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MIL-HDBK-1783B
w/CHANGE 2
22 September 2004

SUPERSEDING
MIL-HDBK-1783B
w/NOTICE 1
31 May 2002

DEPARTMENT OF DEFENSE HANDBOOK

ENGINE STRUCTURAL INTEGRITY PROGRAM (ENSIP)

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FOREWORD

1. This handbook is approved for use by all Departments and Agencies of the Department of Defense.

2. This handbook is for guidance only. This handbook cannot be cited as a requirement. If it is, the contractor does not have to comply.

3. The purpose of this handbook is to establish structural performance, design development, and verification guidance to ensure structural integrity for engine systems. The guidance contained herein includes the experience and lessons learned achieved during development of U.S. Air Force engine systems since the mid-1940's. Recent experience indicates superior structural safety and durability, including minimum structural maintenance, can be achieved on an engine system if the guidance contained herein is included and successfully executed during system development. This handbook is intended for use in conjunction with JSSG-2007, Aircraft Turbine Engines, on engine development programs or by itself when used for commercial (off-the-shelf) acquisitions.

4. Comments, suggestions, or questions on this document should be addressed to ASC/ENOI, 2530 LOOP RD WEST, WRIGHT-PATTERSON AFB OH 45433-7101 or e-mailed to Engineering.Standards@wpafb.af.mil. Since contact information can change, you may want to verify the currency of this address information using the ASSIST Online database at www.dodssp.daps.mil.

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SUMMARY OF CHANGE 2 MODIFICATIONS

The following modifications to MIL-HDBK-1783B have been made:

Paragraph	Modification
A.4.6 – Material characterization	Added new single crystal materials Requirement Guidance
A.5.6 – Material characterization	Replaced fretting fatigue Verification Guidance subparagraphs “a.” and “b.” with subparagraphs “a.” through “f.”
A.5.6 – Material characterization	Replaced foreign object damage Verification Guidance with new information
A.4.13 – Vibration	Added new Requirement Lessons Learned
A.5.13 – Vibration	Added new Verification Lessons Learned
A.4.13.1 – Engine vibration limits	Replaced Requirement Lessons Learned with new information
A.5.13.1 – Engine vibration limits	Replaced last paragraph in Verification Guidance with mechanical impedance test guidance
A.5.13.1 – Engine vibration limits	Added new Verification Lessons Learned
A.4.13.3 – High cycle fatigue	Deleted airfoil foreign object damage/domestic object damage Requirement Guidance
A.4.13.3 – High cycle fatigue	Added new Requirement Guidance and figures 5a, 5b, 5c, and 5d
A.4.13.3 – High cycle fatigue	Added new Requirement Lessons Learned
A.5.13.3 – High cycle fatigue	Replaced Verification Guidance for laser shock peening
A.5.13.3 – High cycle fatigue	Replaced Verification Lessons Learned with new information
A.4.13.3.1 – Engine system modes	Added new Requirement Lessons Learned
A.5.13.3.1 – Engine system modes	Added new Verification Lessons Learned
A.4.13.3.2 – Component vibrations	Added new Requirement Lessons Learned
A.4.13.3.2 – Component vibrations	Added new damping Requirement Lessons Learned
A.5.13.3.2 – Component vibrations	Added new information to engine test Verification Guidance subparagraph “d)”
A.5.13.3.2 – Component vibrations	Replaced Verification Lessons Learned with new information
A.4.13.3.3 – Non-integral vibrations	Added new Requirement Lessons Learned
A.5.13.3.3 – Non-Integral vibrations	Replaced Verification Lessons Learned with new information
A.4.15 – Foreign object damage	Replaced Requirement Rationale, Guidance, and Lessons Learned with new information; added figure 10 .

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1. SCOPE

1.1 Scope.

This handbook establishes structural performance, design development, and verification guidance for turbine engines. This handbook also establishes the need for an Engine Structural Integrity Program (ENSIP). This handbook is for guidance only. This handbook cannot be cited as a requirement. If it is, the contractor does not have to comply.

1.2 Use.

This handbook cannot be used for contractual purposes without supplemental information required for specific application.

1.2.1 Structure.

The supplemental information required is identified by blanks within this handbook.

1.2.2 Instructional handbook.

The instructional handbook, which is contained in the appendix herein, provides the rationale for specific guidance, guidance for inclusion of supplemental information, and a lessons learned depository.

2. APPLICABLE DOCUMENTS

2.1 General.

The documents listed below are not necessarily all of the documents referenced herein, but are those needed to understand the information provided by this handbook.

2.2 Government documents

2.2.1 Specifications, standards, and handbooks.

The following specifications, standards, and handbooks form a part of this document to the extent specified herein.

DEPARTMENT OF DEFENSE

SPECIFICATIONS

STANDARDS

HANDBOOKS

(Copies of these documents are available online at http://assist.daps.dla.mil/quicksearch/ or www.dodssp.daps.mil or from the Standardization Document Order Desk, 700 Robbins Avenue, Building 4D, Philadelphia PA 19111-5094; telephone [215] 697-2179.)

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2.3 Non-Government publications.

The following documents form a part of this document to the extent specified herein.

3. DEFINITIONS

3.1 Definitions.

Definitions applicable to this handbook follow.

3.1.1 Cold parts.

Cold parts are parts not in the hot gas path. Cold parts are those parts not defined as hot parts.

3.1.2 Containment.

Containment is the ability of the circumferential case structure of the engine to prevent penetration of failed elements subsequent to specified conditions of primary and secondary failures.

3.1.3 Damage tolerance.

Damage tolerance is the ability of the engine to resist failure due to the presence of flaws, cracks, or other damage, for a specified period of unrepaired usage.

3.1.4 Design service life.

Design service life is the life duration specified in section 4.3.

3.1.5 Design usage.

Design usage is the usage specified in section 4.4.

3.1.6 Deterioration.

Deterioration is the gradual increase in gas temperature and corresponding specific fuel consumption at rated thrust.

3.1.7 Durability.

Durability is the ability of the engine to resist cracking (including vibration-, corrosion-, and hydrogen-induced cracking), corrosion, deterioration, thermal degradation, delamination, wear, and the effects of foreign and domestic object damage for a specified period of time.

3.1.8 Durability-critical component.

A durability-critical component is a component whose failure or deterioration will result in a significant maintenance burden, but will not impair flight safety or mission completion.

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3.1.9 Engine structure.

Engine structure encompasses all parts of the engine designed and sized to meet the structural integrity guidance of this handbook. Engine structure includes but is not limited to the following components: ducts, cases, augmentor, nozzle, blades, vanes, disks, spacers, seals, shrouds, plumbing, actuators, gears, shafts, housings, controls, and accessories (including pumps, gearboxes, oil tanks, etc.), etc.

3.1.10 Economic life.

Economic life is the operational life indicated by the results of the durability test program (i.e., test performance interpretation and evaluation in accordance with this handbook) to be available with the incorporation of U.S. Air Force-approved and committed production or retrofit changes and supporting application of the structural maintenance plan in accordance with this handbook. In general, production or retrofit changes will be incorporated to correct local design and manufacturing deficiencies disclosed by test. It will be assumed that the economic life of the test article has been attained with the occurrence of widespread damage, which is uneconomical to repair and, if not repaired, could cause functional problems which affect operational readiness. This can generally be characterized by a rapid increase in the number of damage locations or repair costs as a function of cyclic test time.

3.1.11 ENSIP (Engine Structural Integrity Program).

The Engine Structural Integrity Program (ENSIP) is an organized and disciplined approach to the structural design, analysis, qualification, production, and life management of gas turbine engines. The goal of ENSIP is to ensure engine structural safety, durability, reduced life cycle costs, and increased service readiness.

3.1.12 Expendable parts.

Expendable parts are those parts normally replaced at maintenance or overhaul, such as minor hardware, O-rings, and gaskets.

3.1.13 Fracture-critical component.

A fracture-critical component is a component whose failure will result in probable loss of the aircraft as a result of noncontainment or power loss which prevents sustained flight either due to direct part failure or by causing other progressive part failures or will result in failure to be able to complete the intended mission. Components can be further classified as safety critical or mission critical, if desired.

3.1.14 Hot parts.

Hot parts are those parts subjected to combustor exit gas flow (such as combustor liner, turbine vanes, blades, and shrouds).

3.1.15 Limit load.

The limit load is the maximum load a component is expected to encounter when operated for the design service life and design usage. The factor of safety associated with this load is defined as the limit load factor.

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3.1.16 Mission-critical component.

A mission-critical component is a fracture-critical component whose failure results in inability to complete the intended mission.

3.1.17 Operational life.

Operational life is that life expected for components when exposed to the operational usage as determined by the component life management actions specified in 4.17.

3.1.18 Operational usage.

Operational usage is the usage the engine is exposed to during actual service operation as determined by the component life management actions specified in 4.17.

3.1.19 Probabilistic design margin.

Probabilistic design margins replace traditional safety factor or deterministic margins to provide a more accurate representation of component response variations. An understanding of the nature of these variations, particularly in component vibrations, is essential to design robust components and ensure reliability or its complement probability of failure meet specified requirements. Probabilistic design margins should be used, specifically in this document, for resonant condition avoidance for final designs. For preliminary design, deterministic margins may be used. Deterministic margins may also be used when there is insufficient confidence in probabilistic solutions.

Development of probabilistic design margins requires analytical modeling and probabilistic integration methods to predict a probability distribution of the response that is used to determine probability of failure conditions. Such modeling requires sensitivity studies to determine and verify the significant random design variables of a system. Input (design) variables to the probabilistic integration methods are to be characterized in the form of statistical models or probability distributions, together with models for statistical correlations between the design variables. Sufficient data is to be generated such that usual design minimums—such as mean, – 3 Sigma, or B0.1 probability levels—can be defined. These models of the data are used with a verified analytical model and a validated probabilistic integration method to calculate the probabilistic response of the system.

The probability distributions of response are used with failure limits to define probabilities of failure. The probabilities of failure should meet the requirements of component and system reliability defined elsewhere in this document. Probabilistic design margins may be stated in terms of a probability of occurrence or failure. The margins may also be stated in terms of total fleet risk per engine flight hour or per millions of engine flight hours.

Assurance or confidence in the result is to be provided by verifying that the probability of occurrence or failure limit of the combined effects of one or more random variables is dependent on probability levels for each of the random variables that are within the range of the experimental database. Assurance is based on having the underlying physics sufficiently well modeled as defined in the Verification and Validation elements of this document. Deficiencies in this regard CANNOT be remedied by ANY statistical means.

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3.1.20 Residual strength.

Residual strength is the load-carrying capability of a component at any time during the service exposure period considering damage present and accounting for the growth of damage as a function of service exposure time. The intent of the damage tolerance requirement is to provide at least design limit load residual strength capability at all times throughout the service life of the component. The guidance to maintain limit load capability is considered necessary to allow unrestricted operational usage within the flight envelope.

3.1.21 Safety-critical component.

A safety-critical component is a fracture-critical component whose failure results in probable loss of engine or power loss that prevents sustained flight either due to direct part failure or by causing other progressive part failures.

3.1.22 Ultimate load.

Ultimate load is that load obtained by multiplying the limit load, applied singly or in combination, except loads due to thermal effects, by a factor of 1.5. In addition, when pressure loads of those components subject to compressor discharge pressure are combined with maneuver loads and loads due to thermal effects, the ultimate load will be based on the most critical condition of two times (2 X) the maximum operating pressure applied singly or one and one-half times (1.5 X) the maximum operating pressure plus maneuver loads plus loads due to thermal effects.

3.1.23 Usable life.

Usable life is the life required for hot section components prior to attainment of distress limits (low cycle fatigue, stress rupture, erosion) that cause replacement due to repair or safety considerations.

3.2 Acronyms.

Acronyms used in this handbook are defined as follow:

AMT	Accelerated Mission Test
CDR	Critical Design Review
DOD	Domestic Object Damage
EFH	Engine Flight Hours
ENSIP	Engine Structural Integrity Program
FFR	Full Flight Release
FOD	Foreign Object Damage
HCF	High Cycle Fatigue
IFR	Initial Flight Release
ISR	Initial Service Release
LCF	Low Cycle Fatigue
NDI	Nondestructive Inspection
OCR	Operational Capability Release
PDR	Preliminary Design Review.

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4. GUIDANCE

4.1 Coverage.

The guidance of this document will provide the structural performance criteria for turbine engines. This handbook includes coverage of the following:

a. Engine Structural Integrity Program (ENSIP)

b. structural performance and design development for turbine engines.

4.2 Turbine Engine Structural Integrity Program (ENSIP).

Turbine engine structural integrity requirements will be prepared to ensure the engine has adequate structural characteristics to perform the required missions for the required design service life, as specified herein. The ENSIP Master Plan will be used to define and document the specific requirements.

4.3 Design service life.

The engine will have a design service life of at least __(a)__ when subjected to the design usage of 4.4. In addition, the engine will be able to withstand __(b)__ hours at any point in the envelope for both hot and cold parts.

4.3.1 Hot parts.

Hot parts will have a usable life of __(a)__ times the design service life specified in 4.3. Hot parts and their lives will be listed in table V.

4.3.2 Cold parts.

Cold parts will have a usable life of __(a)__ times the design service life specified in 4.3. Cold parts and their lives will be listed in table VI.

4.3.3 Expendables.

The minimum life without replacement of all expendable parts and components will be equal to the minimum maintenance-free operating period. Expendable parts, components, and their lives will be listed in table VII.

4.3.4 Bearings.

The mainshaft and gearbox bearings will have B1.0 lives equal to at least the design service life of the engine. A list of bearings and their lives will be presented in table VIII.

4.3.5 Components.

Engine components will have a usable life of __(a)__ times the design service life specified in 4.3. Engine components and their lives will be listed in table IX.

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4.4 Design usage.

The engine structure will be able to withstand the design usage specified herein for the design service life specified in 4.3. The design service life and design usage will be specified in terms of mission profiles and mission mix, including nonoperating transport of the engine. Important usage parameters will be specified. The flight envelope, mission profiles, mission mix, and environment will be shown.

4.5 Operating envelope.

The engine will meet all the requirements of the document throughout the complete operating envelope without exceeding any limits. The engine operating limits will be specified for the identified environment and displayed with figures 1 and 2, and tables XIII and XIV. If applicable, the thrust augmentation operating envelope will be included on the figures.

4.5.1 Operating attitude and conditions.

The engine operating attitude limits will be shown on figure 3. The engine will meet the requirements of the specification when operated in the normal operation area of the figure, and operate at least __(a)__ seconds continuously in the limited and transient operation areas of figure 3. Operation in the limited operation area will not degrade engine performance or cause any damage. The engine will start, stop, and be stowed in any of the attitudes shown in the normal operation area of figure 3. Engine stowing capability outside the limited operation area will be specified. The engine will function satisfactorily for at least __(b)__ seconds in negative g and for at least __(c)__ seconds in zero g conditions.

4.5.2 Internal environment.

The engine components will be able to withstand the internal thermal and pressure environments that occur during engine operation (steady-state and transient conditions).

4.5.3 Externally-applied forces.

The engine will function satisfactorily and no deformation will occur during or after exposure to the externally-applied forces, which should be indicated in design load diagrams.

4.6 Material characterization.

The materials used in the engine should have such adequate structural properties as strength, creep, low-cycle fatigue, high-cycle fatigue, fracture toughness, crack growth rate, stress corrosion cracking, thermomechanical fatigue, oxidation/erosion, wear, ductility, elongation, and corrosion resistance; so that component design can meet the operational requirements for the design service life and design usage of the engine specified in 4.3 and 4.4.

4.7 Parts classification.

All engine parts, components, controls and externals, and expendables will be classified for criticality.

4.8 Damage tolerance.

Fracture-/safety- and mission-critical engine parts will be able to maintain adequate damage tolerance in the presence of material, manufacturing, processing, and handling defects for the design service life and design usage specified in 4.3 and 4.4.

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4.8.1 Residual strength.

The residual strength will be equal to the maximum stress that occurs during design usage conditions. Residual strength requirements will be established for all damage-tolerant-designed parts and components. Associated static and dynamic loading conditions for these parts and components should be included.

4.8.2 Initial flaw size.

Initial flaws will be assumed to exist as a result of material, manufacturing, and processing operations. Assumed initial flaw sizes will be based on the intrinsic material defect distribution, manufacturing process, and the nondestructive inspection (NDI) methods to be used during manufacture of the component.

