted only if all other criteria of this policy are met, stringent inspection requirements are specified, and a conservative maximum permissible crack size able to sustain ultimate load is specified. A combination of a conservative fracture mechanics analysis correlated to test results, or conservative fracture mechanics analysis with added scatter factors, must be performed to allow continued operation with cracks.

(6) Multiple-Load path structures. For redundant structures in which, with the failure of individual elements, the applied loads would be safely distributed to other load carrying members, adjacent structure must be inspected and be free of cracks or other damage. Various combinations of analysis and test, including that provided at the time of original certification, may be considered as ample substantiation.

(7) Structures with multiple-site damage, corrosion and stress corrosion cracks. Continued operational flight with known cracks in instances of multiple-site damage, cracks in the presence of corrosion, or stress corrosion cracks should be avoided.

c. The approval should provide a limited time for continued flight with known cracks. Under no circumstances can allowance for operation with known cracks be considered more than a temporary condition. The limited period must be based on the need for the deferral of the repair, replacement, or modification. Once the unusual need for the deferral has past, the deferral must be eliminated, and the part repaired or replaced.

d. The ACO approving flight with known cracks should coordinate each approval with the Small Airplane Directorate Standards Office. The approval to allow flight with known cracks must be clearly made, documented, and approved by the FAA before any flight with known cracks. The ACO responsible for the type certificate should make each specific approval for flight with known cracks in coordination with the Small Airplane Directorate Standards Office.

REFERENCES

1. FAA Report No. ACE-100-01, Fatigue Evaluation of Empennage, Forward Wing and Winglets/Tip Fins on Part 23 Airplanes, dated February 15, 1994.

2. DOT/FAA/CT-91/20 General Aviation Aircraft – Normal Acceleration Data Analysis and Collection Project, dated February 1993.

3. FAA Report No. AFS-120-73-2, Fatigue Evaluation of Wing and Associated Structure on Small Airplanes, dated May 1973.

4. FAA Memorandum, Method of Establishing Flight Load Spectra for Safe-Life Fatigue Analysis, dated March 25, 2005.

5. A Review of Fatigue Scatter Factors with Particular Reference to Civil Aircraft, M. B. Benoy, Civil Aviation Authority of Australia, July 1978.

6. DOT/FAA/AR-MMPDS-01, Metallic Materials Properties Development and Standardization, dated January 2003.

7. FAA Memorandum, Example Cumulative Damage Calculation for Safe-Life Fatigue Analysis, dated February 7, 2005.

8. User's Instructions for Computer Program to Calculate Fatigue Safe-Life (Unfactored) for Small Airplanes, dated March 1996.

9. Metal Fatigue, J. A. Pope (Editor), Chapman and Hall Ltd, dated 1959.

10. A Critical Review of Strategies Used to Deal with Metal Fatigue, by Bob Eastin, Federal Aviation Administration, Presented at the International Committee on Aeronautical Fatigue (ICAF) 2003.

11. Commercial Airplane Certification Process Study, An Evaluation of Selected Aircraft Certification, Operations, and Maintenance Process, Federal Aviation Administration, March 2002.

Note: You may obtain References 1, 3, 4, 7, and 8 from the Federal Aviation Administration, Small Airplane Directorate Standards Office, telephone number 816-329-4111.

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA

A1-1. What flight and ground load spectra are presented in this appendix?

The following list summarizes the flight and ground load spectra presented in this appendix.

Flight Spectra, General Usage, Single-Engine Unpressurized			
Description	Graphic Data	Tabulated Data	Comments
Gust spectra for single-engine, unpressurized operations, including basic instruction, personal, executive, and aerobatic usage.	Figure A1-1	Table A1-1	These spectra should also be used for pressurized single-engine airplanes that spend a significant amount of flight time at low altitudes.
Maneuver spectra for single-engine basic instruction usage.	Figure A1-2	Table A1-2	
Maneuver spectra for single-engine personal usage.	Figure A1-3	Table A1-3	An airplane in the personal usage category has a single, reciprocating engine with 185 horsepower or less.
Maneuver spectra for single-engine executive usage.	Figure A1-4	Table A1-4	An airplane in the executive usage category has a single, reciprocating engine with more than 185 horsepower. The executive usage category also includes unpressurized, single-engine turboprop airplanes.
Maneuver spectra for single-engine aerobatic usage.	Figure A1-5	Table A1-5	Applicable to typical aerobatic category airplanes, n_z = +6g / -3g
Flight Spectra, General Usage, Twin-Engine Unpressurized			
Description	Graphic Data	Tabulated Data	Comments
Gust spectra for twin-engine, unpressurized operations, including instruction, and general usage.	Figure A1-6	Table A1-6	These spectra should also be used for pressurized twin-engine airplanes that may spend a significant amount of flight time at low altitudes, 7,000 feet and below. An example of this type of operation is short flight duration commuter and air taxi operations.
Maneuver spectra for twin-engine instruction usage.	Figure A1-7	Table A1-7	
Maneuver spectra for twin-engine general usage.	Figure A1-8	Table A1-8	These spectra should also be used for pressurized and unpressurized twin-engine airplanes that may be operated in short flight duration commuter and air taxi operations.

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Flight Spectra, General Usage, Single-Engine and Twin-Engine Pressurized			
Description	Graphic Data	Tabulated Data	Comments
Gust spectra for pressurized usage.	Figure A1-9	Table A1-9	
Maneuver spectra for pressurized usage.	Figure A1-10	Table A1-10	
Flight Spectra, Special Usage – Agricultural (Aerial Application)			
Description	Graphic Data	Tabulated Data	Comments
Gust spectra for agricultural or aerial application usage.	Figure A1-11	Table A1-11	
Maneuver spectra for agricultural or aerial application usage.	Figure A1-12	Table A1-12	
Flight Spectra, Special Usage – Survey (Pipeline Patrol)			
Description	Graphic Data	Tabulated Data	Comments
Gust spectra for low-level survey or pipeline patrol usage.	Figure A1-13	Table A1-13	
Maneuver spectra for low-level survey or pipeline patrol usage.	Figure A1-14	Table A1-14	
Ground Spectra			
Description	Graphic Data	Tabulated Data	Comments
Landing Impact	Figure A1-15		
Taxi	Figure A1-16		

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

A1-2. How do I compute the gust limit load factor used in the gust load spectra?

a. The equation for computing the gust load limit load factor has evolved through the regulatory history of the Civil Air Regulation Part 3 (CAR 3) and 14 CFR Part 23. These changes to the gust load formula incorporate the improving knowledge of airplane response to atmospheric turbulence. The gust load spectra presented in this AC are normalized to the gust load formula in use before Amendment 23-7 to 14 CFR Part 23. Regardless of the certification basis of your airplane, you must calculate a gust load limit load factor (for use in developing fatigue loading spectra only) using the same equation used to derive the gust spectra.

b. Formula for computing the gust load limit load factor:

$$a_{nLLF} = \frac{U\ K\ V\ m}{498\ \frac{W}{S}}$$

where U = 30.0, nominal gust velocity in feet per second

$$K = \frac{1}{2}\left(\frac{W}{S}\right)^{\frac{1}{4}} \quad \text{for W/S < 16 lbs / ft}^2$$

$$= 1.33 - \frac{2.67}{\left(\frac{W}{S}\right)^{\frac{3}{4}}} \quad \text{for W/S > 16 lbs / ft}^2$$

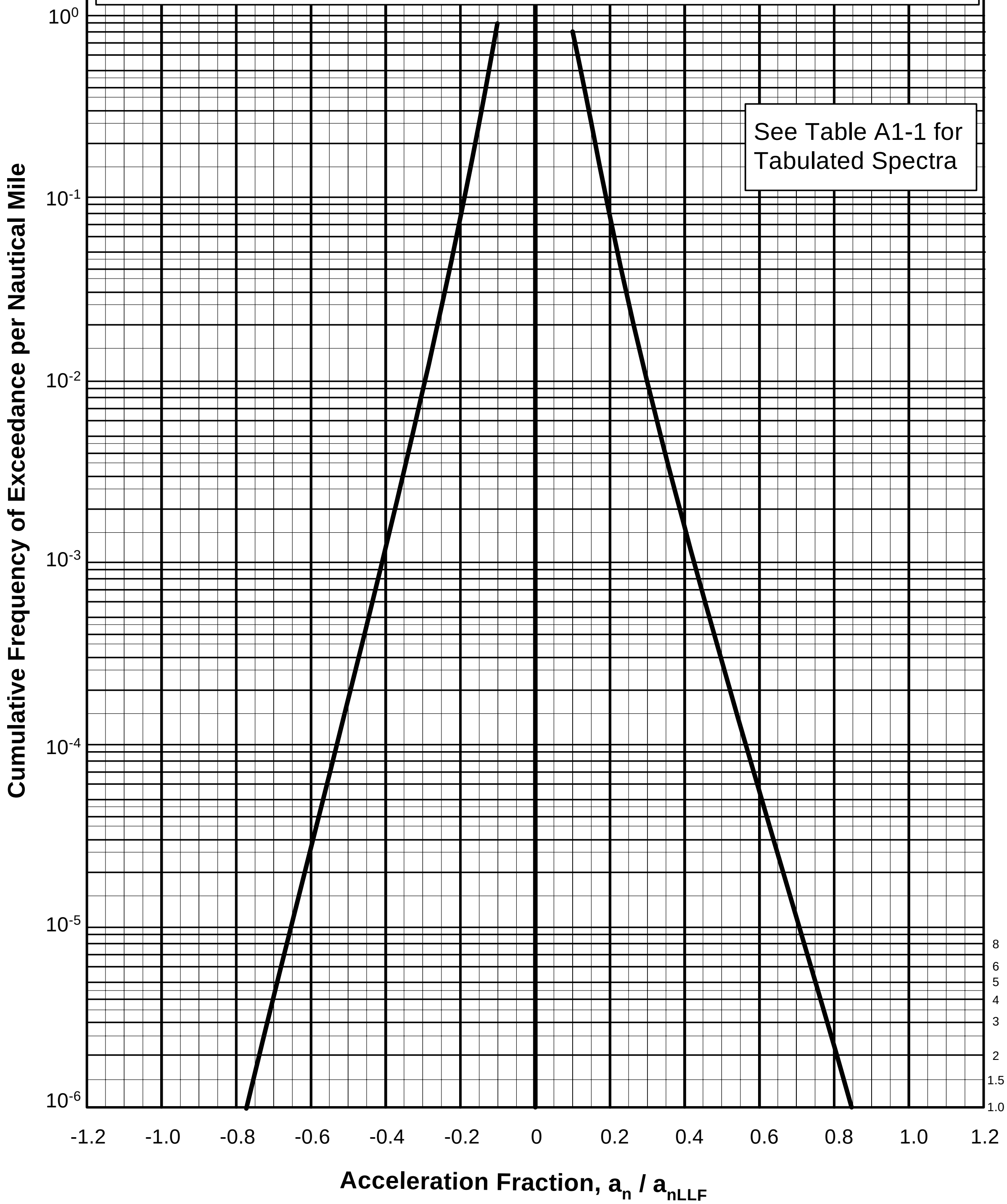
W / S = Wing loading at maximum weight, lbs / ft^2

V = Airplane structural design cruise speed, V_C, KEAS

m = Wing lift curve slope, $C_{L\alpha}$, rad^{-1}

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Figure A1-1
Gust Spectra
For Single-Engine Unpressurized Operations
Including Basic Instruction, Personal, Executive, and Aerobatic Usage



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Table A1-1

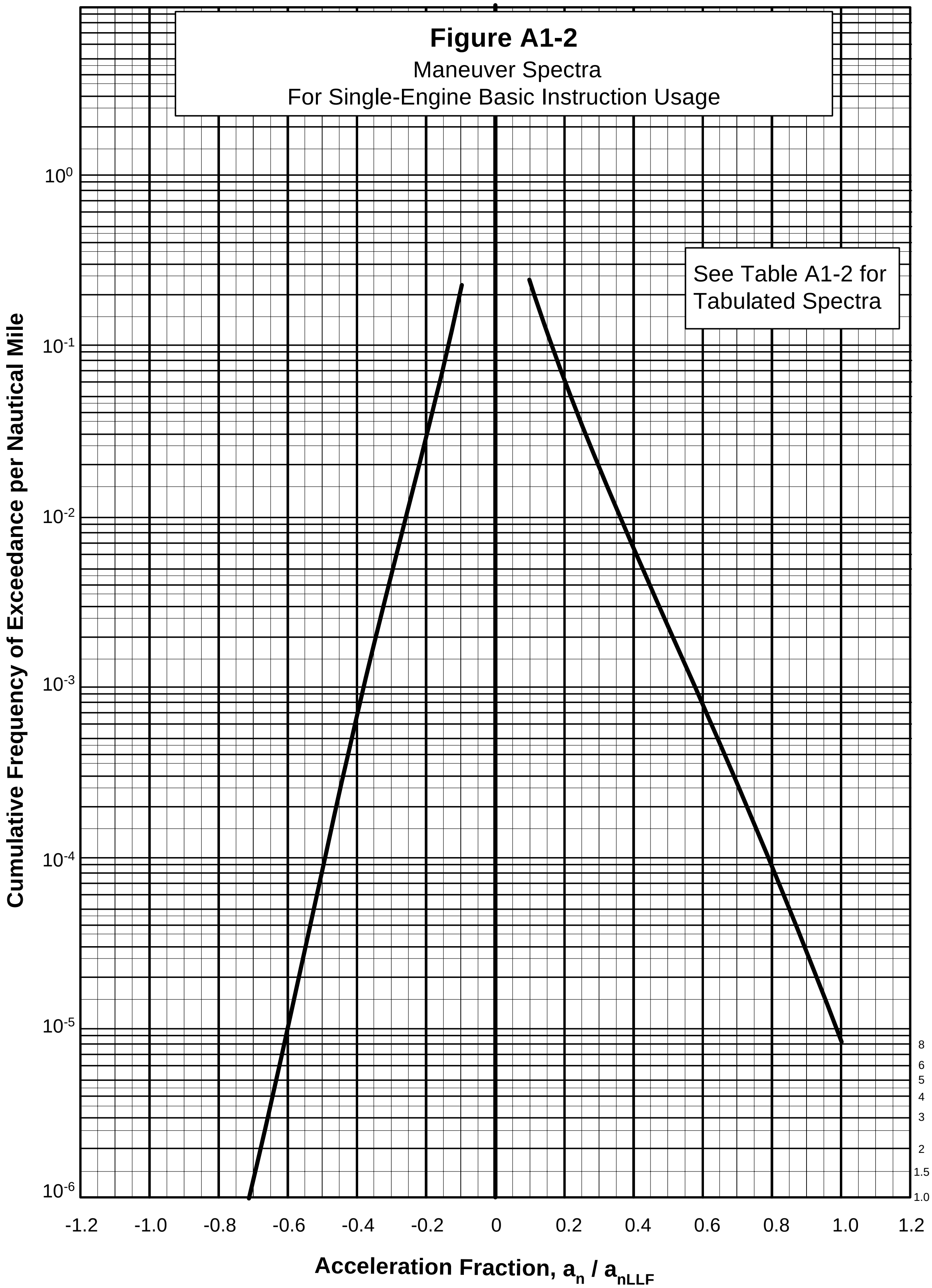
Gust Spectra

For Single-Engine Unpressurized Operations

Including Basic Instruction, Personal, Executive, and Aerobatic Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	7.99040E-01		-0.10	8.86903E-01
0.15	2.39762E-01		-0.15	2.90400E-01
0.20	8.26537E-02		-0.20	7.93185E-02
0.25	2.59957E-02		-0.25	2.50615E-02
0.30	9.43467E-03		-0.30	8.64457E-03
0.35	3.74392E-03		-0.35	3.13716E-03
0.40	1.56672E-03		-0.40	1.17560E-03
0.45	6.76810E-04		-0.45	4.49619E-04
0.50	2.96910E-04		-0.50	1.74053E-04
0.55	1.31101E-04		-0.55	6.77344E-05
0.60	5.79043E-05		-0.60	2.63806E-05
0.65	2.55611E-05		-0.65	1.02760E-05
0.70	1.12764E-05		-0.70	4.00187E-06
0.75	4.97072E-06		-0.75	1.55782E-06
0.80	2.18708E-06		-0.80	6.05622E-07
0.85	9.58226E-07		-0.85	2.34685E-07
0.90	4.15769E-07		-0.90	9.01733E-08
0.95	1.76305E-07		-0.95	3.38733E-08
1.00	7.05919E-08		-1.00	1.19395E-08

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

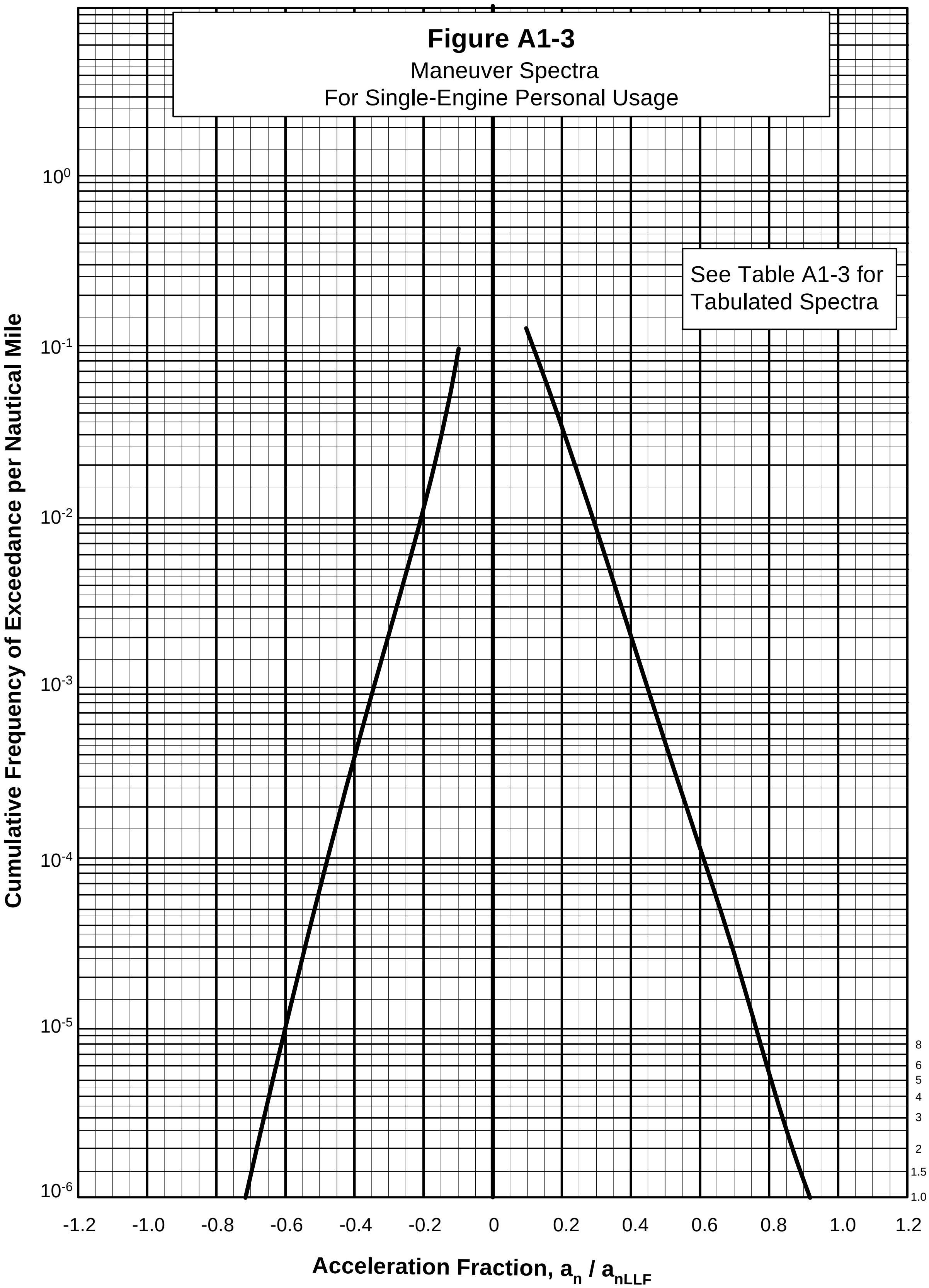
Table A1-2

Maneuver Spectra

For Single-Engine Basic Instruction Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	2.37466E-01		-0.10	2.22202E-01
0.15	1.20023E-01		-0.15	7.79188E-02
0.20	6.20605E-02		-0.20	2.89635E-02
0.25	3.45611E-02		-0.25	1.18361E-02
0.30	1.99278E-02		-0.30	4.73564E-03
0.35	1.16586E-02		-0.35	1.81743E-03
0.40	6.84325E-03		-0.40	6.69379E-04
0.45	4.01154E-03		-0.45	2.39525E-04
0.50	2.34706E-03		-0.50	8.45918E-05
0.55	1.36820E-03		-0.55	2.98155E-05
0.60	7.93694E-04		-0.60	1.04723E-05
0.65	4.59287E-04		-0.65	3.66958E-06
0.70	2.65584E-04		-0.70	1.28565E-06
0.75	1.53197E-04		-0.75	4.50315E-07
0.80	8.79528E-05		-0.80	1.57610E-07
0.85	5.01008E-05		-0.85	5.50454E-08
0.90	2.81951E-05		-0.90	1.91064E-08
0.95	1.55016E-05		-0.95	6.51330E-09
1.00	8.15702E-06		-1.00	2.10063E-09