4.8.3 In-service inspection flaw size.

The flaw size, which will be presumed to exist in a component after completion of a depot-, intermediate-, or base-level inspection, will be specified.

4.8.4 Inspection intervals.

The frequency of inspection in terms of the required design lifetime will be specified in terms of

a. in-service noninspectable—once at the end of one design lifetime, or

b. depot- or base-level inspectable.

4.8.5 Flaw growth.

The initial flaw sizes specified in 4.8.2 will not grow to critical size and cause failure of the part due to the application of the required residual strength load within two times (2 X) the specified inspection interval.

4.8.6 Composites.

Composite parts will be damage-tolerant with defects resulting from material quality, manufacturing processing, and handling damage.

4.9 Durability/economic life.

The durability/economic life of the engine will not be less than the required design service life when subjected to the design usage.

4.9.1 Low cycle fatigue (LCF) life.

Engine parts will have a minimum LCF life, which is at least equivalent to the design service life of 4.3.

4.9.2 Life design margin.

A life margin will be applied during design of engine components.

4.9.3 Corrosion prevention and control.

The engine will operate satisfactorily without detrimental material degradation in the environmental conditions specified in 4.5 through 4.5.3 for the design service life.

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4.10 Strength.

The engine will meet all the requirements of the specification during and after exposure to limit loads, singly and in combination. The engine will not experience catastrophic failure when subjected to ultimate loads, singly and in combination. In addition, the engine will meet the following strength criteria.

4.10.1 Factors of safety.

Factors of safety will be applied to design usage induced loads to establish limit and ultimate conditions.

4.10.2 Blade and disk deflection.

The blades and disks will not contact any static parts of the engine other than seals and shrouds, during all phases of engine operation including surge and stall occurrences. Seals and clearances will remain effective under all internal and external operational loads.

4.10.3 Containment.

Uncontained failures will not cause fire or catastrophic damage to engine external systems or aircraft systems, or injury to personnel.

4.10.4 Blade out.

Subsequent to a single blade failure, with resulting secondary loss of another blade in the same stage at maximum allowable transient speed, the engine will not experience uncontained fire; catastrophic rotor, bearing, support, or mount failures; overspeed conditions; leakage from flammable fluid lines; or loss of ability to shutdown the engine.

4.10.5 Overspeed/overtemperature.

The engine will meet all the requirements of the specification during and after overspeed and overtemperature conditions.

4.10.6 Disk burst speed.

The minimum loaded disk burst speed of the complete disk assembly will be greater than or equal to the overspeed requirements of 4.10.5.

4.10.7 Output shaft torque limits.

The maximum allowable steady-state delivered shaft torque (mechanical) limit for turboprop and turboshaft engines will be at least _(a)_ percent greater than the rating value.

4.10.8 Output shaft speed limits.

The maximum allowable steady-state delivered shaft speed (mechanical) limit for turboprop and turboshaft engines will be at least _(a)_ percent greater than the rating value. The shaft will be able to operate at this speed for at least _(b)_ and function satisfactorily thereafter. Following loss of load, the output shaft speed will not exceed the maximum shaft speed predicted with the engine at Intermediate power and the output shaft running at the maximum attainable rotor speed.

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4.10.9 Pressure vessel/case.

All engine cases and pressure-loaded parts and components will withstand the ultimate loading conditions defined in 4.10.1. The cases should remain intact, although permanent deformation and distress which require repair or replacement is permitted. Engine cases will not fail due to combustion process burning or erosion.

4.10.10 Pressure balance.

The engine thrust bearings will provide sufficient thrust load to ensure satisfactory bearing operation without skid damage during the design service life.

4.10.11 Gyroscopic moments.

The engine will meet all the requirements of the specification at maximum allowable steady-state engine speeds when subjected to the rotational velocities and accelerations within the flight envelope and the gyroscopic moment conditions.

4.10.12 Main mounts.

The engine mounts will have adequate strength to retain the engine, including retained fluids and externals, at all flight, takeoff and landing, and ground conditions.

4.10.13 Ground-handling mounts.

The ground-handling mounts will support the engine, including all engine-mounted equipment and externals, components, and operating fluids, under the following maximum inertia load conditions, without deformation to the mounts or damage to the engine: __(a)__ axial, __(a)__ lateral, and __(a)__ vertical acting in combination at the engine center of gravity.

The locations and descriptions for the individual ground-handling mounts will be specified. The arrangement will be compatible with ground-handling equipment specified herein by the Using Service.

4.10.14 Engine stiffness.

The estimated stiffness of the engine in resisting loads and moments applied at the outboard end of the output shaft, relative to the engine mounting points, will be specified herein. The first "free-free" lateral and vertical engine bending modes will be specified herein.

4.11 Deterioration.

The engine will be able to attain the hot-part design life when operated at temperature conditions which represent a typical rate of performance deterioration. The temperature margin above the production acceptance engine maximum steady-state gas temperature under standard day conditions will be consistent with that required for the engine as stated in the engine specification for the design service life of 4.3.

4.12 Creep.

The engine static and rotating parts will not creep to the extent that acceptable field engine operation is impaired for the operating conditions and the lifetime specified in 4.3. Part creep will not affect disassembly and reassembly of the engine or new part replacement at overhaul throughout the specified life of the engine.

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4.13 Vibration.

The engine as defined in 3.1.9 should be free of destructive vibration at all engine speeds and thrusts (including steady-state and transient conditions) within the flight and ground envelope.

4.13.1 Engine vibration limits.

Maximum engine mechanical vibration limits should be established as a function of frequency, engine order, and location and direction of measurement. Maximum engine mechanical vibration limits should be based on an acceptable margin of safety for the structural capability.

Any rotor critical speeds which excite detrimental engine system modes existing above or below the engine operating range should have a probabilistic design margin established on speed to account for the variation in speeds for different operating conditions, new engine performance, minimum engine, and fully-deteriorated engine. The system modes should be defined within the expected engine operating envelope from production until being pulled from service for low performance. Adequate damping and appropriate balancing should be provided so that any critical speed existing below maximum operating speed should be traversed safely with smooth engine operation. The variation in speeds based on operating conditions, etc., should be included.

4.13.2 Surge and stall.

The engine should operate satisfactorily without structural degradation which would cause the engine to not meet the requirements of this document for the design service life in the event of surges and stalls within the flight envelope.

4.13.3 High cycle fatigue (HCF).

The probability of failure due to high cycle fatigue (HCF) for any component within or mounted to the engine should be below $1x10^{-7}$ per EFH on a per-stage basis, provided the system-level safety requirements are met.

4.13.3.1 Engine system modes.

The engine should be free of detrimental resonance conditions at all speeds in the operating range. Any rotor critical speeds which excite detrimental engine system modes which exist above or below the engine operating range should have a probabilistic design margin established on speed to account for the variation in speeds for different operating conditions, new engine performance, minimum engine, and fully-deteriorated engine. The system modes should be defined within the expected engine operating envelope from production until being pulled from service for low performance. Adequate damping and appropriate balancing should be provided so that any critical speed that exists below maximum operating speed should be traversed safely with smooth engine operation. The variation in speeds based on operating conditions, etc., should be included. Operational conditions that are not part of the expected operational range but do exhibit significant stresses should be clearly documented to assist the USAF in future assessment of the engine's ability to perform changing missions.

4.13.3.2 Component vibrations.

Engine components should be free of detrimental resonance at all speeds in the operating range. This can be accomplished by intentionally designing modes out of the engine operating speed range or by providing sufficient damping, a probabilistic design margin on frequency and a probabilistic prediction of vibratory stress with respect to steady-state operating speeds, or excitation control to ensure that modes which remain in the running range do not respond detrimentally. A detrimental response is one that exceeds criteria outlined in A.4.13.3.

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4.13.3.3 Non-integral vibrations.

Self-excited vibrations such as flutter, separated flow vibration, or other non-integral vibrations should not occur within the operating range of the engine. In the specific case of flutter, the engine will encounter other limits before flutter. For example, stall boundaries should be encountered before stall flutter.

4.14 Noise.

The engine will meet the strength and design service life requirements in the presence of the noise environment produced during installed and uninstalled operation at the flight and ground operating conditions consistent with the design usage conditions.

4.15 Foreign object/domestic object damage (FOD/DOD).

The engine will operate satisfactorily when foreign objects/domestic objects are ingested.

4.16 Structural maintainability.

The engine will be economically maintainable for the design service life and design usage of 4.3. Engine components will fit and function with new components after being operated to the design service life and design usage of 4.3. The function of structural components, elements, and major bearing surfaces will not be degraded by wear, erosion, or corrosion to the extent that performance or structural capability will be impaired. Authorized repairs will be established for critical components that experience detrimental wear, erosion, or corrosion during developmental testing and service operation. The structural life of repaired components specified by the contractor will be equal to or greater than the inspection intervals set forth in 4.8.4. Any repairs should be structurally sound and cost effective.

4.17 Inspectability.

Critical engine components will be inspectable by use of borescope ports and diagnostic methods so that detrimental damage or other deterioration will be detected to facilitate economical repair and to prevent engine failure. A listing of the inspectable components and their methods of inspection will be specified.

4.18 Engine/airframe structural compatibility.

The engine will meet the structural requirements of this document when installed in the airframe. The installed engine will operate satisfactorily in the thermal and aerodynamic environment produced by the engine/airframe configuration. The installed engine will possess flutter margin throughout the engine flight envelope.

4.19 Component life management.

Required maintenance actions (component inspection, repair, or replacement requirements) will be defined to ensure adequate structural integrity and operational readiness of each engine for the design service life. Required maintenance actions will be based on duty cycles defined by operational usage of the airframe/engine. Individual component maintenance times will be based on the parameter that causes life degradation.

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5. VERIFICATIONS

5.1 General.

The evaluation (inspections/analyses/tests) specified herein will verify conformance with the guidance of section 4 herein. All evaluations will be the responsibility of the contractor; the Government reserves the right to witness, or conduct, any evaluation.

5.2 Turbine engine structural integrity evaluation program.

The ENSIP Master Plan will be used to define and document the specific evaluation tasks.

5.3 Design service life.

The requirements of 4.3 will be evaluated by analysis, inspection, demonstration, and test.

5.3.1 Hot parts.

The requirement of 4.3.1 will be evaluated by analyses and tests.

5.3.2 Cold parts.

The requirement of 4.3.2 will be evaluated by analyses and tests.

5.3.3 Expendables.

The requirement of 4.3.3 will be evaluated by analyses and tests.

5.3.4 Bearings.

The requirement of 4.3.4 will be evaluated by analyses and tests.

5.3.5 Components.

The requirement of 4.3.5 will be evaluated by analyses and tests.

5.4 Design usage.

Verification of design usage will be accomplished by analysis, design development tests, and engine tests, in accordance with the ENSIP Master Plan to ensure that the engine and its components meet the design service life and design usage requirements of 4.3 and 4.4. A design duty cycle(s) will be derived from the design service life and design usage specified in 4.3 and 4.4. The design duty cycle will be supplied.

5.5 Operating envelope.

The requirements of 4.5 will be evaluated by analysis, demonstration, and test.

5.5.1 Operating attitude and conditions.

The requirements of 4.5.1 will be evaluated by analysis, demonstration, and test.

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5.5.2 Internal environment.

Evaluation of the capability of the engine components to withstand the internal thermal and pressure environments that occur during engine operation will be evaluated by analysis and test.

5.5.3 Externally-applied forces.

Verification of flight and ground externally-applied forces will be in accordance with 4.5.3, and will be evaluated by analysis and test.

5.6 Material characterization.

Material structural properties should be established by test and modeling. Anticipated properties under damage states (e.g.; fretting, etc.) should be verified through combinations of laboratory specimen, sub-element and component testing, material damage models which have been validated against databases and supplemented with historical data which cover the range of potential damage states, or databases which cover the properties under damage states.

5.7 Parts classification.

The requirement of 4.7 will be evaluated by analysis, inspection, and test.

5.8 Damage tolerance.

Damage tolerance of fracture-critical engine components will be in accordance with 4.8. Verification will be evaluated by analysis and test.

5.8.1 Residual strength.

The requirements of 4.8.1 will be evaluated by analyses and tests.

5.8.2 Initial flaw size.

Material controls, manufacturing process controls, and in-process nondestructive inspection (NDI) will be performed on each fracture-critical component to ensure the requirements of 4.8.2 are met.

5.8.3 In-service inspection flaw size.

The requirements of 4.8.3 will be evaluated by analysis, inspection, demonstration, and test.

5.8.4 Inspection intervals.

The requirements of 4.8.4 will be evaluated by analyses and tests.

5.8.5 Flaw growth.

The requirements of 4.8.5 will be evaluated by analyses and tests.

5.8.6 Composites.

The requirements of 4.8.6 will be evaluated by analyses and tests.

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5.9 Durability.

The requirements of 4.9 will be evaluated by a strength and life analysis, inspection, demonstration, and part, component, and full-scale engine tests.

5.9.1 Low cycle fatigue (LCF) life.

The requirement of 4.9.1 will be evaluated by analyses and tests.

5.9.1.1 Accelerated mission test (AMT).

An accelerated mission test (AMT) will be performed on the initial flight release (IFR) engine configuration. The test run schedule will simulate the design duty cycle of 5.3. The minimum test duration will be two times (2 X) the initial flight test usage. This test will be completed prior to first flight.

5.9.1.2 Full-scale development engine.

An AMT will be performed on the full-scale development engine configuration. The test schedule will simulate the design duty cycle of 5.3. The minimum test durations will be one-half the design service life at full flight release (FFR) and one times (1 X) the design service life at initial service release (ISR).

5.9.1.3 Production-tooled engine.

Accelerated Mission Test will be performed on a production-tooled engine configuration. The test schedule will simulate the design duty cycle of 5.3. The minimum test duration will be one times (1 X) the design service life at operational capability release (OCR). Accelerated Mission Test of any proposed design changes will be conducted to a duration of one times (1 X) the design service life at OCR.

5.9.1.4 Production-tooled engine configuration.

Accelerated Mission Test will be performed on a production-tooled engine configuration. The test schedule will simulate a service duty cycle that is derived from operational usage data. The minimum test duration will be one times (1 X) the design service life.

5.9.1.5 Inspections.

Major inspection programs will be conducted as an integral part of the AMT programs.

5.9.1.6 Interpretation and evaluation of test results.

Each structural problem, such as failure, cracking, yielding, wear, and erosion, discovered during inspection of the AMT engines will be analyzed to determine cause, corrective action, and operational implications relative to meeting the design requirements contained in this handbook. Specific requirements will be identified.

5.9.2 Life design margin.

Attainment of the life design margin will be evaluated by analysis and test.

5.9.3 Corrosion prevention and control.

The corrosion resistance of the engine materials, processes, and protection systems will be evaluated as follows: _____.

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5.10 Strength.

The requirements of 4.10 will be evaluated by structural analysis and part, component, and full-scale engine tests.

5.10.1 Factors of safety.

The requirements of 4.10.1 will be evaluated by analyses and tests.

5.10.2 Blade and disk deflection.

The requirements of 4.10.2 will be evaluated by analyses and tests.

5.10.3 Containment.

The requirements of 4.10.3 will be evaluated by analysis and test.

5.10.4 Blade out.

The requirements of 4.10.4 will be evaluated by analysis and test.

5.10.5 Overspeed/overtemperature.

The requirements of 4.10.5 will be evaluated by analysis and test.

5.10.6 Disk burst speed.

The requirements of 4.10.6 will be evaluated by analysis and test.

5.10.7 Output shaft torque limits.

The requirements of 4.10.7 will be evaluated by analysis and test.

5.10.8 Output shaft speed limits.

The requirements of 4.10.8 will be evaluated by analysis and test.

5.10.9 Pressure vessel/case.

The requirements of 4.10.9 will be evaluated by analyses and tests.

5.10.10 Pressure balance.

The requirement of 4.10.10 will be evaluated by analysis and test.

5.10.11 Gyroscopic moments.

The requirements of 4.10.11 will be evaluated by analysis and test.

5.10.12 Main mounts.

The requirements of 4.10.12 will be evaluated by analysis and test.

5.10.13 Ground handling mounts.

The requirements of 4.10.13 will be evaluated by analysis, demonstration, and test.

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5.10.14 Engine stiffness.