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

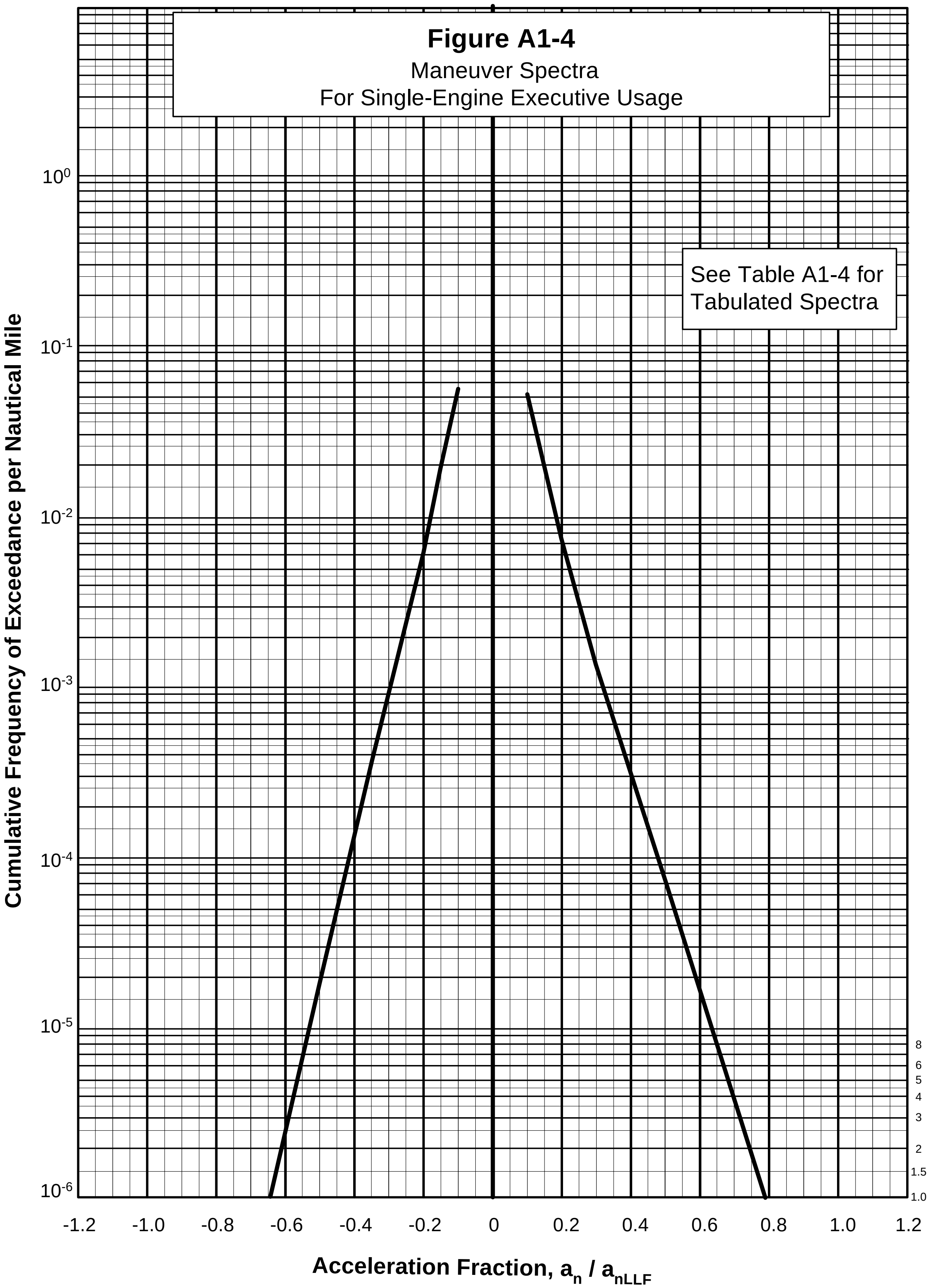
Table A1-3

Maneuver Spectra

For Single-Engine Personal Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	1.28156E-01		-0.10	9.71117E-02
0.15	6.27614E-02		-0.15	3.13880E-02
0.20	3.09500E-02		-0.20	1.09533E-02
0.25	1.58120E-02		-0.25	4.71523E-03
0.30	8.12433E-03		-0.30	2.07230E-03
0.35	4.12127E-03		-0.35	8.91311E-04
0.40	2.04241E-03		-0.40	3.75403E-04
0.45	9.97803E-04		-0.45	1.55129E-04
0.50	4.83819E-04		-0.50	6.29010E-05
0.55	2.33291E-04		-0.55	2.50877E-05
0.60	1.12320E-04		-0.60	9.84679E-06
0.65	5.40555E-05		-0.65	3.80374E-06
0.70	2.59889E-05		-0.70	1.44615E-06
0.75	1.24766E-05		-0.75	5.41094E-07
0.80	5.97051E-06		-0.80	1.99191E-07
0.85	2.83754E-06		-0.85	7.20825E-08
0.90	1.32888E-06		-0.90	2.55778E-08
0.95	6.02384E-07		-0.95	8.83264E-09
1.00	2.52545E-07		-1.00	2.89847E-09

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

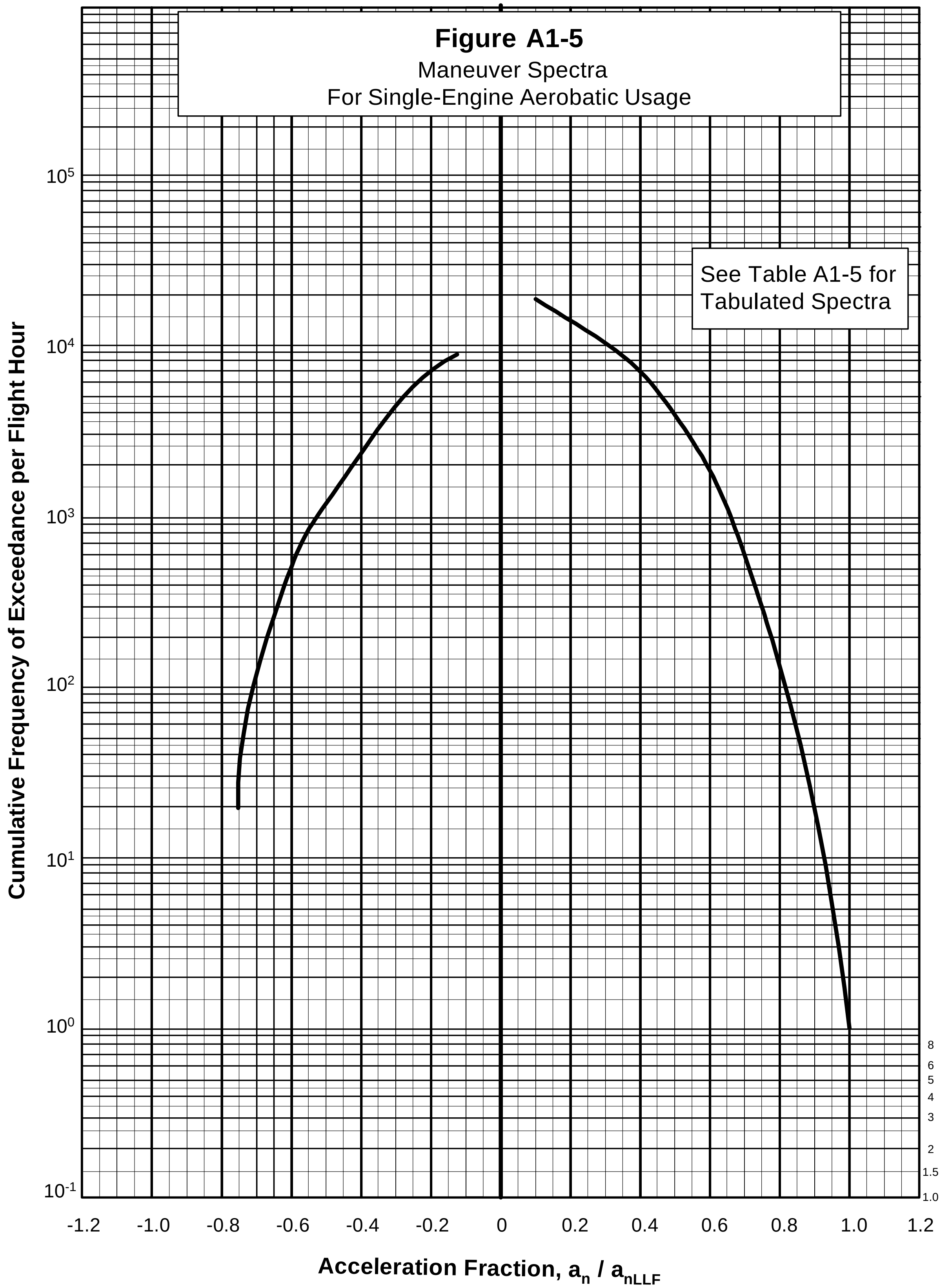
Table A1-4

Maneuver Spectra

For Single-Engine Executive Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	5.10093E-02		-0.10	5.54495E-02
0.15	1.97508E-02		-0.15	1.88747E-02
0.20	7.38092E-03		-0.20	6.05428E-03
0.25	3.13527E-03		-0.25	2.32888E-03
0.30	1.40483E-03		-0.30	9.20494E-04
0.35	6.60426E-04		-0.35	3.59308E-04
0.40	3.12393E-04		-0.40	1.35529E-04
0.45	1.48798E-04		-0.45	4.99636E-05
0.50	7.13657E-05		-0.50	1.81837E-05
0.55	3.43582E-05		-0.55	6.53422E-06
0.60	1.65390E-05		-0.60	2.31853E-06
0.65	7.96002E-06		-0.65	8.12393E-07
0.70	3.82975E-06		-0.70	2.81096E-07
0.75	1.84127E-06		-0.75	9.60362E-08
0.80	8.83928E-07		-0.80	3.23840E-08
0.85	4.23025E-07		-0.85	1.07639E-08
0.90	2.01127E-07		-0.90	3.51182E-09
0.95	9.42952E-08		-0.95	1.10946E-09
1.00	4.28623E-08		-1.00	3.23514E-10

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

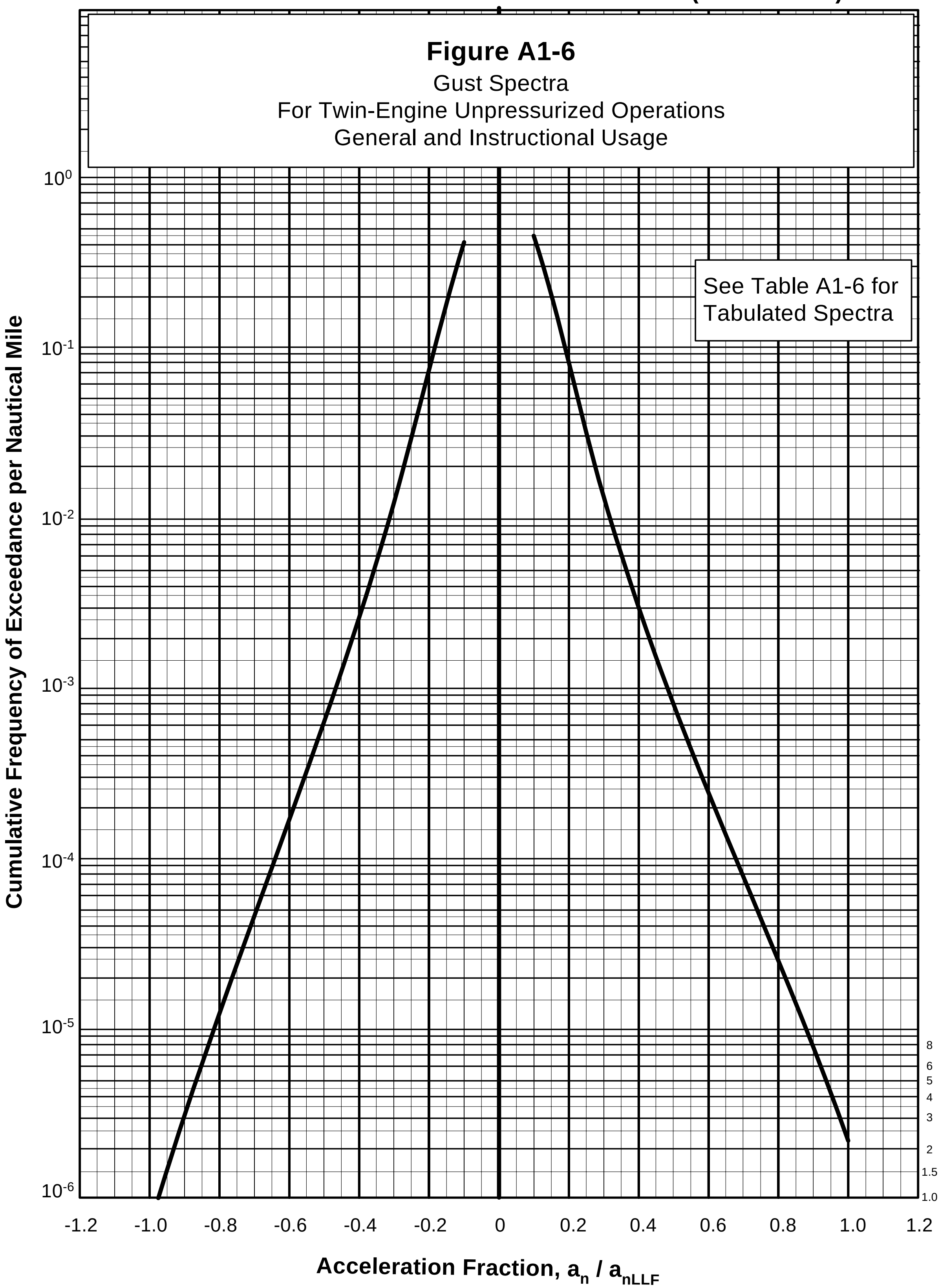
Table A1-5

Maneuver Spectra

For Single-Engine Aerobatic Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Flight Hour		Acceleration Fraction	Cumulative Frequency of Exceedance per Flight Hour
0.10	$1.8078 \times 10^{+4}$		-0.125	$8.7670 \times 10^{+3}$
0.20	$1.4840 \times 10^{+4}$		-0.250	$5.5320 \times 10^{+3}$
0.30	$1.0334 \times 10^{+4}$		-0.375	$2.6830 \times 10^{+3}$
0.40	$6.9640 \times 10^{+3}$		-0.500	$1.1400 \times 10^{+3}$
0.50	$3.8520 \times 10^{+3}$		-0.625	$3.7300 \times 10^{+2}$
0.60	$1.7490 \times 10^{+3}$		-0.750	$2.0000 \times 10^{+1}$
0.70	$5.9100 \times 10^{+2}$			
0.80	$1.3900 \times 10^{+2}$			
0.90	$2.0000 \times 10^{+1}$			
1.00	$1.0000 \times 10^{+0}$			

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Table A1-6

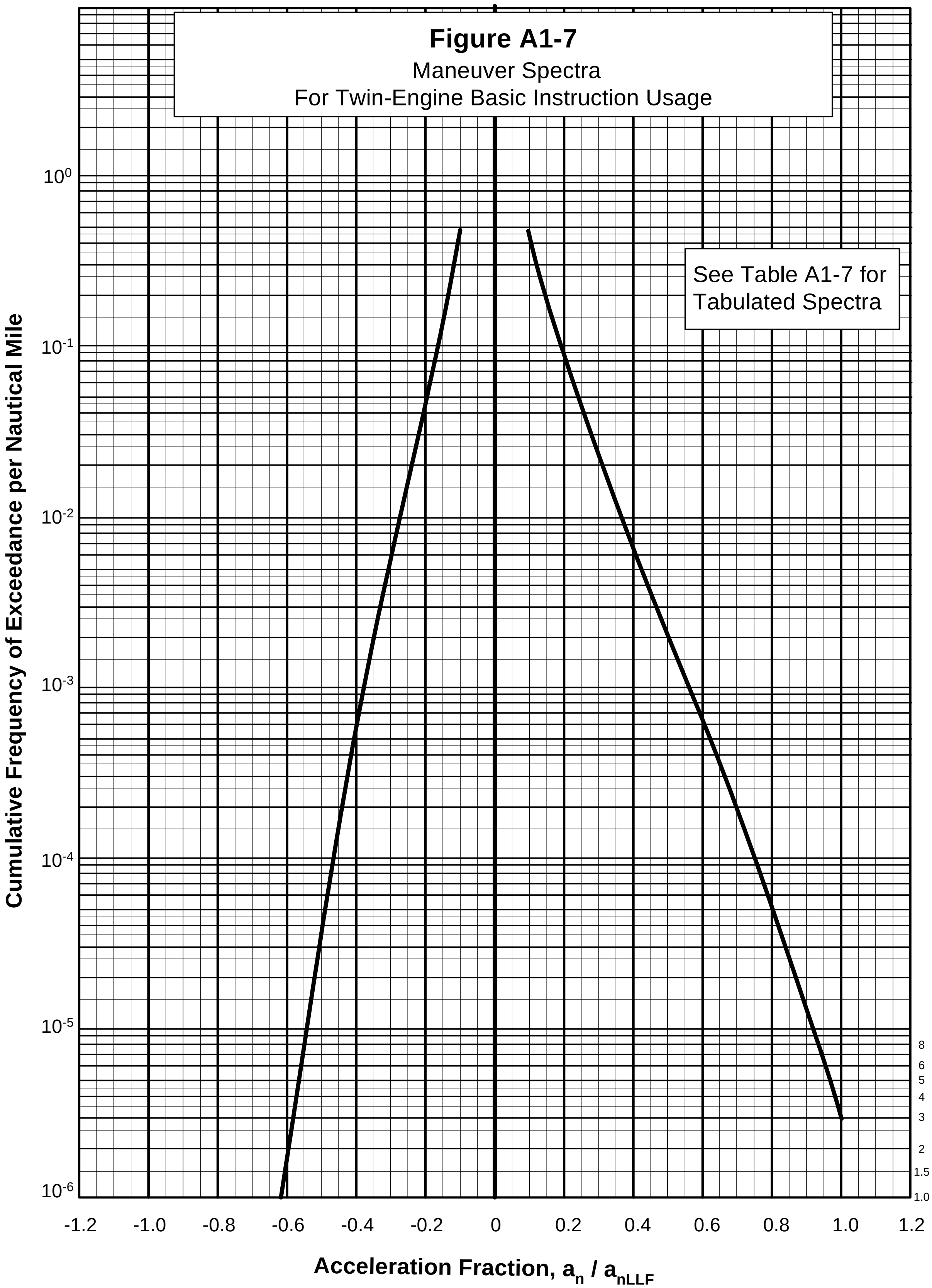
Gust Spectra

For Twin-Engine Unpressurized Operations

General and Instructional Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	4.44668E-01		-0.10	4.06116E-01
0.15	1.61492E-01		-0.15	1.34138E-01
0.20	7.99116E-02		-0.20	7.30813E-02
0.25	3.17301E-02		-0.25	2.83576E-02
0.30	1.30374E-02		-0.30	1.18229E-02
0.35	5.87427E-03		-0.35	5.35232E-03
0.40	2.76767E-03		-0.40	2.55578E-03
0.45	1.30141E-03		-0.45	1.26417E-03
0.50	5.28867E-04		-0.50	6.41583E-04
0.55	4.71092E-04		-0.55	3.29008E-04
0.60	2.58561E-04		-0.60	1.70215E-04
0.65	1.42661E-04		-0.65	8.72554E-05
0.70	7.91060E-05		-0.70	4.41745E-05
0.75	4.51160E-05		-0.75	2.30083E-05
0.80	2.52331E-05		-0.80	1.16222E-05
0.85	1.40821E-05		-0.85	6.12841E-06
0.90	7.83366E-06		-0.90	2.98732E-06
0.95	4.24939E-06		-0.95	1.50864E-06
1.00	2.18891E-06		-1.00	7.03079E-07

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

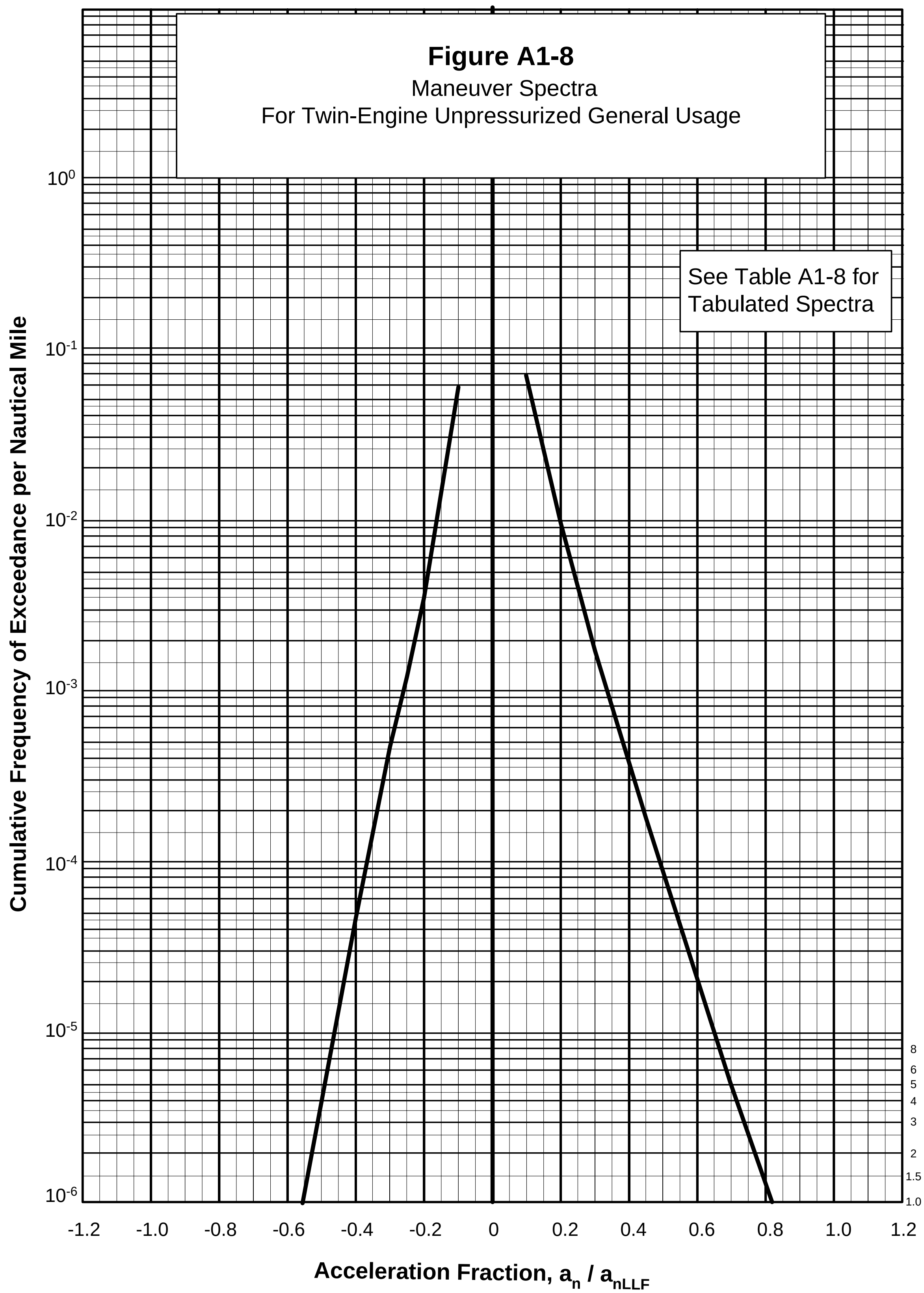
Table A1-7

Maneuver Spectra

For Twin-Engine Basic Instruction Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	4.73568E-01		-0.10	4.78815E-01
0.15	1.90555E-01		-0.15	1.38794E-01
0.20	8.52330E-02		-0.20	4.55853E-02
0.25	4.27211E-02		-0.25	1.65114E-02
0.30	2.27318E-02		-0.30	5.66718E-03
0.35	1.24765E-02		-0.35	1.80926E-03
0.40	6.94308E-03		-0.40	5.30806E-04
0.45	3.87445E-03		-0.45	1.41855E-04
0.50	2.15159E-03		-0.50	3.45231E-05
0.55	1.18268E-03		-0.55	7.68064E-06
0.60	6.41120E-04		-0.60	1.56206E-06
0.65	3.41647E-04		-0.65	2.90578E-07
0.70	1.79389E-04		-0.70	4.94860E-08
0.75	9.35425E-05		-0.75	7.76661E-09
0.80	4.85944E-05		-0.80	1.17677E-09
0.85	2.50637E-05		-0.85	2.26425E-10
0.90	1.27451E-05		-0.90	1.01278E-10
0.95	6.29624E-06		-0.95	1.84240E-12
1.00	2.92019E-06		-1.00	1.89361E-13