The requirements of 4.10.14 will be evaluated by analysis, demonstration, and test.

5.11 Deterioration.

Capability of engine components to attain hot section part life under deterioration conditions will be evaluated as follows:

5.11.a Analysis.

Analysis of LCF, creep, stress rupture, and erosion capability accounting for the required temperature margin above maximum steady-state gas temperature will be performed.

5.11.b Performance.

Component structural performance during conduct of the several engine tests will be verified.

5.12 Creep.

Creep characteristics of the engine static and rotating parts will be verified per 5.12.a through 5.12.c.

5.12.a Analysis.

An analysis will be performed to demonstrate that sustained stress and temperature combinations will not result in detrimental permanent set/growth for the required design service life and design usage.

5.12.b Test.

A design development test plan and tests for creep evaluation will be developed and performed.

5.12.c Inspection.

Inspection and evaluation of components will be performed subsequent to conduct of the several engine tests required by this handbook. These inspections will as a minimum be equivalent to the field and depot inspections.

5.13 Vibration.

Vibration characteristics of the engine (ref. 3.1.9) should be verified by analysis and test. The variation in vibration characteristics should be validated by probabilistic analysis and test.

5.13.1 Engine vibration limits.

Verification of engine vibration limits should be by analysis and test. An analytical dynamic analysis of the engine and accessories should be performed to identify critical engine system modes, potential forcing functions, and resonance conditions. This model should be verified with engine testing. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance is to be provided by verifying that the probability levels for each contributing random variable used to compute probabilistic design margins or probability of failure are within the experimental data range for that variable.

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5.13.2 Surge and stall.

Verification of the engine's capability to withstand surges and stalls should be by analysis and test.

5.13.3 High cycle fatigue.

Verification of the engine's ability to withstand high cycle fatigue should be through analysis and test. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance is to be provided by verifying that the probability levels for each contributing random variable are within the experimental data range for that variable.

5.13.3.1 Engine system modes.

An analytical dynamic model of the engine and accessories should be performed to identify critical engine system modes, potential forcing functions, and resonance conditions. This model should be verified with engine testing. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance in these margins are to be provided by verifying that the probability levels for each contributing random variable used to compute probabilistic design margins or probability of failure are within the experimental data range for that variable.

5.13.3.2 Component vibrations.

Verification of model validity, modal characteristics, vibration amplitudes, steady stresses, and all other aspects of the HCF problem should be performed at each step of the design and verification process. An integrated approach where each stage of the design/verification process builds upon the previous one should be utilized. Verification should include numerical verification (sensitivities to key parameters), and data generated in component bench testing, rig testing, engine testing, and, ultimately, operational use. Established methods to compare experimental and analytical results should be employed where possible. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance is to be provided by verifying that the probability levels for each contributing random variable used to compute probabilistic design margins or probability of failure are within the experimental data range for that variable.

5.13.3.3 Non-integral vibrations.

Verification of sufficient flutter margin should be by analysis and test. To the maximum extent possible, prediction of flutter boundaries using physics based models should be performed prior to engine test. If physics based models are unavailable, empirical models or databases should be used. Analytical predictions should be verified using engine tests under simulated altitude conditions. The effects of mis-rigging and engine deterioration should be evaluated to insure that flutter is not encountered during start-up or at other operational speeds at any flight condition.

5.14 Noise.

The capability of the engine to meet the strength and durability requirements in the presence of the noise environment generated during engine operation will be verified by test. Specific tests required by this document that will be used to demonstrate compliance with the noise requirement of 4.14 will be as follows: _____.

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5.15 Foreign object/domestic object damage (FOD/DOD).

Evaluation of the capability of the engine to meet the foreign object/domestic object damage requirements will be by analysis and test.

5.16 Structural maintainability.

Maintainability of the engine will be verified per 5.16.a and 5.16.b.

5.16.a Inspection.

Inspection and evaluation of changes in critical dimensions and finish of components after conduct of the several engine tests detailed by this handbook. A maintainability assessment plan will be developed and implemented.

5.16.b Test.

Structural life of component repair procedures will be verified by test, as required.

5.17 Inspectability.

The ability to accomplish inspection requirements established by 4.17 will be verified during conduct of the engine tests detailed by this handbook.

5.18 Engine/airframe compatibility.

Engine/airframe compatibility will be verified by an instrumented engine test installed in the aircraft. The scope of these tests will be contained in the Interface Control Document.

5.19 Component life management.

Component life management will be defined and implemented by analysis, test, and recording of the operational usage of the engine as follows:

5.19.a Plan.

A structural maintenance plan will be prepared.

5.19.b Data recording.

Engine signals will be provided to the airframe data recording system to record parameters required to establish operational usage duty cycles for the engine. The data recording system will record the following parameters: _____.

5.19.c Counter.

Each engine will contain a counter which will record parameter events that control the structural limits of engine components. The counter will record the following events: _____.

5.19.d Tracking program.

A critical component tracking program plan will be established. This system will define the analysis procedures, serialization, data collection, and computer programs necessary to establish maintenance times of individual components based on accrual of parameter events.

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6. NOTES

6.1 Intended use.

This document should be used as guidance for identifying the structural integrity characteristics of all propulsion systems for military acquisition, which includes the acquisition of commercial off-the-shelf propulsion systems.

6.2 Subject term (key word) listing.

AMT
Durability
Fatigue
Fracture
Life Management
Propulsion

6.3 Change notations.

The margins of this handbook are marked with vertical lines to indicate modifications generated by this change. This was done as a convenience only and the Government assumes no liability whatsoever for any inaccuracies in these notations.

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APPENDIX

ENGINE STRUCTURAL INTEGRITY PROGRAM (ENSIP)

A.1 SCOPE

A.1.1 Scope.

This appendix provides propulsion structural integrity rationale, guidance, lessons learned, and instructions necessary to tailor sections 4 and 5 of the basic document (MIL-HDBK-1783B) for a specific application.

A.1.2 Purpose.

This appendix provides information to assist the Government Procuring Activity in the use of MIL-HDBK-1783B.

A.1.3 Use.

This appendix is designed to help the project engineer tailor MIL-HDBK-1783B. This handbook provides guidance on performance requirements to be provided by the Procuring Activity in the Request for Proposal (RFP) and those verification tasks to be identified by the contractor in response to the RFP.

A.1.4 Format

A.1.4.1 Requirement/verification identity.

Section A.4 and section A.5 of this appendix parallel sections 4 and 5 of the basic handbook; paragraph titles and numbering are in the same sequence. Sections A.4 and A.5 provide each requirement (section A.4) and associated verification (section A.5) as stated in the basic handbook. Both the requirement and verification have sections for rationale, guidance, and lessons learned.

A.1.4.2 Requirement/verification package.

Sections A.4 and A.5 of this appendix have been arranged so that the requirement and associated verification is a complete package to permit addition to, or deletion from, the criteria as a single requirement. A requirement is not specified without an associated verification.

A.1.5 Responsible engineering office.

The Responsible Engineering Office (REO) for this appendix is ASC/ENFS (MR. VINCE SPANEL), 2530 LOOP ROAD WEST, WRIGHT-PATTERSON AFB OH 45433-7101; DSN 785-8515, COMMERCIAL (937) 255-8515.

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A.2 APPLICABLE DOCUMENTS

A.2.1 General.

Unless otherwise indicated, the documents specified herein are referenced solely to provide supplemental technical data.

A.2.2 Government documents

A.2.2.1 Specifications, standards, and handbooks.

The following specifications, standards, and handbooks form a part of this document to the extent specified herein.

DEPARTMENT OF DEFENSE

SPECIFICATIONS

JSSG-2007	Engines, Aircraft, Turbine
AFGS-87233	Support Systems and Equipment

HANDBOOKS

MIL-HDBK-5	Metallic Materials and Elements for Aerospace Vehicle Structures *(cancelled)*
MIL-HDBK-1530	Aircraft Structural Integrity Program
MIL-HDBK-1823	Nondestructive Evaluation System, Reliability Assessment

(Copies of these documents are available online at http://assist.daps.dla.mil/quicksearch/ or www.dodssp.daps.mil or from the Standardization Document Order Desk, 700 Robbins Avenue, Building 4D, Philadelphia PA 19111-5094; telephone [215] 697-2179.)

A.2.2.2 Other Government documents, drawings, and publications.

The following other Government documents, drawings, and publications form a part of this document to the extent specified herein.

MCIC-HB-01	Damage Tolerance Design Handbook
AFWAL-TR-81-2045	Damage Tolerant Design for Cold Section Turbine Engine Disks, June 1981
AFWAL-TR-83-2079	Weibull Analysis Handbook
ASD-TR-82-5012	Handbook of Military Aircraft Design Normal Load Factor Exceedance Data
	Scientific Advisory Board (SAB) Special Report on Turbine Engines, January 1976

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	Scientific Advisory Board (SAB) Special Report on HCF in Turbine Engines, October 1992
	DoD Procurement Management Review, "Aircraft Gas Turbine Engine Acquisition and Logistics Support," February 1976
AFRL-ML-WP-TR-2001-4159	Improved High Cycle Fatigue (HCF) Life Prediction
GAO Report PSAD-80-72	"Are Management Problems in the Acquisition of Aircraft Gas Turbine Engines Being Corrected?" September 30, 1980

(Application for copies of MCICs should be addressed to Advanced Materials and Process Technology Information (AMPTIAC), 201 Mill Street, Rome NY 13440-6916 amptiac.iitri.org; TR requests should be addressed to National Technical Information Service (NTIS), 5285 Port Royal Rd, Springfield VA 22161-0002, www.ntis.gov; SAB report requests should be addressed to HQ USAF/SB,1180 Air Force Pentagon, Rm. 5D982, Washington DC 20330, www.sab.hq.af.mil; GAO report requests should be addressed to General Accounting Office Headquarters, 700 4th St., NW, Washington DC 20001-2608, www.gao.gov.)

The following documents are not referenced in this appendix, but provide supplemental information to the extent specified herein.

AFFDL-TR-79-3021	USAF Damage Tolerant Design Handbook: Guidelines for the Analysis and Design Tolerant Aircraft Structures, March 1979

Cowie, W.D. "Turbine Engine Structural Integrity Program (ENSIP)." *Journal of Aircraft.* Volume 12, Number 4, April 1975, 366-369.

Tiffany, C.F. and Cowie, W.D. "Progress on the ENSIP Approach to Improved Structural Integrity in Gas Turbine Engines/An Overview." *The American Society of Mechanical Engineers.* 78-WA/GT-13, August 1978.

(Application for copies should be addressed to ASC/ENFS, ATTN: VINCE SPANEL, WRIGHT-PATTERSON AFB OH 45433-7101; DSN 785-8515, COMMERCIAL (937) 255-8515.)

A.2.3 Non-Government publications.

The following documents form a part of this document to the extent specified herein.

El Haddad, M.H.; Smith, K.N.; and Topper, T.H. "Fatigue Crack Propagation of Short Cracks." *Jour. Eng. Mat. Tech. 101*. 1979, 42-46.

Fransson, T., and Sieverding, C. H., Eds. Aeroelasticity in Axial-Flow Turbomachines. LS 1999-05, von Karman Institute for Fluid Dynamics, 1999.

Harris, Cyril M. *Shock and Vibration Handbook*. 4th ed, NY: McGraw-Hill Book Co, 1995.

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Platzer, M.F. and Carta, F.O., Eds. AGARD Manual on Aeroelasticity in Axial-flow Turbomachines. AGARDograph No. 298, NATO, Advisory Group for Aerospace Research and Development, Neuilly sur Seine, France, 2 Vols. (Vol. 1: Unsteady Turbomachinery Aerodynamics, March, 1987. Vol. 2: Structural Dynamics and Aeroelasticity, June, 1988.)

A.3 DEFINITIONS

The acronyms introduced in this appendix are defined as follow:

a. AFWAL - Air Force Wright Aeronautical Laboratory
b. ASIP - Aircraft Structural Integrity Program
c. CDRL - Contract Data Requirements List
d. CIP - Component Improvement Program
e. CL - Confidence Level
f. CLDS - Constrained Layer Viscoelastic Damping Systems
g. EDM - Electro-Discharge Machining
h. EMD - Engineering and Manufacturing Development
i. EoC - Edge of Contact
j. ERA - Eigenvalue Realization Algorithm
k. FAA - Federal Aviation Administration
l. FAR - Federal Acquisition Regulation
m. FEM - Finite Element Model
n. FMECA - Failure Mode and Effects Criticality Analysis
o. FPI - Fluorescent Penetrant Inspection
p. FSED - Full-Scale Engineering Development
q. HEX - High Energy X-ray
r. HPC - High Pressure Compressor
s. HPT - High Pressure Turbine
t. IBR - Integrally-Bladed Rotor
u. ICD - Interface Control Document
v. IGV - Inlet Guide Vane
w. IRP - Intermediate Rated Power
x. LCC - Life Cycle Cost
y. LPB - Low Plasticity Burnishing
z. LPT - Low Pressure Turbine
aa. LSP - Laser Shock Peening

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bb.	MAC	-	Modal Assurance Criteria
cc.	MIMO	-	Multiple Input, Multiple Output
dd.	MISO	-	Multiple Input, Single Output
ee.	NDT	-	Nondestructive Testing
ff.	NRIFSD	-	Non-Recoverable In-flight Shut Down
gg.	NSMS	-	Non-interference Stress Measurement System
hh.	OMC	-	Organic Matrix Composite
ii.	PDF	-	Probability Density Function
jj.	PFD	-	Polyreference Frequency Domain
kk.	PLA	-	Power Lever Angle
ll.	POD	-	Probability of Detection
mm.	PTD	-	Polyreference Time Domain
nn.	PTO	-	Power Take-Off
oo.	REO	-	Responsible Engineering Office
pp.	RFP	-	Request for Proposal
qq.	RMS	-	Root Mean Square
rr.	RPM	-	Revolutions per Minute
ss.	SFC	-	Specific Fuel Consumption
tt.	SIMO	-	Single Input, Multiple Output
uu.	SON	-	Statement of Need
vv.	TOT	-	Total Operating Time
ww.	USA	-	United States Army
xx.	USAF	-	United States Air Force
yy.	USN	-	United States Navy
zz.	VEM	-	Viscoelastic Material
aaa.	XRD	-	X-ray Diffraction.

A.4 GUIDANCE

A.4.1 Coverage.

The guidance of this document will provide the structural performance criteria for turbine engines. This handbook includes coverage of the following:

a. Engine Structural Integrity Program (ENSIP)

b. structural performance and design development for turbine engines.

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A.4.1.1 VERIFICATION

A.4.1.1.1 General.

The evaluation (inspections/analyses/tests) specified herein will verify conformance with the guidance of section A.4 herein. All evaluations will be the responsibility of the contractor; the Government reserves the right to witness, or conduct, any verification.

A.4.2 Turbine Engine Structural Integrity Program (ENSIP).

Turbine engine structural integrity requirements will be prepared to ensure the engine has adequate structural characteristics to perform the required missions for the required design service life, as specified herein. The ENSIP Master Plan will be used to define and document the specific requirements.

REQUIREMENT RATIONALE (A.4.2)

In past years, numerous structural problems have occurred in USAF turbine engines. Many of these problems resulted in loss of aircraft and an even greater number have affected durability, which has caused high maintenance and modification costs. All of the problems have adversely affected operational readiness. These problems have highlighted the need for a disciplined approach to turbine engine structural development. The need has been identified by no less than 23 studies, assessments, and investigations conducted during 1970 - 1995 to review aircraft engine development, management, and acquisition. The Turbine Engine Structural Integrity Program (ENSIP) is intended to reduce these problems and was established by the U.S. Air Force to provide an organized and disciplined approach to the structural design, analysis, qualification, production, and life management of gas turbine engines. The ENSIP is organized into five (5) basic tasks as shown in table I and described herein.

(Task I) Design information. Detailed structural design criteria and design usage will be applied during engine material selection and structural design to meet operational needs and requirements. Initial usage definition will be supplied by the Procuring Activity.

(Task II) Design analyses, material characterization, and development tests. Design analyses will be performed to determine the environments (load, temperature, vibratory, acoustic, and chemical) to which the engine structure will be exposed during operation and transport. Design analyses, material characterization, and development tests will be performed to design and size the components.

(Task III) Component and core engine testing. Component tests will be performed to assess strength, damage tolerance, durability, and dynamic characteristics. Thermal, vibratory, and flutter boundary surveys will be performed during core engine tests.