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Table A1-8

Maneuver Spectra

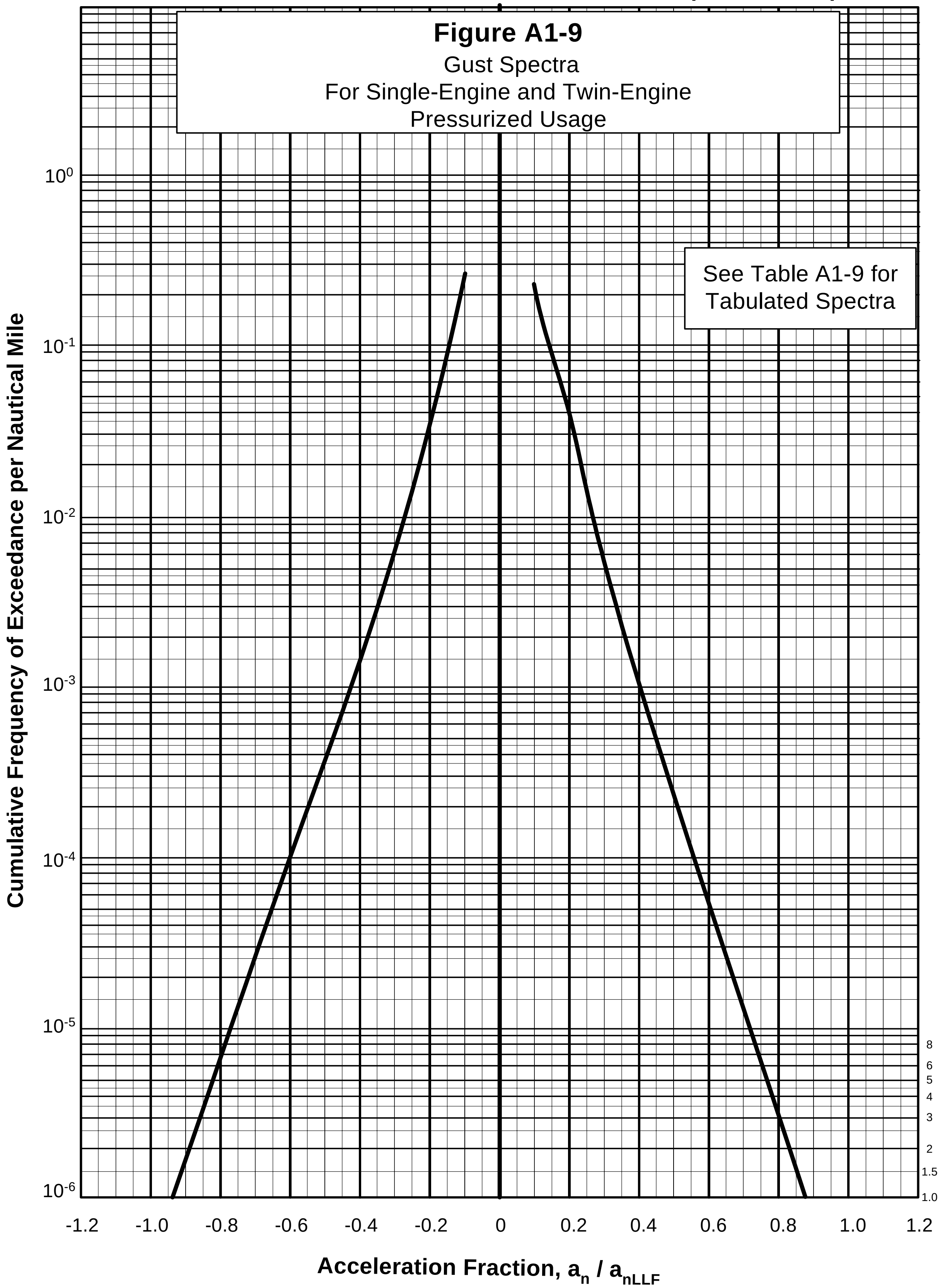
For Twin-Engine Unpressurized General Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	6.89490E-02		-0.10	5.91059E-02
0.15	2.45115E-02		-0.15	1.49523E-02
0.20	9.65055E-03		-0.20	3.47737E-03
0.25	3.95338E-03		-0.25	1.23851E-03
0.30	1.73800E-03		-0.30	4.62353E-04
0.35	7.95244E-04		-0.35	1.52520E-04
0.40	3.75183E-04		-0.40	4.60500E-05
0.45	1.81414E-04		-0.45	1.37289E-05
0.50	8.94887E-05		-0.50	4.06783E-06
0.55	4.53754E-05		-0.55	1.20075E-06
0.60	2.40712E-05		-0.60	3.54448E-07
0.65	1.37295E-05		-0.65	1.04628E-07
0.70	8.69829E-06		-0.70	3.08837E-08
0.75	6.19922E-06		-0.75	9.11504E-09
0.80	4.92938E-06		-0.80	2.68914E-09
0.85	6.64545E-07		-0.85	7.92276E-10
0.90	3.36409E-07		-0.90	2.32341E-10
0.95	1.69605E-07		-0.95	6.70536E-11
1.00	8.48118E-08		-1.00	1.82622E-11

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Figure A1-9

Gust Spectra
For Single-Engine and Twin-Engine
Pressurized Usage



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Table A1-9

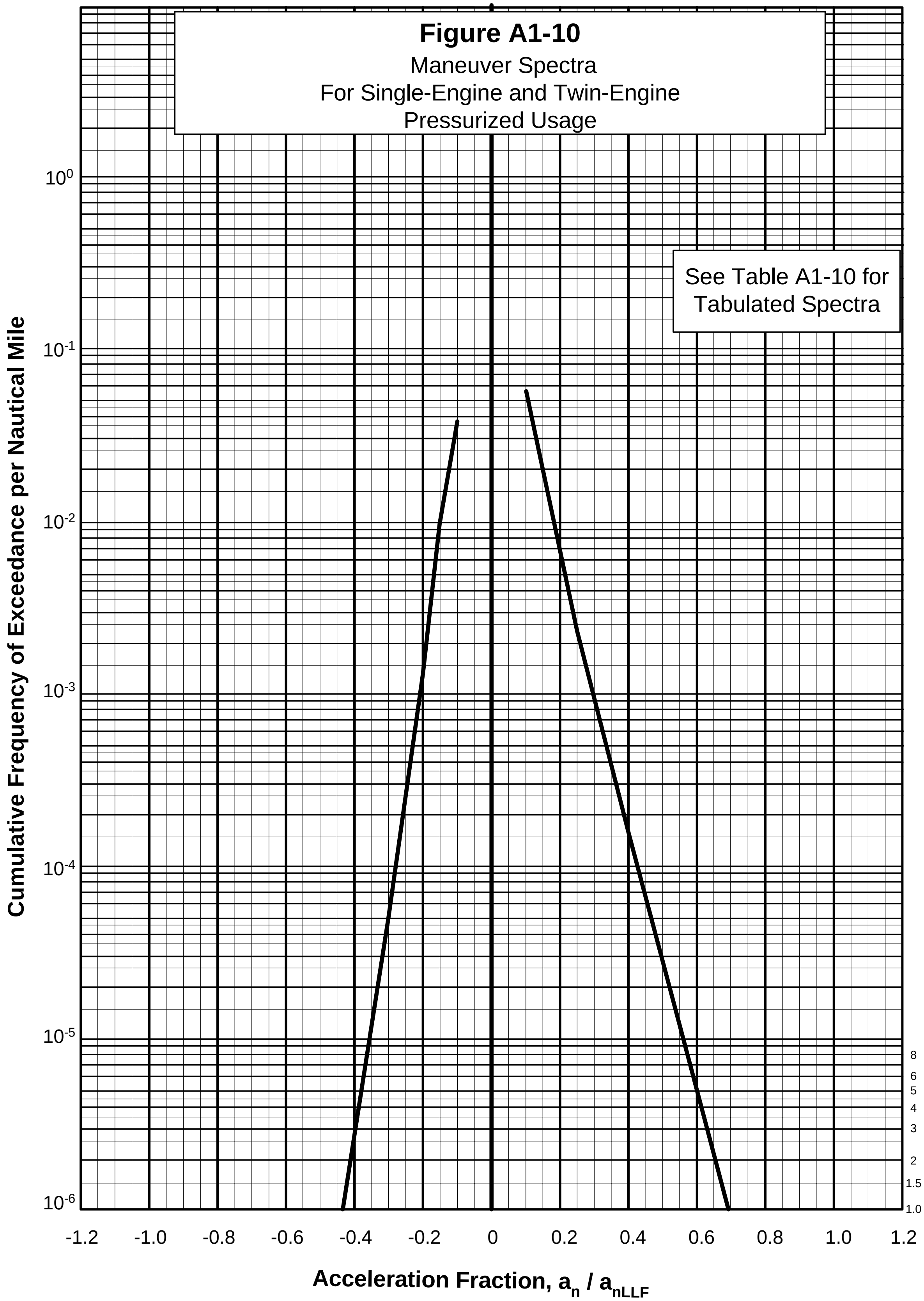
Gust Spectra

For Single-Engine and Twin-Engine Pressurized Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	2.23784E-01		-0.10	2.54285E-01
0.15	8.49264E-02		-0.15	9.72598E-02
0.20	3.88721E-02		-0.20	3.64316E-02
0.25	1.40381E-02		-0.25	1.44418E-02
0.30	5.65101E-03		-0.30	6.36368E-03
0.35	2.42471E-03		-0.35	2.99910E-03
0.40	1.08187E-03		-0.40	1.47318E-03
0.45	4.95557E-04		-0.45	7.42192E-04
0.50	2.33345E-04		-0.50	3.79530E-04
0.55	1.12397E-04		-0.55	1.95259E-04
0.60	5.47654E-05		-0.60	1.00525E-04
0.65	2.68277E-05		-0.65	5.17244E-05
0.70	1.31310E-05		-0.70	2.65940E-05
0.75	6.41930E-06		-0.75	1.36445E-05
0.80	3.13035E-06		-0.80	6.97417E-06
0.85	1.51868E-06		-0.85	3.53834E-06
0.90	7.28911E-07		-0.90	1.76854E-06
0.95	3.41903E-07		-0.95	8.56921E-07
1.00	1.52259E-07		-1.00	3.87347E-07

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Figure A1-10
Maneuver Spectra
For Single-Engine and Twin-Engine
Pressurized Usage



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

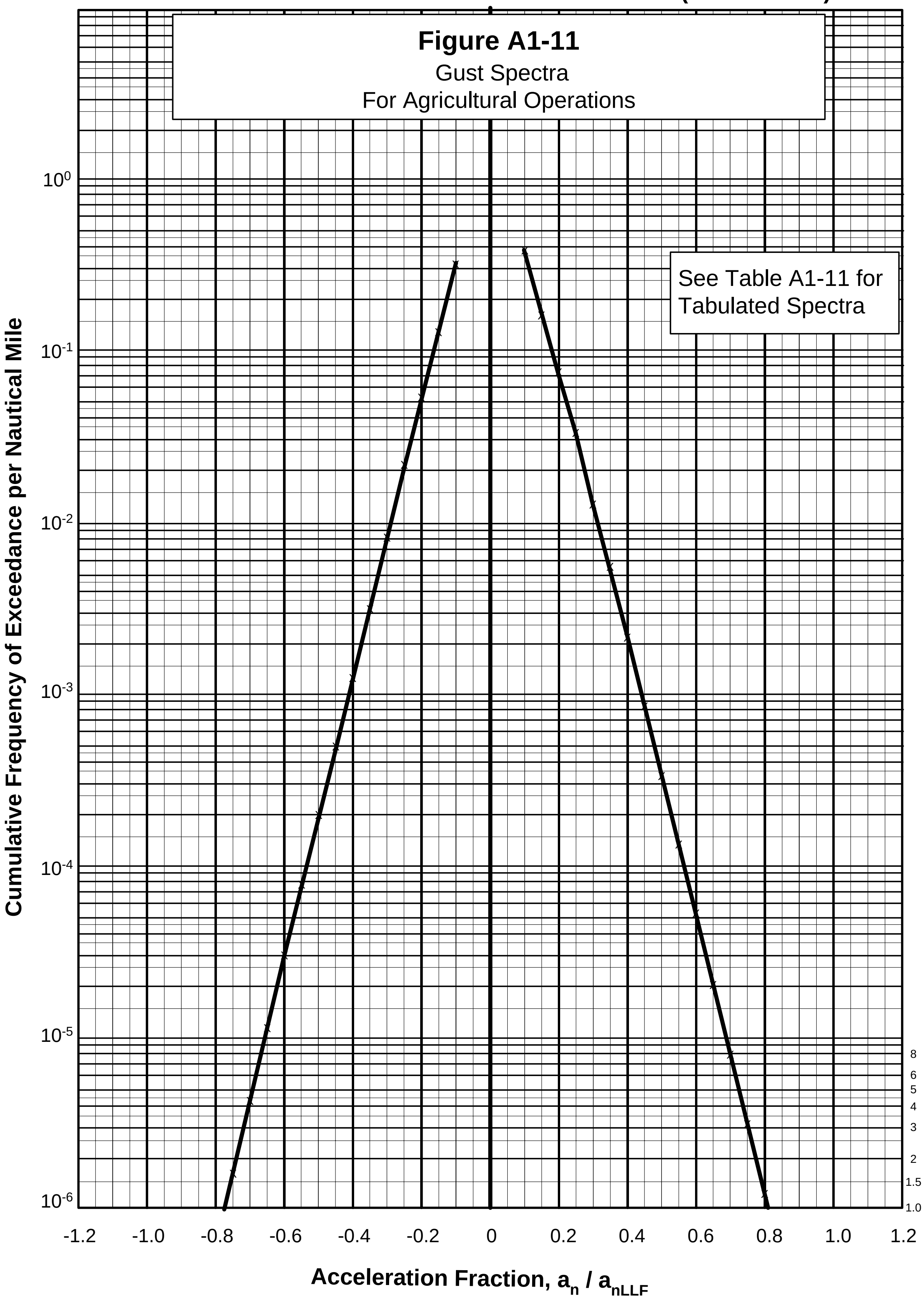
Table A1-10

Maneuver Spectra

For Single-Engine and Twin-Engine Pressurized Usage

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	5.61695E-02		-0.10	3.78784E-02
0.15	2.02968E-02		-0.15	9.82270E-03
0.20	6.63342E-03		-0.20	1.36144E-03
0.25	2.36553E-03		-0.25	2.41970E-04
0.30	9.20697E-04		-0.30	5.03001E-05
0.35	3.74938E-04		-0.35	1.16053E-05
0.40	1.56632E-04		-0.40	2.79594E-06
0.45	6.60459E-05		-0.45	6.79568E-07
0.50	2.78816E-05		-0.50	1.71237E-07
0.55	1.17951E-05		-0.55	4.91413E-08
0.60	5.01701E-06		-0.60	1.98149E-08
0.65	2.15974E-06		-0.65	1.27710E-08
0.70	9.55665E-07		-0.70	2.69318E-09
0.75	4.48349E-07		-0.75	2.72578E-10
0.80	2.34443E-07		-0.80	1.74971E-10
0.85	1.44293E-07		-0.85	3.53231E-11
0.90	1.06299E-07		-0.90	1.78106E-12
0.95	1.09189E-08		-0.95	4.28533E-13
1.00	4.17046E-09		-1.00	1.03669E-13

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

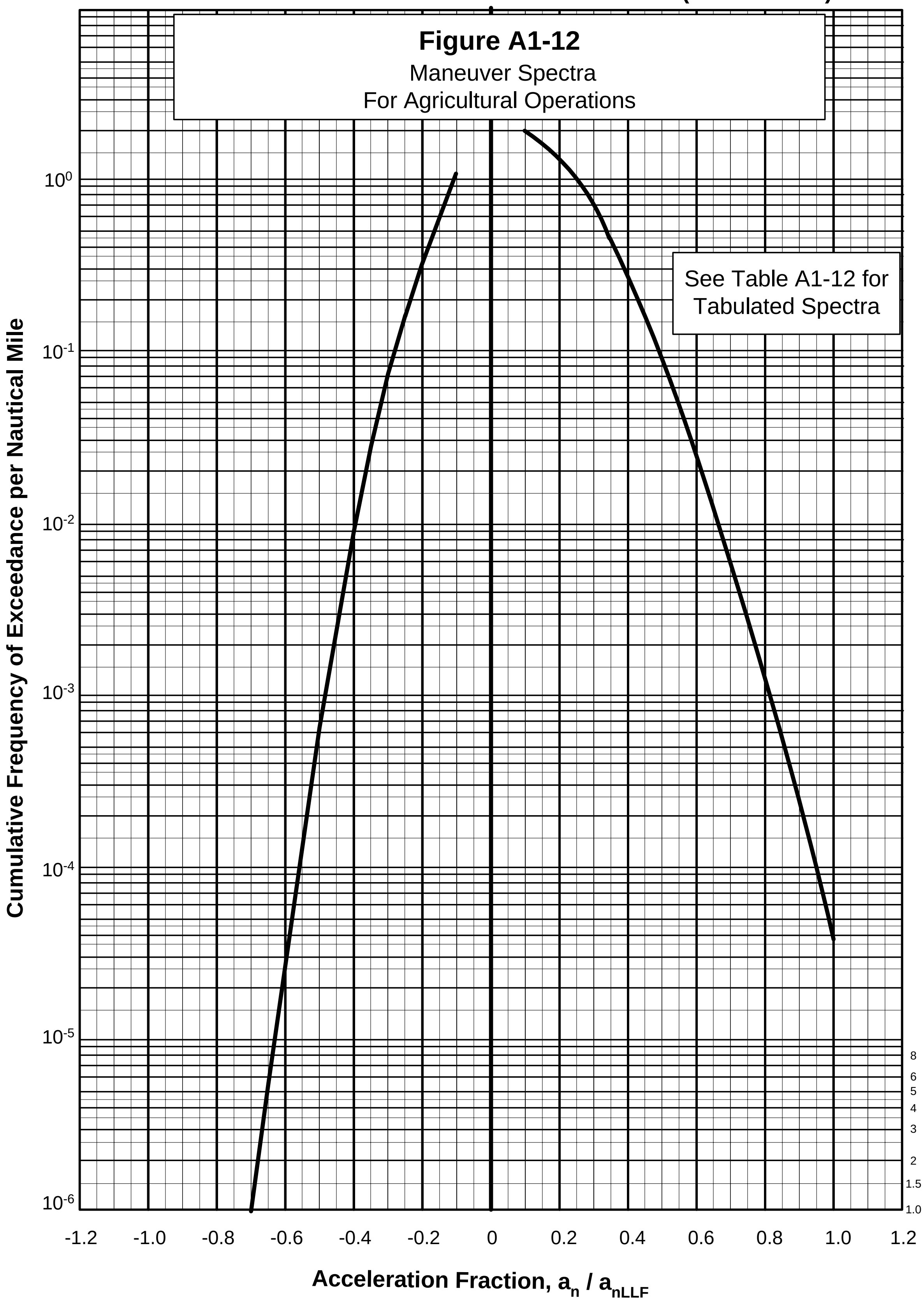


APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

Table A1-11
Gust Spectra
For Agricultural Operations

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	3.77707E-01		-0.10	3.11916E-01
0.15	1.61542E-01		-0.15	1.31643E-01
0.20	7.22319E-02		-0.20	5.16164E-02
0.25	3.20268E-02		-0.25	2.15305E-02
0.30	1.34855E-02		-0.30	8.16495E-03
0.35	5.36510E-03		-0.35	3.21070E-03
0.40	2.13189E-03		-0.40	1.26787E-03
0.45	8.47618E-04		-0.45	5.01627E-04
0.50	3.36518E-04		-0.50	1.98343E-04
0.55	1.33203E-04		-0.55	7.78616E-05
0.60	5.22586E-05		-0.60	3.02250E-05
0.65	2.04295E-05		-0.65	1.15962E-05
0.70	7.94281E-06		-0.70	4.39690E-06
0.75	3.12936E-06		-0.75	1.64734E-06
0.80	1.23810E-06		-0.80	6.09684E-07
0.85	4.86588E-07		-0.85	2.22658E-07
0.90	1.89600E-07		-0.90	8.00213E-08
0.95	7.28972E-08		-0.95	2.80746E-08
1.00	2.72943E-08		-1.00	9.38019E-09