(Task IV). Ground and flight engine tests. Ground and flight engine tests will be performed to verify the environment in the full-scale engine under steady-state and transient conditions and to verify damage tolerance and durability. Types of tests to be performed will include: ground vibration, temperature, and flutter surveys; external components resonant tests and clearance control tests; and accelerated mission tests. These tests will include measurement of steady-state and transient conditions including shutdown and cool-down parameters. Installed engine tests will be performed. Telemetry capability will be provided.

(Task V) Engine life management. A data package, monitoring equipment, and analysis methods will be provided so the Air Force can accomplish the required life management actions. Requirements will include updated strength and life analyses, structural

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maintenance plan, mission utilization recorder, and critical parts tracking system including individual engine recorder. This task contains basic ENSIP requirements to be performed by the contractor but, unlike Tasks I through IV, will not be for the purpose of providing compliance to the operational requirements. Tasks scheduled after full-scale development (FSD) will be identified by the contractor.

The major subtasks or elements contained in each of the five tasks are also shown in table I.

REQUIREMENT GUIDANCE (A.4.2)

The contractor should identify the engine structural integrity requirements tailored to meet the needs of each engine development program. Specific guidance and suggested requirements for identifying supplemental information are contained in this handbook for specific structural requirements. In general, the following guidance should be followed:

a. Design stress spectra, component test spectra, and full-scale engine test spectra should be based on anticipated service usage of the engine.

b. Materials and processes should be thoroughly characterized including fracture properties.

c. It is not realistic to assume defect-free structure in fracture-critical components.

d. Cost considerations make it important to extend the useful life of engine components when it can be done without jeopardizing safety. This philosophy is called, “retirement-for-cause.” Considerations for this activity are covered under Task V (Life Management).

e. Critical parts (and part details) and potential failure modes should be identified early, and appropriate control measures implemented.

f. Internal thermal and vibratory environments should be identified early in the engine development.

g. Predicted analytical stresses should be verified by test for critical components, where practical.

h. Potential engine/airframe structural interactions should be defined and accounted for.

i. Closed-loop force management procedures should be defined and implemented. This includes realistic inspection and maintenance requirements, individual engine tracking procedures, deficiency reporting, and updates based on actual usage.

j. Life verification test results should be available to support production decisions.

REQUIREMENT LESSONS LEARNED (A.4.2)

Refer to A.4.2 Requirement Guidance.

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TABLE I. The ENSIP task.

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANALYSIS MATERIAL CHARACTERISTICS & DEVELOPMENT TESTS	COMPONENT & CORE ENGINE TESTS	GROUND & FLIGHT ENGINE TESTS	ENGINE LIFE MANAGEMENT
DEVELOPMENT PLANS • ENSIP Master Plan • Durability & Damage Tolerance Control Plans • Materials Process Characteristics Plan • Corrosion Prevention & Control • Inspect & Diagnostics Plan OPERATIONAL REQUIREMENTS • Design Service Life & Design Usage • Design Criteria	• Design Duty Cycle • Material Characterization • Design Dev Tests • Analyses – Sensitivity – Critical Parts List – Thermal – Strength – Containment – Vibration/Flutter – Stress/Environment Spectra – Durability – Damage Tolerance – Creep • Installed Engine Inspectability • Manufacturing, Process, & Quality Controls – VSR – NDI Demo	COMPONENT TESTS • Strength • Vibration • Durability • Damage Tolerance • Containment CORE ENGINE TESTS • Thermal Survey • Vibration Strain and Flutter Boundary Survey	GROUND ENGINE TESTS • Thermal Survey • Ground Vibration Strain & Flutter Boundary • Unbalanced Rotor Vibration • Strength • Impedance • Clearance • Containment • Ingestion • Accelerated Mission Tests (AMT) • Damage Tolerance FLIGHT ENGINE TEST • Fan Strain Survey • Nacelle Temp Survey • Installed Vibration • Deterioration	• Updated Analyses • Engine Structural Maintenance Plan • Operational Usage Survey • Individual Engine Tracking • Durability & Damage Tolerance Control Actions (Production) • Retirement-for-Cause Actions/Plan

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A.5.2 Turbine engine structural integrity verification program.

The ENSIP Master Plan will be used to define and document the specific verification tasks.

VERIFICATION RATIONALE (A.5.2)

Past experience with airframe and engine development programs has demonstrated the usefulness in using plans to show the approach to be used to conduct structural development. As a result, the ENSIP uses such plans in several instances as specified herein.

An ENSIP Master Plan is used to define in detail the supplemental information needed in conjunction with this handbook to write a contractual document and to integrate the various analysis and test tasks. Adequacy of the tasks proposed for structural design, development, qualification, and life management of a specific engine system will be evaluated by review of the ENSIP Master Plan.

VERIFICATION GUIDANCE (A.5.2)

The ENSIP Master Plan should follow the format of A.4 and A.5 of this document. Supplemental information required to make the master plan a contractual document should be developed using guidelines contained in this handbook and through reviews with the Procuring Activity. The plan should contain the time-phased scheduling and integration of all required ENSIP tasks for design, development, qualification, and life management. The schedules for ENSIP tasks should be integrated with the full-scale development and production decision milestones. The ENSIP task milestones are contained in table II relative to the four-step full-scale development of Initial Flight Release (IFR), Full Flight Release (FFR), Initial Service Release (ISR) and Operational Capability Release (OCR). The plan should include discussion of unique features, identification of exceptions to the guidelines and requirements of this handbook and the associated rationale, and any problems anticipated with execution of the plan. The plan and schedules should be kept current.

VERIFICATION LESSONS LEARNED (A.5.2)

Refer to A.4.2 Requirement Guidance.

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TABLE II. ENSIP/4 milestone development schedule.

	RFP	CONTRACT AWARD	IFR	FFR	ISR	OCR
DESIGN INFORMATION						
• ENSIP Master Plan (5.2)		X PLUS PERIODIC UPDATES				
• Design Service Life and Design Usage (5.3)	X					
• Design Duty Cycle (5.3)		X				
• AMT Spectrum (5.9.1.2)		X				
• Material Characterization Plan (5.6)		X				
• Design Criteria (4.4 through 4.15)	X	X				
• Damage Tolerance and Durability Control Plans (5.8 and 5.9)		X PLUS PERIODIC UPDATES				
• Corrosion Prevention and Control Plan (5.9.3)		X PLUS PERIODIC UPDATES				
ANALYSES*						
• Thermal (5.5.2)			X			
• Damage Tolerance (5.8)			X			
• Durability (5.9)			X			
• Creep/Stress Rupture (5.12)			X			
• Strength (5.10)			X			
• Dynamic (5.13.1)			X			
• Containment (5.10.3)			X			

NOTE:
**Periodic updates are required per the Contract Data Requirements List to incorporate results of tests and usage surveys.*

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TABLE II. ENSIP/4 milestone development schedule - Continued.

	RFP	CONTRACT AWARD	IFR	FFR	ISR	OCR
COMPONENT TESTS						
• Damage Tolerance (5.8)				X		
• NDI Demonstration (5.8.2)					X	
• Intrinsic Material Defect Distribution (5.8.2)					X	
• Durability (5.9)				X		
• Strength (5.10)			X			
• Creep/Stress Rupture (5.12)			X			
• Containment (5.10.3)			X			
CORE ENGINE TESTS						
• Thermal (5.5.2)			X----------	----------	----------X	
• Strength (5.10)			X			
• Aeromechanical (5.9)			X----------	----------	----------X	
• Containment (5.10.3)			X----------	----------	----------X	
ENGINE LIFE MANAGEMENT						
• Interpretation and Evaluation of Test Results (5.9.1.6)			X	X	X	X
• Structural Maintenance Plan (5.19.a)					X----------	----------X
• Engine Usage Recording System (5.19.b)		X----------	----------	----------	----------X	
• Individual Engine Tracking System (5.19.c)		X----------	----------	----------	----------X	
• Component Tracking System (5.19.d)		X----------	----------	----------	----------X	

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TABLE II. ENSIP/4 milestone development schedule - Continued.

	RFP	CONTRACT AWARD	IFR	FFR	ISR	OCR
FULL-SCALE ENGINE TESTS						
• Thermal (5.5.2)			X----------	----------	----------X	
• AMT (5.9.1.1)			X			
(5.9.1.2)				X	X	
(5.9.1.3)						X
(5.9.1.4)						
• Strength (5.10)			X			
• Mechanical Impedance (5.12.b)			X			
• Aeromechanical (5.12.c)			X----------	----------	----------X	
• Noise (5.14)			X----------	----------	----------X	
• Containment (5.10.3)			X----------	----------	----------X	
• FOD/DOD (5.15)			X----------	----------	----------X	
• Engine/Airframe Compatibility (5.18)			X----------	----------X		

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A.4.3 Design service life.

The engine will have a design service life of at least __(a)__ when subjected to the design usage of 4.4. In addition, the engine will be able to withstand __(b)__ hours at any point in the envelope for both hot and cold parts.

REQUIREMENT RATIONALE (A.4.3)

The service life should be established since it is one of the primary design goals.

REQUIREMENT GUIDANCE (A.4.3)

The following should be used to tailor the handbook paragraph:

(a): The design service life should be determined by the Using Service and based on the Mission Need Statement. The units (e.g.; cycles, mission hours, flight hours, etc.) for design service life should be determined by the Using Service. The information in table III should be used as a guide to determine the design service life.

(b): a value of ten (10).

TABLE III. Guide to determine design service life.

		SERVICE LIFE			
SYSTEM CATEGORY	PARTS	FLIGHT (HOURS)	GROUND RUN (HOURS)	FLIGHT (MISSIONS)	GROUND RUNS (MISSIONS)
Fighter/Attack	Cold Parts	4,000	400	3,000	200
	Hot Parts	2,000	200	1,500	100
Bomber	Cold Parts	10,000	1,000	2,500	200
	Hot Parts	4,000	500	1,250	100
Cargo	Cold Parts	30,000	3,000	9,000	1,000
	Hot Parts	15,000	1,500	4,500	500
Trainer	Cold Parts	18,000	5,400	13,500	1,500
	Hot Parts	9,000	2,700	6,750	750
Helicopter	Cold Parts	6,000	400	3,000	750
	Hot Parts	6,000	400	3,000	750

REQUIREMENT LESSONS LEARNED (A.4.3)

Structural life requirements are the most difficult primary design goals to fulfill. This was the case on many legacy engine development programs due to a lack of adequate usage parameter definitions. Although requirements for mission profiles, mission mix, and flight hours have often been defined accurately, the important usage parameters that govern cyclic life (major throttle cycles other than the start-stop excursion, time at or above Intermediate power, dwell times, etc.) have not been accurately defined. As a result, operational data has revealed usage parameters not accounted for in design and has resulted in significant reduction in life limits for critical parts and the associated need for redesign and spare parts. Therefore, it is important that realistic design usage information be identified at the outset of the development program for use in design, analysis, and test. The information in table IV presents design service life requirements used in the past.

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TABLE IV. Past design service life requirements.

		SERVICE LIFE			
SYSTEM CATEGORY	PARTS	FLIGHT (HOURS)	GROUND RUN (HOURS)	FLIGHT (MISSIONS)	GROUND RUN (MISSIONS)
Fighter F-22 (F119)	Cold Parts	4,000	1,350	2,938	35
	Hot Parts	2,000	675	1,469	18
Bomber B-2 (F118)	Cold Parts	10,000	N/A	*Note 1*	*Note 1*
	Hot Parts	4,000	TBD	*Note 1*	*Note 1*
Cargo C-17 (F117)	Cold Parts	30,000	TBD	8,516	TBD
	Hot Parts	15,000	TBD	4,258	TBD
Trainer T-1A (JT15D-5)	Cold Parts	9,000	12,600	2,760	TBD
	Hot Parts	4,500	6,300	1,380	TBD
Turboshaft T800	Cold Parts	6,000	N/A	N/A	N/A
	Hot Parts	6,000	N/A	N/A	N/A

NOTE:
[1] Information is classified.

A.5.3 Design service life.

The requirements of 4.3 will be evaluated by analysis, inspection, demonstration, and test.

VERIFICATION RATIONALE (A.5.3)

Design service life requirements should be evaluated to ensure the desired levels of damage tolerance, durability, functional capability, operability, performance, reliability, and strength are attained.

VERIFICATION GUIDANCE (A.5.3)

Evaluation is generally accomplished by analysis and test. The process of conducting an evaluation program is the responsibility of the contractor and should be an integral part of the tailored integrity program.

VERIFICATION LESSONS LEARNED (A.5.3)

None.

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A.4.3.1 Hot parts.

Hot parts will have a usable life of __(a)__ times the design service life specified in 4.3. Hot parts and their lives will be listed in table V.

TABLE V. Hot parts.

REQUIREMENT RATIONALE (A.4.3.1)

Hot parts life is specified to achieve logistic and economic effectiveness.

REQUIREMENT GUIDANCE (A.4.3.1)

The following should be used to tailor the specification paragraph:

(a): a value between one-half and one times ($^1/_2$ X and 1 X).

A tabular listing of hot parts and their lives should be provided by the engine manufacturer. Hot parts include all parts exposed to the hot gas stream such as the combustor liner, turbine blades and vanes, and exhaust nozzle.

Helicopters designed to operate in environments with minimum maintenance facilities available should require longer hot parts lives for readiness purposes, and be equal to cold parts life.

REQUIREMENT LESSONS LEARNED (A.4.3.1)

In the past, time to cracking did not necessarily become the life limit for a hot part; therefore, the design of hot section parts has been based on evaluation of wear, LCF, creep, stress rupture, oxidation/erosion, and sulphidation.

Cooled turbine vanes were designed such that cooling air would outflow in the presence of cracking or other distress that extends through the thickness. Dual-compartment positive outflow design has been needed to minimize erosion rates subsequent to the occurrence of thermal mechanical fatigue cracks, thereby maximizing total usable life of the airfoil.

Turbine vanes have also been designed for positive retention so that vane segments would not fall into the gas flow path and cause secondary damage subsequent to total burn-through or severance of an airfoil.

A.5.3.1 Hot parts.

The requirement of 4.3.1 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.3.1)

Hot parts life is evaluated to ensure compliance with the requirement of A.4.3.1.

VERIFICATION GUIDANCE (A.5.3.1)

A sensitivity analysis should be conducted (on selected hot parts) to identify the effect on parts lives which result from a range of usage parameters (above and below the design points).

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Failure modes (e.g.; LCF, creep, stress rupture, etc.) analyses should be conducted by the contractor to establish design stress levels and lives for engine hot parts based on the design usage.

Usage parameters to be considered in the sensitivity analysis should include airspeed, altitude, ambient temperature, partial throttle cycles, and dwell times at minimum and maximum power levels.

Evaluation of hot part lives should be attained as part of the required mission endurance testing. Evaluation of hot parts lives should also be accomplished via the other evaluations conducted in A.4.8 through A.4.15. Pass/fail criteria (i.e., allowable post-test part condition) should be established for all hot parts life testing. Pass/fail criteria for hot parts life testing should be quantified through definition of the post-test condition in terms of dimensional tolerances and wear limits.

VERIFICATION LESSONS LEARNED (A.5.3.1)

Improper definition of allowable post-test condition of hot parts has been a shortfall in most engine development programs.

A.4.3.2 Cold parts.

Cold parts will have a usable life of __(a)__ times the design service life specified in 4.3. Cold parts and their lives will be listed in table VI.

TABLE VI. Cold parts.

REQUIREMENT RATIONALE (A.4.3.2)

Cold parts life is specified to achieve logistic and economic effectiveness.

REQUIREMENT GUIDANCE (A.4.3.2)

The following should be used to tailor the specification paragraph:

(a): a value of at least one, minimum.

A tabular listing of cold parts and their lives is to be provided by the engine manufacturer.

Parts not listed as hot parts in A.4.3.1 are considered to be cold parts.

High cycle fatigue problems affect those parts subjected to aero-induced and vibratory loading. All rotating parts that contact static and/or other rotating parts are susceptible to wear. Although creep is a phenomenon typically associated with hot parts, cold and hot section disks have been known to creep due to high centrifugal stresses and the thermal environment.

REQUIREMENT LESSONS LEARNED (A.4.3.2)

Cold parts have been typically designed to an LCF requirement that ensured cracking would not occur before the required durability limit was reached. The incidence of LCF failures has been

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reduced after many years of emphasis on designing against LCF. However, other failure modes have become increasingly bothersome (e.g.; HCF, wear, and creep).

A.5.3.2 Cold parts.

The requirement of 4.3.2 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.3.2)

Cold parts lives are evaluated to ensure compliance with the requirement.

VERIFICATION GUIDANCE (A.5.3.2)

A sensitivity analysis will be conducted (on selected cold parts) to identify the effect on parts lives which results from a range of usage parameters (above and below the design points).

Failure modes (e.g.; LCF, HCF, creep, etc.) analyses should be conducted by the contractor to establish design stress levels and lives for engine cold parts based on the design usage.

Usage parameters to be considered in the sensitivity analysis should include airspeed, altitude, ambient temperature, partial throttle cycles, and dwell times at minimum and maximum power levels.

Evaluation of cold part lives can be attained as part of the required mission endurance testing. Evaluation of cold parts lives should also be accomplished via the other evaluations conducted in A.4.8 through A.4.15. Pass/fail criteria (i.e., allowable post-test part condition) should be established for all cold parts life testing. Pass/fail criteria for cold parts life testing should be quantified through definition of the post-test condition in terms of dimensional tolerances and wear limits.

VERIFICATION LESSONS LEARNED (A.5.3.2)

Improper definition of allowable post-test condition of cold parts has been a shortfall in most engine development programs.

A.4.3.3 Expendables.

The minimum life without replacement of all expendable parts and components will be equal to the minimum maintenance-free operating period. Expendable parts, components, and their lives will be listed in table VII.

TABLE VII. Expendable parts.

REQUIREMENT RATIONALE (A.4.3.3)

It is necessary to specify the minimum life of expendable parts and components since their failure or degradation will affect life cycle cost, maintainability, and functional readiness of the engine and subsystems.

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REQUIREMENT GUIDANCE (A.4.3.3)

The contractor should provide a tabular listing of expendables with their respective functional lives.

Expendable parts are those normally replaced at maintenance or overhaul, such as minor hardware, O-rings, and gaskets. Expendable components include starters and ignitors.

REQUIREMENT LESSONS LEARNED (A.4.3.3)

None.

A.5.3.3 Expendables.

The requirement of 4.3.3 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.3.3)

Functional life of expendables are evaluated to insure practical and economical maintenance intervals.

VERIFICATION GUIDANCE (A.5.3.3)

Evaluation of expendable parts and components lives can be attained as part of the durability test program. Expendables will typically be replaced during an AMT.

Analyses are not always a practical means to evaluate the required lives of expendables.

VERIFICATION LESSONS LEARNED (A.5.3.3)

None.

A.4.3.4 Bearings.

The mainshaft and gearbox bearings will have B1.0 lives equal to at least the design service life of the engine. A list of bearings and their lives will be presented in table VIII.

TABLE VIII. Bearing lives.

BEARING	TYPE (ROLLER OR BALL)	LIFE (HOURS)

REQUIREMENT RATIONALE (A.4.3.4)

Main shaft and engine gearbox bearings life are specified to ensure the bearings meet engine durability requirements.

REQUIREMENT GUIDANCE (A.4.3.4)

The contractor should provide a tabular listing of the bearings with their respective lives.

Special attention to life testing should be taken when specifying high DN (diameter x rpm) (2.5×10^6) or unusual mounting configuration bearings; e.g., outer race rotating bearings or in shaft bearings in which both races rotate.

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REQUIREMENT LESSONS LEARNED (A.4.3.4)

Predictions of bearing performance and/or life have not always been reliable, especially those bearings whose operating conditions exceed recent engine experience or whose designs are unusual. For example, bearing performance analytical models and limited life tests did not have the capability to predict or reveal roller dynamic instability which occurred in some high DN engine roller bearings. More extensive life testing may have uncovered this potentially catastrophic bearing failure mode.

A.5.3.4 Bearings.

The requirement of 4.3.4 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.3.4)

Analyses and tests are required to determine the lives of the engine bearings.

VERIFICATION GUIDANCE (A.5.3.4)

A bearing life analysis and bearing tests should be conducted.

The bearing life presentation should clearly identify whether the Weibull slope in the analysis is assumed or whether it is the slope acquired from rig testing. Evaluation should be accomplished early enough in the program to allow for redesign and requalification tests and to make the necessary inputs into the spares provisioning programs.

It is critical that analytical and empirical evaluation of rotor thrust balance occur in IFR and FFR milestones. Rotor thrust balance reports should show agreement between analytical and test data. Rotor thrust evaluation tests should be conducted on appropriate configurations. The intent of the test and analysis is twofold: (1) to ensure peak thrust loads are consistent with thrust bearing life requirements, and (2) to ensure rotor thrust crossovers occur only transiently and in the absence of significant radial loads. The latter is needed to preclude skidding damage to ball bearings.

VERIFICATION LESSONS LEARNED (A.5.3.4)

None.

A.4.3.5 Components.

Engine components will have a usable life of _(a)_ times the design service life specified in 4.3. Engine components and their lives will be listed in table IX.

TABLE IX. Components.

REQUIREMENT RATIONALE (A.4.3.5)

Engine components life is specified to achieve logistic and economic effectiveness.

REQUIREMENT GUIDANCE (A.4.3.5)

The value inserted should be:

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(a): a value of one, minimum.

A tabular listing of engine components and their lives is to be provided by the engine manufacturer. Engine components include: fuel pumps, engine controls, jet nozzle and actuators, anti-icing valves, and the temperature sensing system.

REQUIREMENT LESSONS LEARNED (A.4.3.5)

U.S. Air Force engine-related mishap data from 1976-1987 shows a high incidence of failure/malfunction of engine components and externals. Engine component and external failures/malfunctions accounted for no less than 35 percent of the total engine-related mishaps during that period.

A.5.3.5 Components.

The requirement of 4.3.5 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.3.5)

Engine components lives are evaluated to ensure compliance with the requirement.

VERIFICATION GUIDANCE (A.5.3.5)

Failure modes analyses should be conducted by the contractor to establish design lives and stress levels when subject to the design usage. Evaluation of engine component lives can be attained as part of the required mission endurance testing. Evaluation of engine component lives should also be accomplished via other evaluations in this specification.

VERIFICATION LESSONS LEARNED (A.5.3.5)

Insufficient qualification testing (methods and duration) has resulted in the fielding of engine components that were not able to meet the desired operational life.

A.4.4 Design usage.

The engine structure will be able to withstand the design usage specified herein for the design service life specified in 4.3. The design service life and design usage will be specified in terms of mission profiles and mission mix, including nonoperating transport of the engine. Important usage parameters will be specified. The flight envelope, mission profiles, mission mix, and environment will be shown.

REQUIREMENT RATIONALE (A.4.4)

One of the major shortcomings in legacy USAF airframe and engine development programs has been inadequate definition of the operational usage parameters critical to the durability of engine components. Although requirements for mission profiles, mission mix, and flight hours have often been defined accurately, the important usage parameters that govern cyclic life (major throttle cycles other than the start-stop excursion, time at or above intermediate power, dwell times, etc.) have not been accurately defined as part of design information on many systems. As a result, operational data has revealed usage parameters not accounted for in design and has resulted in significant reduction in life limits for critical parts and the associated need for redesign and spare parts. Therefore, it is important that realistic design usage information be identified at the outset of full-scale development for use in design, analysis, and test of the engine.

REQUIREMENT GUIDANCE (A.4.4)

The design service life and design usage will be supplied by the Procuring Activity as part of the request for proposal. The contractors should identify for the Procuring Activity's consideration

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any recommended changes based on their experience. It is recommended the contractors conduct trade studies to establish cost (Life Cycle Cost [LCC], weight, performance, etc.) as a function of structural life (inspection intervals, economic life, etc.). The results of these trade studies should be presented to the Procuring Activity for consideration to establish a preferred engine design service life.

If specific design usage requirements are not specified by the Procuring Activity, the contractor will convert the airframe mission profile information supplied by the Procuring Activity to engine usage profiles, as required (i.e., to convert airplane thrust requirements for profile segments into engine power settings). The design service life and design usage should be included as part of the contract specifications (Prime Item Development Specification and the ENSIP Master Plan).

a. The design usage should include:

(1) Missions and Mission Mix

(2) Usage parameters

(3) Externally-applied forces

(4) Operating envelope

(5) Engine attitude limits

(6) Ambient temperature extremes

(7) Icing environment conditions

(8) Corrosive atmosphere conditions

(9) Noise environment conditions

(10) Customer bleed air extraction, loaded accessory pads, and power take-off usage

(11) Engine performance retention characteristics.

Table X contains engine usage parameters critical to structural design as a function of aircraft type. These data should be used as guidance in early design efforts (i.e., advanced engine programs and preliminary design).

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TABLE X. Guidance data for early design efforts.

System Category	Parts Classification	Type I Cycles	Type III Cycles	Type IV Cycles	Augmentor Lights or Thrust Reversing	Augment Time or Vector Cycles (hrs)	Time at IRP and Above (hrs)
		0-Max-0	Idle-Max-Idle	Cruise-Int-Cruise			
Fighter	Cold Parts	3,200	20,000	24,000	17,000	200	800
	Hot Parts	1,600	10,000	12,000	8,500	100	400
Bomber	Cold Parts	2,700	30,000	30,000	16,000	250	1,80
	Hot Parts	1,350	15,000	15,000	8,000	15	90
Cargo	Cold Parts	10,000	14,000	TBD	N/A	N/A	6,300
	Hot Parts	5,000	7,000	TBD	N/A	N/A	3,000
Trainer	Cold Parts	15,000	150K	150K	TBD	TBD	3,600
	Hot Parts	7,500	75,000	75,000	TBD	TBD	1,800
Helicopter	Cold Parts	15,000	N/A	N/A	0	0	N/A
	Hot Parts	15,000	N/A	N/A	0	0	N/A

Typical total flight hours as a function of aircraft type are listed in table X. Total flight hours includes all the time spent at power from the onset of takeoff roll to touchdown at landing. It is the intent that the total flight hours specified for the engine system be consistent with (equal to) that specified for the airframe weapon system. If a total flight-hour requirement is identified in the Statement of Need (SON) by the Using Command, this requirement will be contained in the table.

Total operating time (TOT) includes mission time from engine start through taxi, engine flight time as defined above, and taxi after landing to engine shutdown. Past data shows that ground operation during a mission is approximately 0.75-1.00 hours and can be added to engine flight hours per mission to derive TOT per mission.

Typical ground run hours as a function of aircraft type are listed in table X. Engine ground run hours pertain to time spent on the ground running for functional checks such as trim checks and system equipment checks. Past data indicates these ground run hours (excluding taxi time) can be approximately 5 to10 percent of total flight hours. Surveys and reviews of past engine usage should be taken periodically to establish ground run time for specific aircraft weapon systems.

Typical number of flight and ground runs as a function of aircraft type are listed in table X. The number of runs is a derived value dependent on the total flight and ground run hours and the length of each type of flight mission and the ground run. The length of each mission type can

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vary significantly depending on the aircraft type. For example, a fighter aircraft mission flight time can be 1.0 to1.5 hours duration while a bomber aircraft mission flight time can be 4.0 hours duration. The number of flight and ground runs should be explicitly stated since it establishes the number of 0-immediate/max-0 throttle excursions.

Typical values for the number and type of throttle excursions as a function of aircraft type are listed in table X. Throttle excursions principally drive the low cycle fatigue failure mode and dictate component crack growth lives and inspection periods in a turbine engine and need to be taken into account in design. Therefore, the structural design of a turbine engine should consider all transients which will produce fatigue damage and/or crack growth from the selected initial flaw. As a minimum, the 0-intermediate/max-0 cycles, idle-intermediate/max-idle cycles and cruise-intermediate/max-cruise cycles will be taken into account. In some cases, it may be necessary to include other types of cycles, such as throttle reburst cycles. A throttle reburst particularly occurs in a system which experiences air-to-air combat or air-to-ground usage. This type of cycle occurs when an idle dwell follows a period of sustained maximum power and the idle dwell is sufficient in duration to achieve thermal reversal on the components (usually a turbine disk and attached blade retainers). Thermal reversals are also possible in the compressor after shutdown and during refueling after high altitude, high mach number operation. A throttle reburst at this point will add mechanical stress to the already present thermal stress. Numerous throttle activities of this type can significantly affect life and should be taken into account in design, where appropriate. Hold times at idle power after a sustained period at intermediate/maximum power need to be defined and used in design on new engine development programs. Recorded data is now available or becoming available on operational usage of several weapon systems (F-15, F-16, B-1 Flight Test, etc.). These data have been used to establish distribution of dwell times as a function of aircraft type and mission type. It is the intent that these distributions and results for other usage parameters be placed in this appendix as soon as possible.

Typical values of "hot time" as a function of aircraft type are listed in table X. Aircraft drag, gross weight, and mission altitude requirements affect climb time, which is usually accomplished at maximum or intermediate power settings and therefore are large drivers in the structural design of engine components subject to creep and stress rupture failure modes. It is suggested that a conservative approach be taken for design purposes in deriving time at or above intermediate power since all past systems historically have had higher drag and gross weight than originally predicted. Further, in certain cases, system flight tactics may evolve which will require maximum power for extended purposes not foreseen in the construction of early mission profiles. Time at high mach number need to be thoroughly investigated since this flight condition usually accelerates creep and stress rupture of some engine components. It is suggested a sensitivity analysis be accomplished on components critical in creep and stress rupture to all those variables mentioned above. Although time at or above intermediate rated power (IRP) is a derived value dependent on the mission profiles, mission mix, and specific duty cycles, it should be stated as an explicit value since it drives creep and stress rupture life for many engine components. It is recommended that time at or above intermediate power not be less than 20 percent of the total flight hour requirement in table X.

Typical values for number of augmentor lights and time spent in augmentation as a function of aircraft type are listed in table X. Careful analysis of mission profiles and mix should be accomplished to determine augmentor usage in terms of number of lights and time spent in augmentation. The number of augmentor lights will affect the thermal low cycle fatigue life of the augmentor liner as well as erosion capability. It is recommended margins be provided in design for more severe usage of the augmentor than indicated in the mission profiles. Many engine programs in the past have had augmentor durability problems which required extensive field repair. It is recommended the time spent in augmentation be no less than 5 percent of the

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total flight hour requirement in table X. It is also recommended the number of augmentor lights be no less than 50 percent of the total number of throttle excursions contained in table X.

Although time spent at key points in the flight envelope is a derived value dependent on the mission profiles, mission mix, and specific duty cycles, it should be stated as an explicit design requirement since it drives creep, stress rupture, and flutter for many engine components. Also, specific times at flight envelope extremities should be stated as an explicit design requirement in addition to values derived from the duty cycles to ensure engine capability to meet future usage requirements of the airframe weapon system.

High mach number/altitude design requirements (flight envelope points and duration) should be established between the Procuring Activity and the Using Command for a particular engine and aircraft weapon system, and these requirements should be contained in A.4.4.

The design usage should include, but not be limited to, the defined missions and mission mix, design duty cycle, usage parameters, nozzle usage, environmental (external, internal, and installation) conditions, unique flight conditions, and the non-operating environment. The internal environment is specified in A.4.5.2. The engine operating envelope is determined by the engine contractor. The missions and mission mix of A.4.4 are presented in table XI. Unique flight conditions are discussed in A.4.5 through A.4.5.1 and presented on figure 1. Vibration and dynamic response characteristics are specified in A.4.13 through A.4.13.1. Design usage should also include the external environmental conditions specified in A.4.5, A.4.5.1, and A.4.15, which cover atmospheric conditions and engine ingestion capability such as bird, ice, water, steam, sand, and dust.

The typical rate of performance deterioration should be based on the performance program and performance deterioration model. The contractor should address deteriorated engine conditions as part of the design practice and account for it in the life predictions.

Engines with control systems that maintain minimum thrust levels, by increasing engine temperature and speed, will decrease the potential parts lives by exposing the engine to increased thermal and mechanical stresses.