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

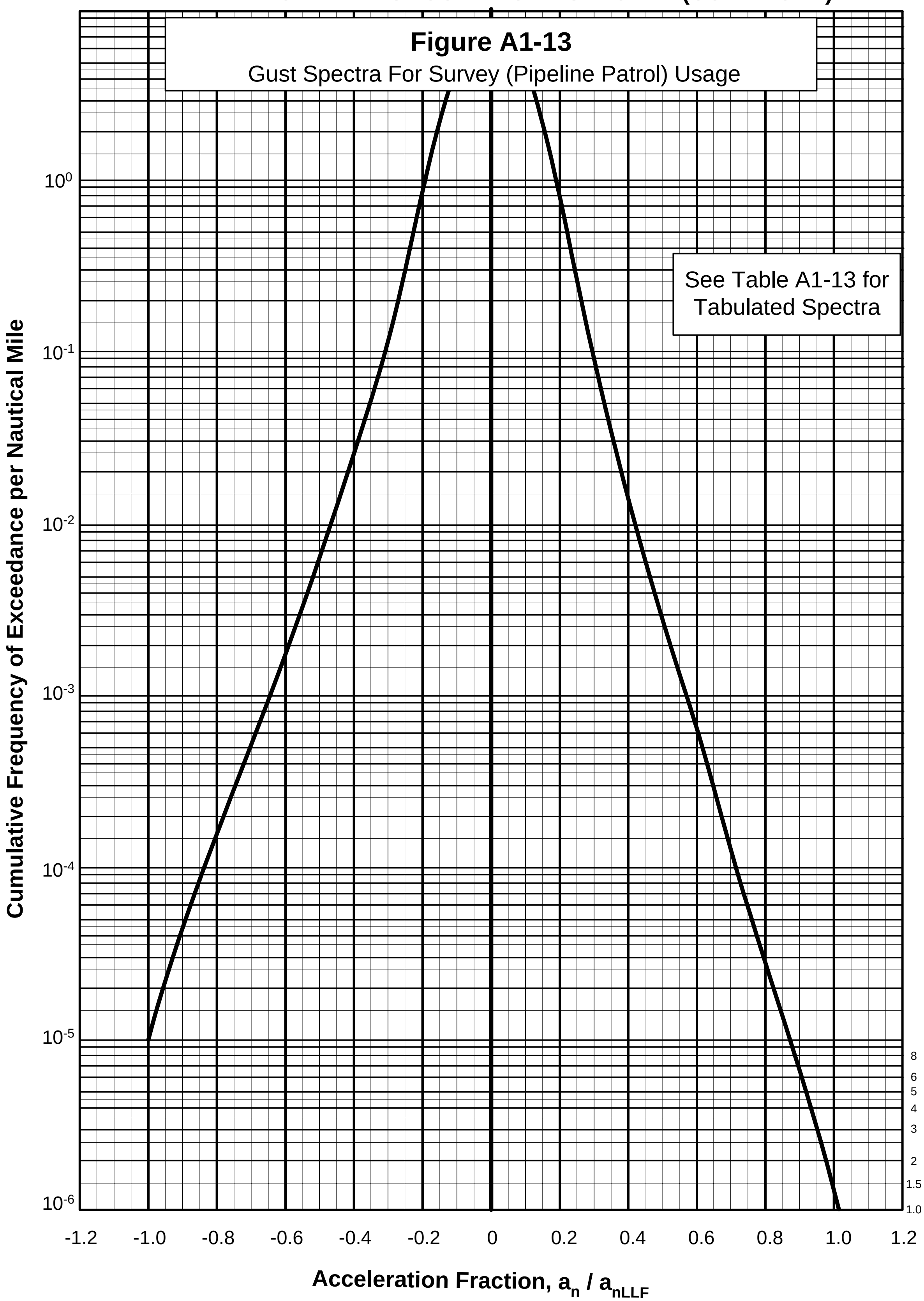
Table A1-12

Maneuver Spectra

For Agricultural Operations

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	1.95680E+00		-0.10	1.06968E+00
0.15	1.62395E+00		-0.15	5.94801E-01
0.20	1.30823E+00		-0.20	3.13178E-01
0.25	9.97985E-01		-0.25	1.61990E-01
0.30	6.88328E-01		-0.30	7.11840E-02
0.35	4.44979E-01		-0.35	2.71675E-02
0.40	2.73441E-01		-0.40	8.78560E-03
0.45	1.61710E-01		-0.45	2.43350E-03
0.50	8.97378E-02		-0.50	6.28618E-04
0.55	4.80394E-02		-0.55	1.42839E-04
0.60	2.48058E-02		-0.60	2.96876E-05
0.65	1.23385E-02		-0.65	5.42758E-06
0.70	5.88998E-03		-0.70	9.62766E-07
0.75	2.76188E-03		-0.75	2.06299E-07
0.80	1.26089E-03		-0.80	9.21867E-08
0.85	5.56561E-04		-0.85	1.41585E-09
0.90	2.38497E-04		-0.90	1.11517E-09
0.95	9.89316E-05		-0.95	1.79203E-11
1.00	3.75956E-05		-1.00	1.43870E-12

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

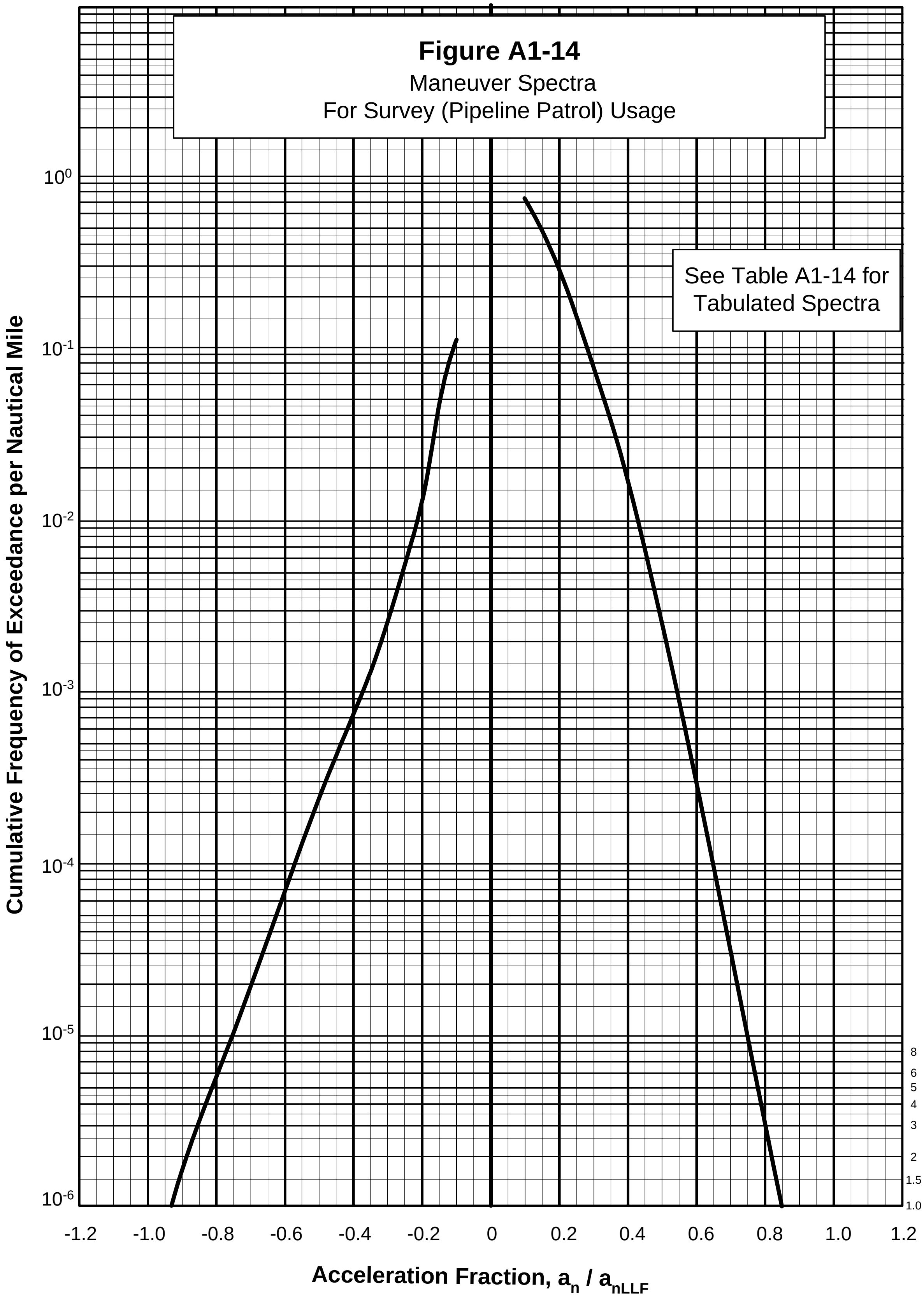
Table A1-13

Gust Spectra

For Survey (Pipeline Patrol) Operations

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	4.82042E+00		-0.10	4.81125E+00
0.15	2.07621E+00		-0.15	2.11771E+00
0.20	7.96900E-01		-0.20	8.25646E-01
0.25	2.46602E-01		-0.25	2.88959E-01
0.30	8.89211E-02		-0.30	1.17488E-01
0.35	3.52779E-02		-0.35	5.26061E-02
0.40	1.48846E-02		-0.40	2.51299E-02
0.45	6.53546E-03		-0.45	1.25490E-02
0.50	2.94288E-03		-0.50	6.46049E-03
0.55	1.34544E-03		-0.55	3.39526E-03
0.60	6.20257E-04		-0.60	1.80803E-03
0.65	2.87768E-04		-0.65	9.69846E-04
0.70	1.34389E-04		-0.70	5.21987E-04
0.75	6.29908E-05		-0.75	2.80579E-04
0.80	2.95677E-05		-0.80	1.50280E-04
0.85	1.38490E-05		-0.85	7.99172E-05
0.90	6.43049E-06		-0.90	4.18344E-05
0.95	2.92107E-06		-0.95	2.12119E-05
1.00	1.26226E-06		-1.00	1.00431E-05