The ability of engine hot parts to meet design life requirements can be significantly reduced due to engine uptrim or other conditions that result in hot gas stream temperatures higher than that of a production engine. To account for the impact on hot parts life, by operation at increased temperatures, margins of 30°F to 70°F above production acceptance (non-degraded) maximum steady-state gas temperature have been imposed by the Procuring Activity during the design of hot parts to ensure design life goals will be met. Aerospace Industries Association (AIA) PC Project 338-2A members made a consensus recommendation in 1982, that the Procuring Activity not establish a specific temperature margin since this number will vary with engine type and application. For analysis purposes, the F100-PW-229 was designed for $^{1}/_{3}$-life at nominal production performance and $^{2}/_{3}$-life at full deterioration levels to provide full life, even with deteriorated engines.

REQUIREMENT LESSONS LEARNED (A.4.4)

One of the major shortcomings in past engine development programs has been inadequate definition of the design usage parameters critical to engine durability. Although requirements for mission profiles, mission mix, and flight hours have often been defined accurately, the important usage parameters that govern cyclic life (major throttle cycles other than the start-stop excursion, time at or above Intermediate power, dwell times, etc.) have not been accurately defined. As a result, operational data has revealed usage parameters not accounted for in design and has resulted in significant reduction in life limits for critical parts and the associated need for redesign and spare parts. Therefore, it is important that realistic design usage

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information be identified at the outset of the development program for use in design, analysis, and test.

TABLE XI. Design duty cycle summary.

COLD PARTS									
	Time (hrs)	TAIAA (hrs)	Type I Cycles	Type III Cycles	Type IV Cycles	A/B lts	A/B Time (hrs)	Vector Cycles	Other Cycles
Flight Operations									
Ground Operations									
Test Cell Trouble-shooting, etc.									
TOTAL									

TAC = Type I + Type III/4 + Type IV/40 + Kx (other cycles)

HOT PARTS									
	Time (hrs)	TAIAA (hrs)	Type I Cycles	Type III Cycles	Type IV Cycles	A/B lts	A/B Time (hrs)	Vector Cycles	Other Cycles
Flight Operations									
Ground Operations									
Test Cell Trouble-shooting, etc.									
TOTAL									

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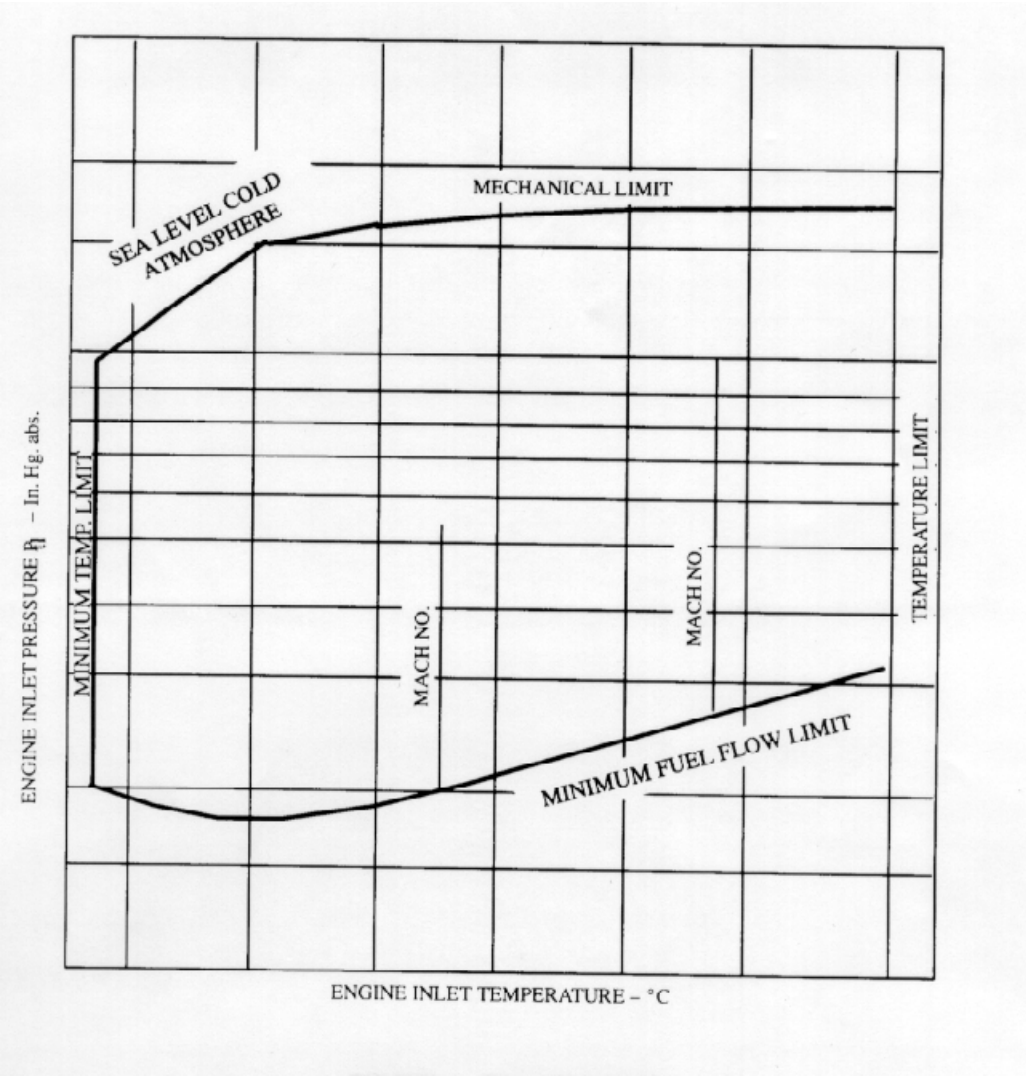

FIGURE 1. Operating limits.

Past program-specific usage parameters are specified in table XII.

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TABLE XII. Past program-specific usage parameters.

System Category	Parts Classification	Type I Cycles	Type III Cycles	Type IV Cycles	Augmentor Lights or Thrust Reversing	Augment Time or Vector Cycles (hrs)	Time at IRP and Above (hrs)
		0-Max-0	Idle-Max-Idle	Cruise-Int-Cruise			
Fighter F-22 (F119)	Cold Parts Hot Parts	2,973 1,487	20,503 10,252	22,074 11,037	20,239 10,165	186 (109) 93 (54)	684 342
Bomber B-2 (F118)	Cold Parts Hot Parts	2,371 948	7,113 2,844	N/A N/A	N/A N/A	N/A N/A	TBD TBD
Cargo C-17 (F117)	Cold Parts Hot Parts	8,516 4,258	25,840 12,920	17,178 8,589	12,700 6,350	N/A N/A	TBD TBD
Trainer T-1A JT15D-5	Cold Parts Hot Parts	28,000 14,000	20,288 10,140	3,817 1,909	N/A N/A	N/A N/A	885 443
Helicopter	Cold Parts Hot Parts	15,000 15,000	TBD TBD	TBD TBD	0 0	0 0	TBD TBD

A.5.4 Design usage.

Verification of design usage will be accomplished by analysis, design development tests, and engine tests, in accordance with the ENSIP Master Plan to ensure that the engine and its components meet the design service life and design usage requirements of 4.3 and 4.4. A design duty cycle(s) will be derived from the design service life and design usage specified in 4.3 and 4.4. The design duty cycle will be supplied.

VERIFICATION RATIONALE (A.5.4)

Usage requirements of A.4.4 will entail multiple mission profiles—each with separate speed, altitude, and throttle excursions as a function of time. It is not practical to analyze and test each of these discrete profiles with appropriate mission mix throughout the various development tasks. Therefore, a minimum number of design duty cycles need to be derived early in the development phase (as early as contract award) for use in all subsequent analysis and test tasks. These design duty cycles may be provided by the Procuring Activity as part of the RFP.

VERIFICATION GUIDANCE (A.5.4)

Sensitivity analysis will be conducted on selected components to identify the effect of probable ranges in usage variables on engine life limits. The results of the sensitivity analysis will be used to condense the design service life and design usage of A.4.4 into a minimum number of design duty cycles. Important parameters to be considered in the sensitivity analysis include airspeed, altitude, partial throttle cycles (cruise to intermediate, idle to cruise, etc.), and dwell

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time at min and max power levels. The Procuring Activity will identify in the RFP the applicable requirement for sensitivity analysis.

VERIFICATION LESSONS LEARNED (A.5.4)

See A.4.4 Lessons Learned.

A.4.5 Operating envelope.

The engine will meet all the requirements of the document throughout the complete operating envelope without exceeding any limits. The engine operating limits will be specified for the identified environment and displayed with figures 1 and 2, and tables XIII and XIV. If applicable, the thrust augmentation operating envelope will be included on the figures.

REQUIREMENT RATIONALE (A.4.5)

This requirement defines the operating envelope wherein the engine meets its functional, performance, and durability requirements (aerothermodynamic and mechanical limitations). The engine air mass flow inlet conditions in terms of pressure and temperature will be different and more severe than sea-level static standard day values during much of its operation. The engine needs to be able to operate in these expected environments and component durability cannot be degraded such that the design life requirements are not attained.

REQUIREMENT GUIDANCE (A.4.5)

If the referenced figures are insufficient to describe the operating envelope, either the Using Service or contractor can add a table.

The specified limits should be predicated on the most critical parameters and characteristics of the engine. The absolute altitude of the engine and the range of Mach numbers applicable at standard day, cold, tropical, and hot atmospheric conditions should be specified on a figure. A figure should represent the operating envelope for pressure/temperature. It is recommended that aerodynamic/thermodynamic limits (total pressure-P versus total temperature-T), the flight envelope (altitude versus flight mach number), and the ambient temperature distribution be diplayed on figures and be included as a design requirement. Specific values for these curves should be selected to be consistent with the intended application and the primary specification.

Both transient and steady-state operations should be specified. The engine operating envelopes should meet or exceed the envelope requirements of all current and anticipated aircraft applications for the engine.

Design requirements are set by envelope extremes, such as the maximum inlet pressure, which helps define the combustor case maximum pressure requirements and blade aero-elastic requirements for blade vibration. Low inlet pressure and temperatures impose design requirements on the combustor and augmentor performance parameters. Altitude and Mach number extremes will impact the cooling requirements of the engine lubrication system. Bearing loads need to be quantified throughout the engine envelope. In summary, most parts of the engine are, in some way, impacted by the engine envelope requirements.

REQUIREMENT LESSONS LEARNED (A.4.5)

In-service engines have operated outside the operating envelope. The aircraft envelope is normally within the engine envelope.

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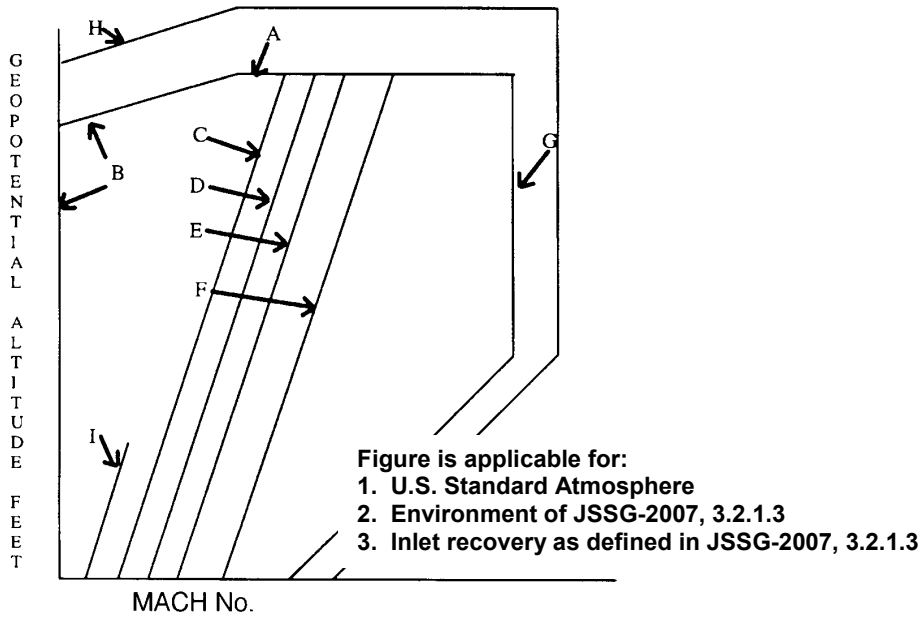

NOTES:

A. *Maximum starting altitude*
B. *Minimum Mach No. for starter-assisted starting*
C. *Minimum Mach No. starting without starter assist, no customer power extraction, no customer bleed air extraction*
D. *Minimum Mach No. for starting without starter assist, Maximum customer power extraction, no customer bleed air extraction*
E. *Minimum Mach No. for starting without starter assist, no customer power extraction, Maximum customer bleed air extraction*
F. *Minimum Mach No. for starting without starter assist, Maximum customer power extraction. Maximum customer bleed air extraction.*
G. *Maximum Mach No. for starting air extraction*
H. *Operating envelope*
I. *Post-loiter starting point*

FIGURE 2. Operating envelope.

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TABLE XIIIa. Performance of standard day, 15°C (59°F), sea level, static conditions[1] with _____ nozzle.

Power Setting[2]	Minimum Thrust N (lbf)		Maximum SFC (lbm/hr/lbf)		Maximum Engine Rotor Speed (rpm)		Maximum Gas Temp[3] (°C [°F])		Maximum Measured Temp[4] (°C [°F])		Total Airflow[5] ± ___% (lbm/sec)	
	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]
Max. Aug. (if applicable)												
Min. Aug. (if applicable)												
Int.												
90% Int.												
75% Int.												
Idle	(Max)	(Max)	*Note 6*	*Note 6*	(Min)	(Min)						

NOTES:
[1] *The engine performance values shown will be from the engine performance computer program of JSSG-2007, 3.2.1.1*
[2] *Power settings and columns will be added or deleted as required by the Using Service (e.g., customer bleed air/power extraction).*
[3] *Defined at the first stage turbine rotor inlet location (Contractor will parenthetically insert, in column heading, the station designation).*
[4] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[5] *Total engine airflow*
[6] *Maximum fuel consumption - lbm/hr*
[7] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

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TABLE XIIIb. Performance at non-standard hot ambient, sea level, static conditions[1] with _____ nozzle.

Power Setting[2]	Minimum Thrust N (lbf)		Maximum SFC (lbm/hr/lbf)		Maximum Engine Rotor Speed (rpm)		Maximum Gas Temp[3] (°C [°F])		Maximum Measured Temp[4] (°C [°F])		Total Airflow[5] ± ___% (lbm/sec)	
	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]
Max. Aug. (if applicable)												
Min. Aug. (if applicable)												
Int.												
90% Int.												
75% Int.												
Idle	(Max)	(Max)	*Note 6*	*Note 6*	(Min)	(Min)						

NOTES:
[1] *The engine performance values shown will be from the engine performance computer program of JSSG-2007, 3.2.1.1*
[2] *Power settings and columns will be added or deleted as required by the Using Service (e.g., customer bleed air/power extraction).*
[3] *Defined at the first-stage turbine rotor inlet location (Contractor will parenthetically insert, in column heading, the station designation).*
[4] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[5] *Total engine airflow*
[6] *Maximum fuel consumption - lbm/hr*
[7] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

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TABLE XIIIc. Performance at non-standard cold ambient, sea level, static conditions[1] with _____ nozzle.

Power Setting[2]	Minimum Thrust N (lbf)		Maximum SFC (lbm/hr/lbf)		Maximum Engine Rotor Speed (rpm)		Maximum Gas Temp[3] (°C [°F])		Maximum Measured Temp[4] (°C [°F])		Total Airflow[5] ± ___% (lbm/sec)	
	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]
Max. Aug. (if applicable)												
Min. Aug. (if applicable)												
Int.												
90% Int.												
75% Int.												
Idle	(Max)	(Max)	*Note 6*	*Note 6*	(Min)	(Min)						

NOTES:
[1] *The engine performance values shown will be from the engine performance computer program of JSSG-2007, 3.2.1.1*
[2] *Power settings and columns will be added or deleted as required by the Using Service (e.g., customer bleed air/power extraction).*
[3] *Defined at the first-stage turbine rotor inlet location (Contractor will parenthetically insert, in column heading, the station designation).*
[4] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[5] *Total engine airflow*
[6] *Maximum fuel consumption - lbm/hr*
[7] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

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TABLE XIVa. Turboprop/turboshaft engine performance at standard day, 15°C (59°F), sea level, static conditions[6] with _____ nozzle.

Power Setting[1]	Max. Gas Generator Speed(s) (rpm)		Max. Gas Temp.[2] (°C [°F])		Max. Measured Temp.[3] (°C[°F])		Total Airflow[4] ± ___% (lb/sec)		Max. SFC (lbm/hr/kw)		Min. Shaft Power kw (shp)		Output Torque N-m (lb-ft)		Min. Output Shaft Speed (rpm)		Resid. Jet Thrust N (lbf)	
	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]
Contingency (if applicable)																		
Maximum (if applicable)																		
Intermediate																		
Maximum Continuous																		
90% Maximum Continuous																		
No Load (if applicable)																		
Idle	Min	Min							*Note 5*	*Note 5*	Max	Max						

NOTES:
[1] *Parameters and columns will be added or deleted as required by the Using Service (e.g., customer bleed air).*
[2] *Defined at the first-stage high pressure turbine rotor inlet location (Contractor will parenthetically insert, in column heading, the station designation.)*
[3] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[4] *Total engine airflow*
[5] *Maximum fuel consumption - lb/hr*
[6] *The engine performance values shown will be from the engine performance computer program of JSSG-2007, 3.2.1.1*
[7] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

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TABLE XIVb. Turboprop/turboshaft engine performance at non-standard hot ambient sea level, static conditions[6] with _____ nozzle.

Power Setting[1]	Max. Gas Generator Speed(s) (rpm)		Max. Gas Temp.[2] (°C [°F])		Max. Measured Temp.[3] (°C [°F])		Total Airflow[4] ± ___% (lb/sec)		Max. SFC (lbm/hr/kw)		Min. Shaft Power kw (shp)		Output Torque N-m (lb-ft)		Min. Output Shaft Speed (rpm)		Resid. Jet Thrust N (lbf)	
	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]
Contingency (if applicable)																		
Maximum (if applicable)																		
Intermediate																		
Maximum Continuous																		
90% Maximum Continuous																		
No Load (if applicable)																		
Idle	Min	Min							*Note 5*	*Note 5*	Max	Max						

NOTES:
[1] *Parameters and columns will be added or deleted as required by the Using Service (e.g., customer bleed air).*
[2] *Defined at the first-stage high pressure turbine rotor inlet location (Contractor will parenthetically insert, in column heading, the station designation.)*
[3] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[4] *Total engine airflow*
[5] *Maximum fuel consumption - lb/hr*
[6] *The engine performance values shown will be from the engine performance computer program of JSSG-2007, 3.2.1.1*
[7] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

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TABLE XIVc. Turboprop/turboshaft engine performance at non-standard cold ambient sea level, static conditions[6] with _____ nozzle.

Power Setting[1]	Max. Gas Generator Speed(s) (rpm)		Max. Gas Temp.[2] (°C [°F])		Max. Measured Temp.[3] (°C [°F])		Total Airflow[4] ± ___% (lb/sec)		Max. SFC (lbm/hr/kw)		Min. Shaft Power kw (shp)		Output Torque N-m (lb-ft)		Min. Output Shaft Speed (rpm)		Resid. Jet Thrust N (lbf)	
	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]	New	Det.[7]
Contingency (if applicable)																		
Maximum (if applicable)																		
Intermediate																		
Maximum Continuous																		
90% Maximum Continuous																		
No Load (if applicable)																		
Idle	Min	Min							*Note 5*	*Note 5*	Max	Max						

NOTES:
[1] *Parameters and columns will be added or deleted as required by the Using Service (e.g., customer bleed air).*
[2] *Defined at the first-stage high pressure turbine rotor inlet location (Contractor will parenthetically insert, in column heading, the station designation.)*
[3] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[4] *Total engine airflow*
[5] *Maximum fuel consumption - lb/hr*
[6] *The engine performance values shown will be from the engine performance computer program of JSSG-2007, 3.2.1.1*
[7] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

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TABLE XV. Performance at altitude conditions[4] with _____ nozzle.

Power Setting[1]	Alt. (ft)	Amb. Temp.	Mach Num.	Minimum Net Thrust N (lbf)		Maximum SFC (lb/hr/lb)		Maximum Engine Rotor Speed (rpm)		Maximum Gas Temp[2] (°C [°F])		Maximum Measured Temp[3] (°C [°F])		Total Airflow (lb/sec) ± __%		Cust. Bld. Air	Cust. Pwr. Ext.
				New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]		

NOTES:
[1] *Power settings will be as required by the Using Service to cover the operating envelope and to be compatible with mission requirements.*
[2] *Defined at the first-stage turbine rotor inlet location (Contractor will parenthetically insert, in column heading, the station designation.)*
[3] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[4] *The engine performance values shown will be from the engine computer program of JSSG-2007, 3.2.1.1.*
[5] *Deteriorated engine performance is specified for _____ TACs (or hours) or use.*

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A.5.5 Operating envelope.

The requirements of 4.5 will be evaluated by analysis, demonstration, and test.

VERIFICATION RATIONALE (A.5.5)

Engine testing in ground test facilities is required to demonstrate satisfactory engine operation and performance throughout the operating envelope of the engine.

VERIFICATION GUIDANCE (A.5.5)

Engine testing should be conducted to demonstrate the engine can meet the functional, performance, and durability requirements of this specification throughout the operating envelope. All the sea level and altitude tests specified should be used to demonstrate satisfactory operation throughout the operating envelope. Test demonstration figures should show both the operating envelope and demonstration points. The Using Service and contractor should negotiate the test points, and the testing should be accomplished in an altitude test facility.

Characteristics of the engine which should be evaluated around the envelope include: steady-state and transient performance, engine stability, starting, internal stresses and temperatures, augmentor operation, lubrication system operation, control system operation, and inlet distortion tolerance.

VERIFICATION LESSONS LEARNED (A.5.5)

Thorough testing of the engine around the envelope extremes in an altitude test facility has been a cost-effective method to reduce the possibility of encountering engine problems during flight testing and deployment of the weapon system. Early identification of engine problems and demonstration of corrective changes around the engine envelope has been accomplished with this testing. Experience has shown that augmentors on afterburning engines should be tested for stable combustion in the upper-lefthand-corner of the envelope as well as at maximum Mach number conditions. Maximum inlet pressure conditions on the exhaust nozzles have caused buckling of nozzle components. This test is, therefore, of particular importance to engines with variable geometry exhaust nozzles. There have been cases where oil hiding has been observed in the engine gearbox at high inlet pressure. Therefore, the behavior of lubrication and fuel systems should be carefully monitored at flight envelope extremes and during altitude starts.

A.4.5.1 Operating attitude and conditions.

The engine operating attitude limits will be shown on figure 3. The engine will meet the requirements of the specification when operated in the normal operation area of the figure, and operate at least __(a)__ seconds continuously in the limited and transient operation areas of figure 3. Operation in the limited operation area will not degrade engine performance or cause any damage. The engine will start, stop, and be stowed in any of the attitudes shown in the normal operation area of figure 3. Engine stowing capability outside the limited operation area will be specified. The engine will function satisfactorily for at least __(b)__ seconds in negative g and for at least __(c)__ seconds in zero g conditions.

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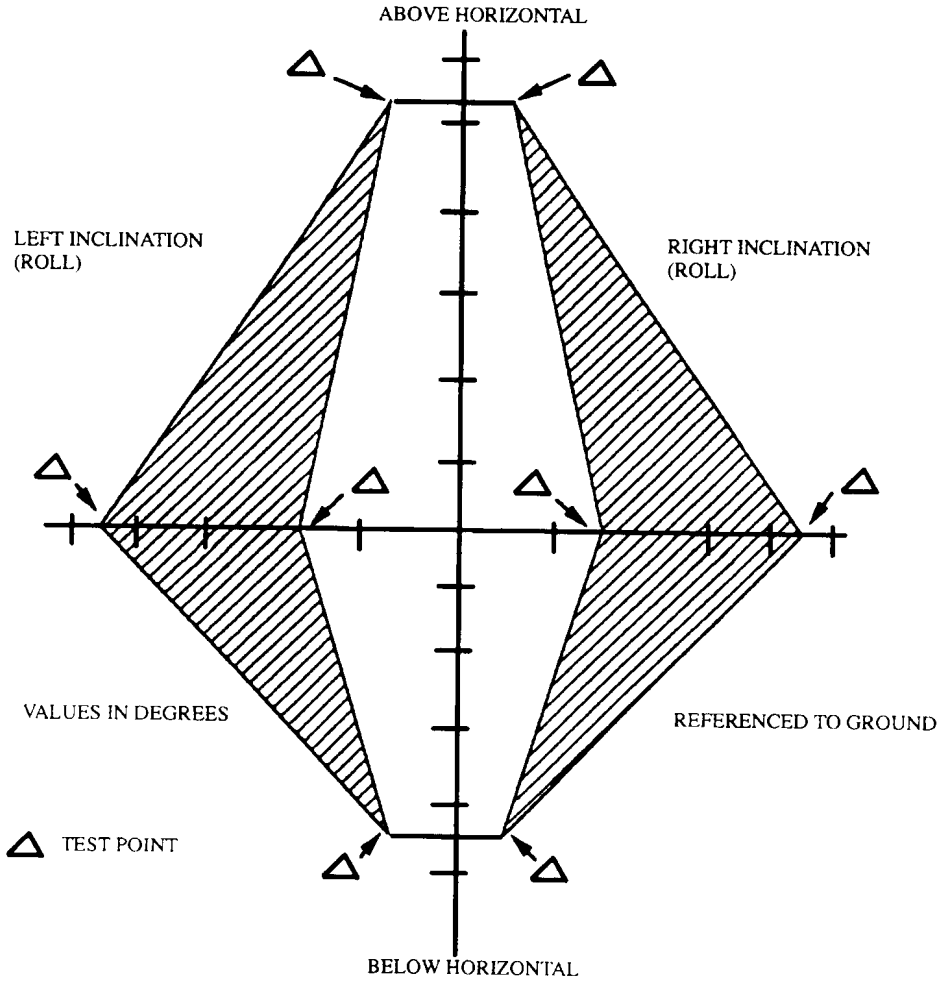

NOTES:

1. *The engine will be able to operate at all possible acceleration conditions. However, the figure assumes no acceleration other than gravity to define the direction of acceleration vector from the engine CG.*
2. *The engine centerline is perpendicular to the plane of the paper.*
3. *Continuous operation in clear area*
4. *_____-second operation in shaded area*

FIGURE 3a. Engine attitude limits (fixed wing aircraft).

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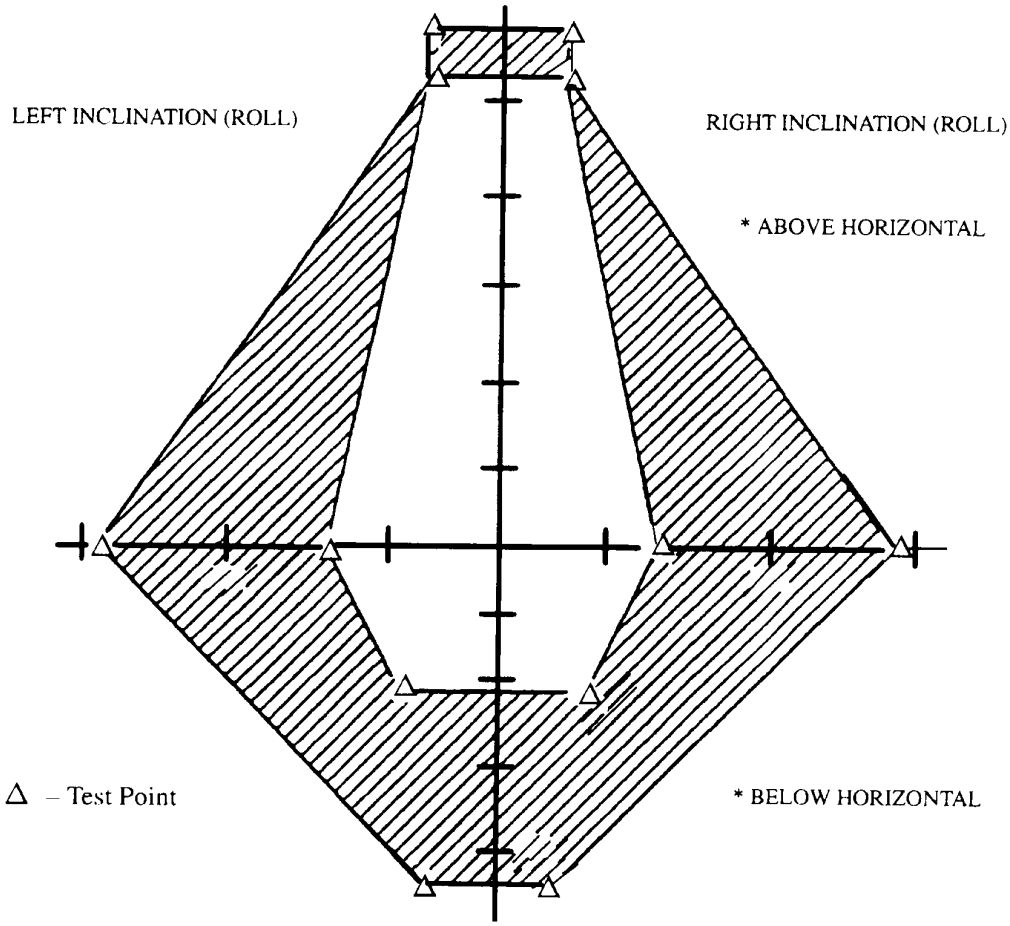

NOTES:
1. *The engine will be able to operate at all possible acceleration conditions. However, the figure assumes no acceleration other than gravity to define the direction of acceleration vector from the engine CG.*
2. ** _____ referenced to ground.*
3. *Engine centerline is perpendicular to the plane of the paper.*
4. *Continuous operation in clear area*
5. *_____-second operation in shaded area*

FIGURE 3b. Engine attitude limits (VSTOL aircraft engines).

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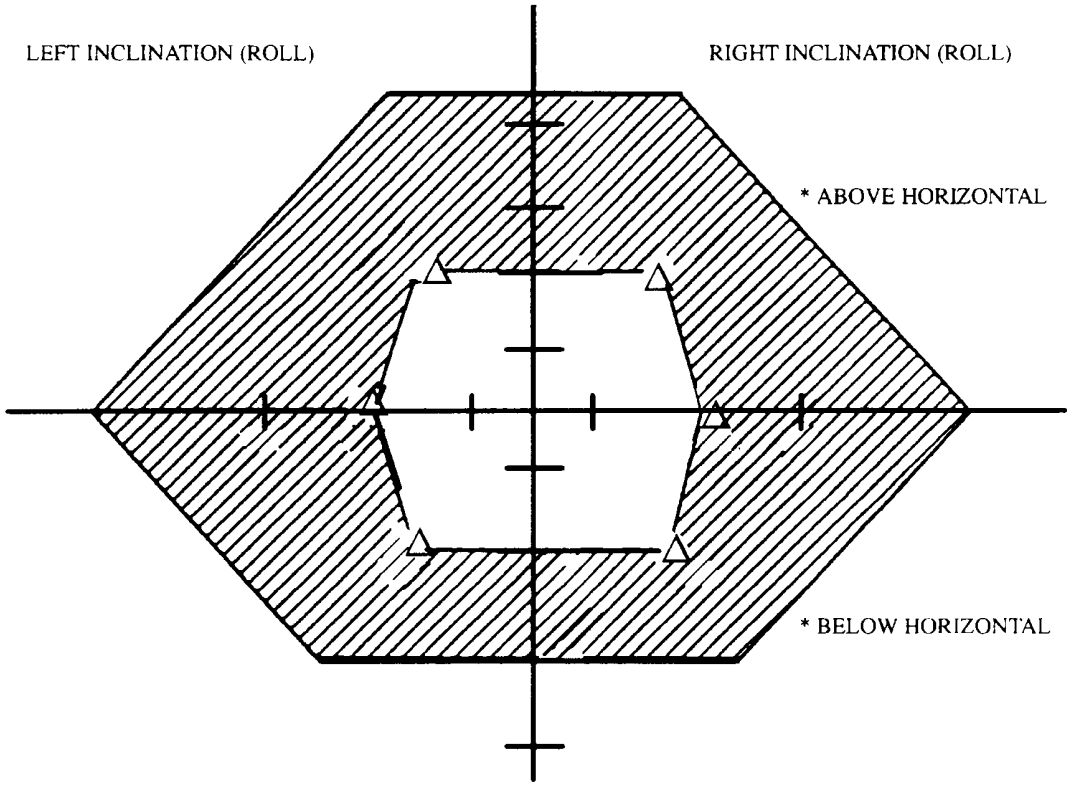

NOTES:

1. *The engine will be able to operate at all possible acceleration conditions. However, the figure assumes no acceleration other than gravity to define the direction of acceleration vector from the engine CG.*
2. ** _____ referenced to ground.*
3. *Engine centerline is perpendicular to the plane of the paper.*
4. *Continuous operation in clear area*
5. *_____-second operation in shaded area*
6. *The symbol, " Δ " indicates points for test.*

FIGURE 3c. Engine attitude limits (rotary-wing aircraft engines).