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

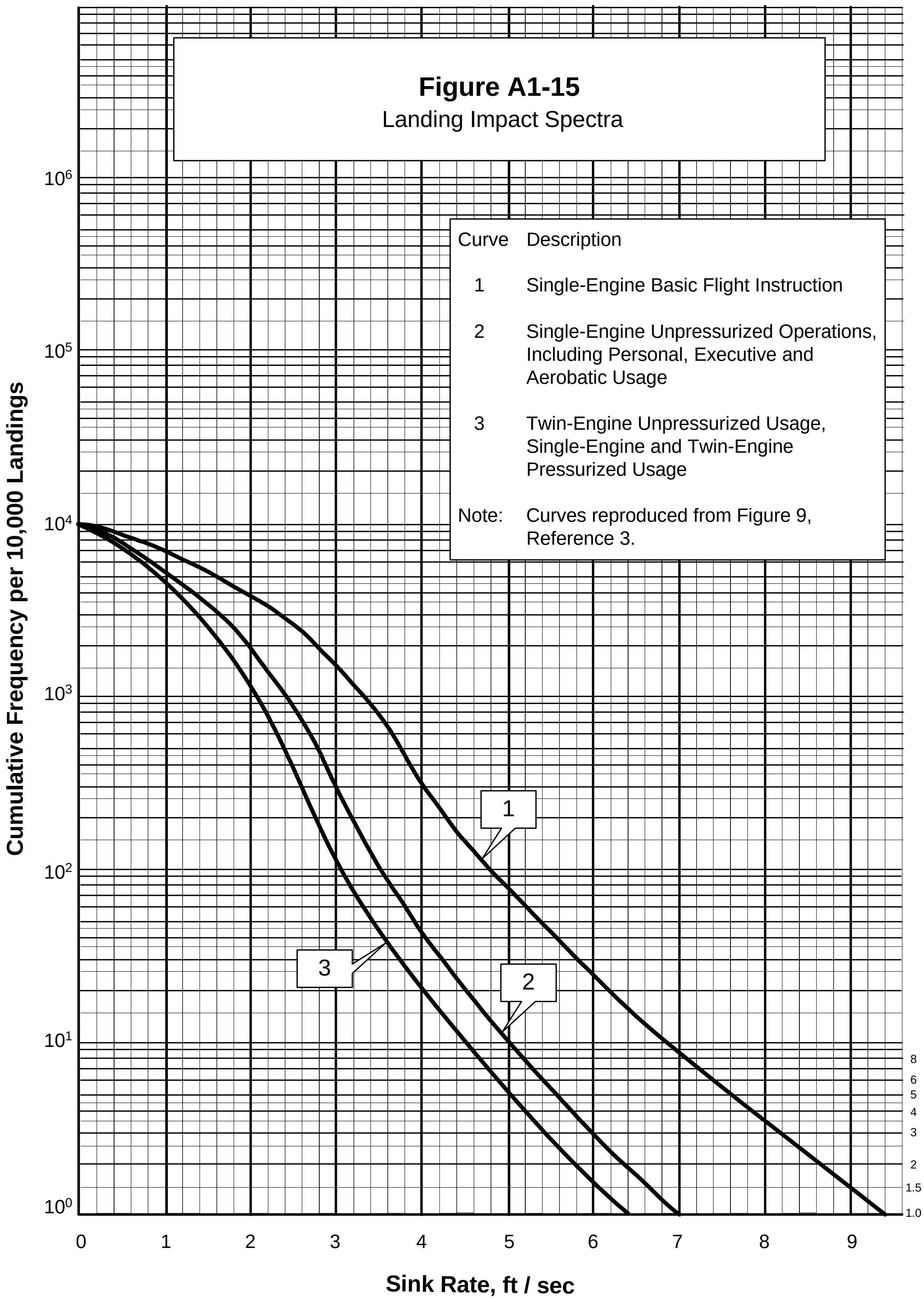


APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)

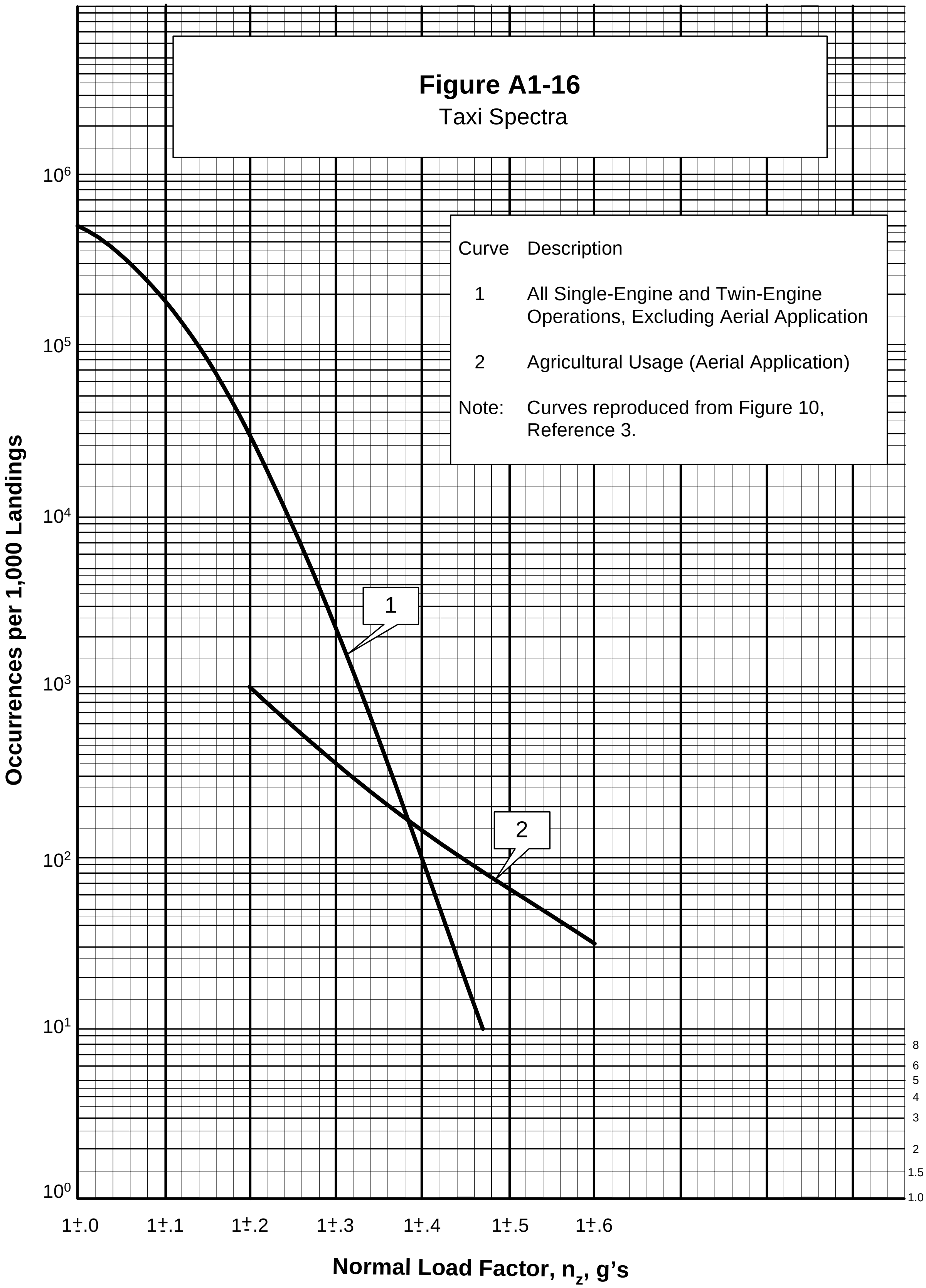
Table A1-14
Maneuver Spectra
For Survey (Pipeline Patrol) Operations

Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile		Acceleration Fraction	Cumulative Frequency of Exceedance per Nautical Mile
0.10	7.42090E-01		-0.10	1.11391E-01
0.15	4.85640E-01		-0.15	4.71406E-02
0.20	2.81225E-01		-0.20	1.35410E-02
0.25	1.53241E-01		-0.25	5.45945E-03
0.30	7.81332E-02		-0.30	2.72563E-03
0.35	3.70863E-02		-0.35	1.41442E-03
0.40	1.63236E-02		-0.40	7.52533E-04
0.45	6.64508E-03		-0.45	4.06358E-04
0.50	2.49947E-03		-0.50	2.21451E-04
0.55	8.71435E-04		-0.55	1.20991E-04
0.60	2.89471E-04		-0.60	6.60759E-05
0.65	9.39729E-05		-0.65	3.60584E-05
0.70	3.01380E-05		-0.70	1.96504E-05
0.75	9.66349E-06		-0.75	1.06815E-05
0.80	3.09532E-06		-0.80	5.77900E-06
0.85	9.89409E-07		-0.85	3.09921E-06
0.90	3.13870E-07		-0.90	1.63439E-06
0.95	9.71706E-08		-0.95	8.33698E-07
1.00	2.76574E-08		-1.00	3.96027E-07

APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



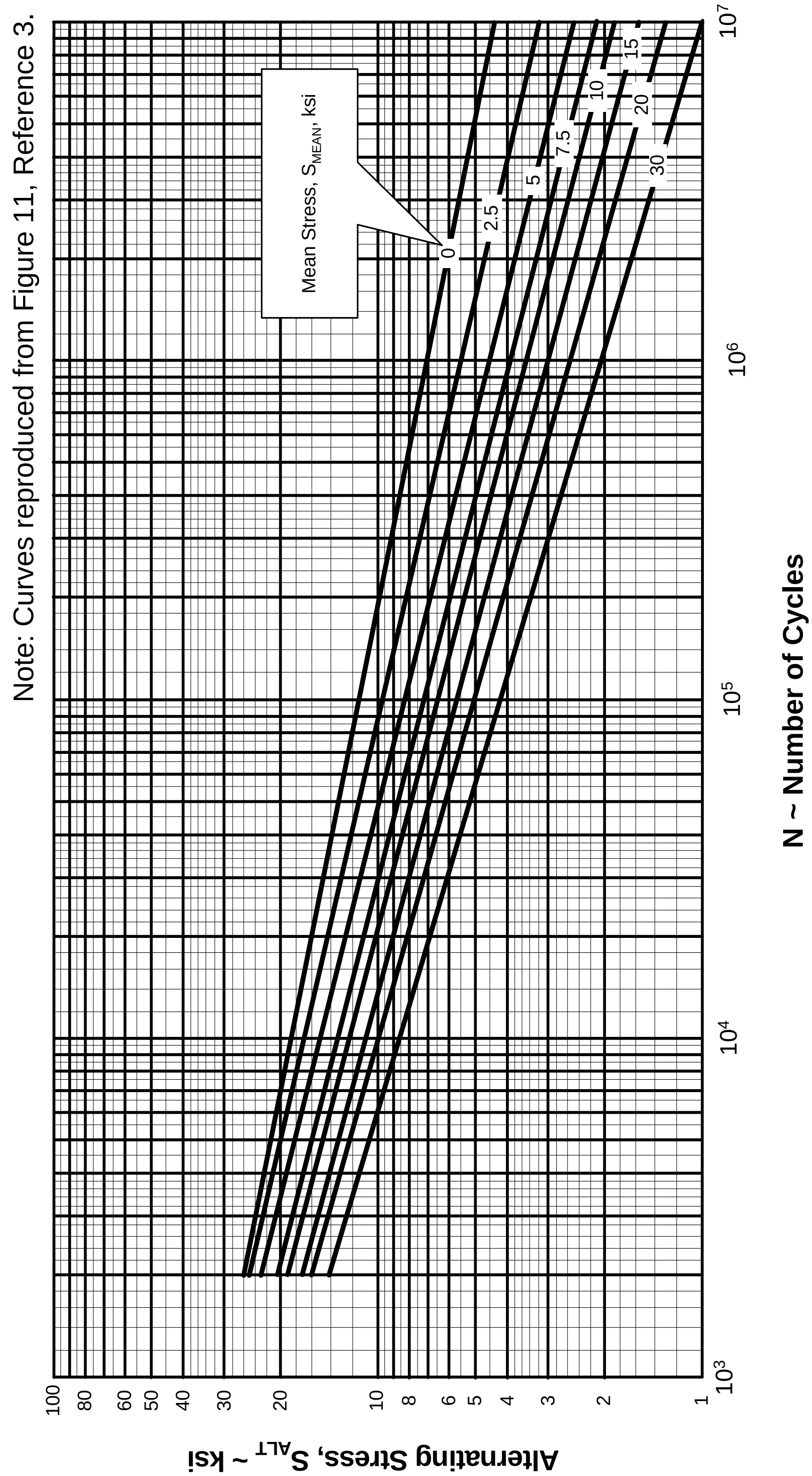
APPENDIX 1. FLIGHT AND GROUND LOAD SPECTRA (CONTINUED)



APPENDIX 2. S/N CURVES FOR ALUMINUM COMPONENTS

FIGURE A2-1

ENDURANCE OF COMPLETE WINGS AND TAILPLANES (ALUMINUM ALLOY MATERIALS)



APPENDIX 2. S/N CURVES FOR ALUMINUM COMPONENTS (CONTINUED)

FIGURE A2-2

TYPICAL CONSTANT LIFE CURVES FOR COMPLETE WINGS AND TAILPLANES (ALUMINUM ALLOY MATERIALS)