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REQUIREMENT RATIONALE (A.4.5.1)

The engine is required to operate throughout the attitudes imposed by the aircraft, such as takeoff, climb, inverted flight, air combat maneuvers, stowage, and terrain following. Time duration of these factors is a significant consideration to the engine design.

REQUIREMENT GUIDANCE (A.4.5.1)

The following should be used to tailor the specification paragraph:

(a): a value of at least 30 seconds,

(b): a value of at least 60 seconds, and

(c): a value of at least 30 seconds.

Background:

The starting and stopping attitude limits for Army V/STOL aircraft which require engine attitude changes should be not less than 105 degrees nose up, 20 degrees nose down, and 30 degrees roll to each side. Above 6 km, continuous operation is not required at nose-up attitudes greater than 45 degrees.

REQUIREMENT LESSONS LEARNED (A.4.5.1)

Extended flight operation in zero or negative "g" conditions resulted in oil system malfunctions, such as oil starvation, bearing sump flooding, gearbox flooding, oil foaming problems, and seal leaks. Special provisions, such as an auxiliary lube system, may be necessary in the lubrication system for extended inverted flight operation.

A.5.5.1 Operating attitude and conditions.

The requirements of 4.5.1 will be evaluated by analysis, demonstration, and test.

VERIFICATION RATIONALE (A.5.5.1)

Testing is required to demonstrate satisfactory engine functional capability under a variety of attitude and maneuvering conditions.

VERIFICATION GUIDANCE (A.5.5.1)

The engine should be subjected to an attitude test to meet the requirements of A.4.5.1. Engine capability to operate for 60 seconds at negative "g" and 30 seconds at zero "g" conditions should be evaluated by analysis or by a rig test of the lubrication oil system. The engine should be started and operated at intermediate thrust/power demand position, for at least 30 minutes, at each of the test points shown in the normal operation area of figures 3a, 3b, and 3c. The engine should also be operated at intermediate thrust/power demand position for at least 30 seconds at each of the test points shown in the limited operation area of figures 3a, 3b, and 3c. This test should be considered completed when the engine starts satisfactorily, remains within all operating limits, and there is no evidence of mechanical damage.

The qualification test program should have a strong foundation of component tests, especially in the oil system component area. Oil tanks, oil pumps, scavenge pumps, gearboxes, deareators, and bearing compartments should all be component-tested in various simulated flight conditions. It may be possible to test complete small engines at various attitudes.

Figures 3a, 3b, and 3c should define engine attitude limits (roll and pitch) under static ground test stand conditions and should not be construed as necessarily defining engine attitude limits during flight (engine attitude capability is usually limited by lubrication system design). The forces generated during maneuvering flight may have a combined or resultant effect that

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permits engine pitch and roll attitudes that exceed the limits of figures 3a, 3b, and 3c without encountering an engine-limiting condition.

VERIFICATION LESSONS LEARNED (A.5.5.1)

None.

A.4.5.2 Internal environment.

The engine components will be able to withstand the internal thermal and pressure environments that occur during engine operation (steady-state and transient conditions).

REQUIREMENT RATIONALE (A.4.5.2)

Thermal and pressure conditions change significantly throughout the engine rotor stages and need to be accounted for in design and analysis to ensure life requirements are attained.

REQUIREMENT GUIDANCE (A.4.5.2)

The internal environment of the engine should be characterized for both steady-state and transient conditions for each critical point in the flight envelope. Conditions to be considered should include shutdown and cool-down effects, stratification affecting rotor bow and subsequent starts as well as installed engine outer case temperature and temperature variations, thermal, pressure, vibration, and dynamic loading. The internal environment should be specified at various radial and axial locations, as necessary, to characterize engine operating conditions completely. The probable variations in radial profiles and pattern factors between combustor systems which may occur due to fabrication and assembly tolerances will be established. Transient conditions will be evaluated to identify critical thermal stresses that can occur during acceleration, deceleration, dwell, and shutdown.

REQUIREMENT LESSONS LEARNED (A.4.5.2)

Inadequate characterization of the internal environment has been a shortcoming of many engine programs.

A.5.5.2 Internal environment.

Evaluation of the capability of the engine components to withstand the internal thermal and pressure environments that occur during engine operation will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.5.2)

Analysis and tests should be performed early to establish component internal temperature distributions and interstage pressure distributions to support initial design of structural components. Recorded data during engine operation is required to establish confidence in the thermal heat transfer model and predicted pressure distributions.

VERIFICATION GUIDANCE (A.5.5.2)

A thermal heat transfer model should be generated for the engine static and rotating structures. The model should have the capability to predict component nodal temperatures as a function of flight conditions via the engine performance deck. Particular care should be taken in analysis of the internal engine aerodynamics and establishment of the convective boundary conditions (recovery temperatures and film coefficient distributions) for the internal structures. Radiation effects should be accounted for, where appropriate (i.e., the combustor). The conduction model should have sufficient detail to establish critical temperatures and gradients for steady-state and

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transient conditions. The thermal heat transfer model should be correlated with data obtained from the instrumented engine tests required by A.4.5.2.

Instrumented engine tests will be performed for both steady-state and transient conditions to measure internal gas stream, cooling flow cavities, and metal temperatures; pressure distributions; external temperatures for the installed configuration; and temperatures and pressures at other engine locations, as required. The contractor should schedule internal environment thermal and pressure surveys as early as possible in the development phase. Both core and full-scale engines should be utilized, as appropriate. Engine run conditions should include: (1) stabilized idle to snap/acceleration to stabilized maximum power to chop/deceleration to stabilized idle, (2) shutdown and cool-down, and (3) the planned AMT power sequence. Thermocouple and pressure measurements should be made throughout the engine modules. Cavity pressures should be measured. Thermocouples should be located at radial locations on disks and on critical seals/spacers to establish gradient data. The scope and plan for the thermal survey program should be identified in the appendix to this specification or in the ENSIP Master Plan.

VERIFICATION LESSONS LEARNED (A.5.5.2)

None.

A.4.5.3 Externally-applied forces.

The engine will function satisfactorily and no deformation will occur during or after exposure to the externally-applied forces, which should be indicated in design load diagrams.

REQUIREMENT RATIONALE (A.4.5.3)

The engine and its components in service use are subject to externally-applied forces due to accelerations, decelerations, angular velocities, external airloads, and gyroscopic moments which result from operation and maneuver of the aircraft. These forces have an impact on design life.

REQUIREMENT GUIDANCE (A.4.5.3)

Externally-applied forces include: loads produced by take-off, landing, in-flight maneuvers, gusts, vibration, installation, and crash conditions. The limit loads will be based on a weight factor that consists of the dry weight of the engine increased by the specified weight allowed for all engine-mounted components and operating fluids. In installations where airframe components are supported by the engine, the weight of these components will also be included in the weight factor.

Load factors specified will be sufficient to meet all ground, flight, and landing operations for the installed engine in the intended aircraft application. Loads due to flexure of the mounts, vibration, "g" factors, engine airloads, or crash conditions will be considered in the establishment of engine strength and life requirements. Aircraft used on carriers are exposed to significant "g" loads which result from catapult launch or carrier arrest landings. The "g" forces presented on the design load diagrams may be too severe for some given applications. A judgment needs to be made whether to reduce the requirements for a particular, less severe application. The diagrams should consider the extremes for all manned aircraft.

Typical aircraft load factor spectra applicable to cargo, fighter/attack, and trainer classes of aircraft are contained in ASD-TR-82-5012. Applicable aircraft vertical load factor spectra, in conjunction with the pitch and yaw velocities of one radian per second, should be used to establish repeated loads for engine design.

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REQUIREMENTS LESSONS LEARNED (A.4.5.3)

Aircraft such as the A-4, A-6, A-7, F-4, F-6, and F-8 have recorded load factors up to 7g's; and F-16s go up to 9g's. Aircraft maneuver requirements have increased so that particular weapon systems need now be capable of 8 to 10 g's.

The F404-GE-400 engine model specification CP45K006 of 15 November 1975 used a maneuver load spectrum which showed various values of positive and negative "g's" and cycles per 1000 hours for each "g" value. The maximum static load requirement was reduced to 1.4 times (1.4 X) the landing loads rather than the 1.5 times (1.5 X) the requirement.

A.5.5.3 Externally-applied forces.

Verification of flight and ground externally-applied forces will be in accordance with 4.5.3, and will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.5.3)

There is a need to evaluate the engine by analysis and test its capability to withstand the external forces to which it may be subjected due to flight maneuvers, landings, and takeoffs.

VERIFICATION GUIDANCE (A.5.5.3)

Stress and deflection data should be obtained at critical locations as determined by analysis and preliminary stress coating tests. Engine cases and mounts should be subjected to a static rig test. The static rig test, utilizing the applicable engine static structure, should be conducted to demonstrate the capability of the engine and its supports to withstand maximum externally-applied forces specified in A.4.5.3 without permanent deformation of any component and 1.5 times (1.5 X) those forces without failure of any component. The loads should be applied separately and in combination.

VERIFICATION LESSONS LEARNED (A.5.5.3)

None.

A.4.6 Material characterization.

The materials used in the engine should have adequate structural properties, such as strength, creep, low-cycle fatigue, high-cycle fatigue, fracture toughness, crack growth rate, stress corrosion cracking, thermomechanical fatigue, oxidation/erosion, wear, ductility, elongation, and corrosion resistance; so that component design can meet the operational requirements for the design service life and design usage of the engine specified in 4.3 and 4.4.

REQUIREMENT RATIONALE (A.4.6)

Material structural properties should be quantified in advance of detail design so that materials selection and design operating stress levels can be established which provide a high degree of confidence that operational requirements will be met. Early generation of sufficient data for use in preliminary and detail design is emphasized since later surprises relative to structural properties will have a significant impact on redesign, substantiation and replacement needs, and weapon system availability.

REQUIREMENT GUIDANCE (A.4.6)

Structural properties used in design (design allowables) should be based on minimum material capability unless otherwise stated in this document. The intent is to base all material properties except fracture toughness and crack growth on minus three Sigma (-3σ) values with a 50 percent confidence level or minus two Sigma (-2σ) values with a 95 percent confidence level.

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Another option is to state that material properties will be based on B0.1 probability values. The confidence level for B0.1 is 50 percent. Another alternative is "A Basis" from MIL-HDBK-5, which uses properties for 99 percent exceedance with 95 percent confidence. Typically, B50 properties may be used to characterize fracture toughness and crack growth rate. In addition, design allowables should be justified by the contractor's experience base design methodology, and design criteria. Specimens fabricated from "as produced" parts should be tested to verify properties relative to different locations within the part (i.e.; locations that receive different amounts of work during manufacture such as the bore, web, and rim regions of disks). If "as produced" parts are unavailable, the use of parts produced by equivalent practices, or parts sufficiently similar, should be considered, if available.

Single Crystal Materials

Single crystal materials used in the engine should have adequate structural properties such as strength, creep, time-dependent behavior at the usage temperature, low cycle fatigue, high cycle fatigue, fracture toughness, crack growth rate, stress corrosion cracking, thermomechanical fatigue, oxidation/erosion, wear, ductility elongation, and corrosion resistance so that component design can meet the operational requirement for the design service life and design usage of the engine specified. Unlike traditional isotropic, polycrystalline material, single crystals require design practices that account for their behavior on each orientation of interest. Specifically, these practices should account for both material and load orientation effects. When considering these effects, it is important to account for possible mis-orientation of the primary and secondary axes due to typical manufacturing procedures. When the orientation is not closely controlled, the direction of either axis may be off up to 10° from nominal and the secondary axis orientation may not even be known. These manufacturing deficiencies can cause significant debits in fatigue properties, creep behavior, and crack growth rates. Therefore, the orientation of the primary and secondary axes should be controlled as closely as is practical. Proof of the component design's ability to handle the manufacturing variance of the production process is necessary. In addition to off-axis orientation, mixed-mode crack behavior tends to be much more problematic in single crystals than in isotropic materials. Because of this, analysis of threshold properties should account for mixed-mode loading over the predicted operating temperature range of the engine. The inclusion of temperature range is due to the fact that crack growth planes and morphology may change significantly at lower temperatures. A lack of well-accepted procedures necessitates provision for mixed-mode crack growth in instances where single-mode crack growth does not describe the crack behavior accurately and the mixed-mode stresses surpass the mixed-mode threshold value.

High Cycle Fatigue

The material properties should be established at stresses (steady and vibratory), frequencies, temperatures, and other parameters representative of the operating environment of the engine. Loading conditions for which stresses in materials are established for high cycle fatigue should be determined from stress and vibration analysis based on a probabilistic formulation of static and dynamic forcing functions. The probability of failure due to these forcing functions should be maintained below 1×10^{-7} per EFH on a per-stage basis, provided the system-level safety requirements are met. Material allowables for high cycle fatigue should be based on basic building block specimen and sub-element laboratory tests, and validated against sub-element and component laboratory bench tests. In the establishment of these allowables, the methodology for transferability of laboratory data to components should be identified. The statistical basis and significance of material allowables should be defined. In the establishment of material allowables, consideration should be given to the combination of applied vibratory stress levels, mean stresses, multiaxial stress state, vibratory frequency, maximum number of

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applied cycles (see section A.4.13.3), and state of material damage. Material damage should include, but not be limited to, variation in initial material quality due to manufacturing, fabrication, or inherent material defect population; in-service damage such as that produced by low cycle fatigue loading, fretting or fretting fatigue, wear, or foreign object damage; any other anticipated damage including, but not limited to, corrosion or other environmentally-induced degradation of material capability, thermal or thermal-mechanical cycling, static or cyclic creep, and break-in or green run engine cycles.

Material damage states which should be addressed in the design process include all conditions which are considered by design, analysis, or field experience to limit the durability of the material or component or produce conditions which require either periodic inspection or replacement at set intervals. The damage accumulated immediately prior to the inspection or periodic maintenance interval should be considered with respect to its effect on the high cycle fatigue behavior of the material or component. Allowable high cycle fatigue vibratory loading should be less than that which would cause such damage to worsen or propagate due to high cycle fatigue loading.

In addition to the above damage states, any combinations of these which are deemed likely to occur during the lifetime of the component and may produce a degradation of the high cycle fatigue capability of the material should be considered in the establishment of the material design allowables.

REQUIREMENT LESSONS LEARNED (A.4.6)

Premature structural failures have occurred prior to design service life (based on average material properties) and have been attributable to components with material capabilities as low as minimum, unforeseen vibratory stresses or damage modes, or errors in analysis.

A.5.6 Material characterization.

Material structural properties should be established by test and modeling. Anticipated properties under damage states (e.g.; fretting, etc.) should be verified through combinations of laboratory specimen, sub-element and component testing, material damage models which have been validated against databases and supplemented with historical data which cover the range of potential damage states, or databases which cover the properties under damage states.

VERIFICATION RATIONALE (A.5.6)

Material properties should be established by test and should be based on specimens fabricated from “as produced” parts, from parts produced by equivalent practices, or from parts sufficiently similar in processing and size, since critical structural properties are dependent upon the manufacturing processes. Damage states in the parts which may occur during field usage should be verified for their potential impact on high cycle fatigue life.

VERIFICATION GUIDANCE (A.5.6)

General

A material characterization plan should be prepared and existing data should be presented. Final definition of structural capability should be based on the testing of specimens fabricated from “as produced” parts, from parts produced by equivalent practices, or from parts sufficiently similar in processing and size. The contractor should review existing data on proposed materials and processes and develop a material characterization plan that identifies and schedules each of the tasks and interfaces in design, material selection, and testing. The tasks to be identified in the plan should include:

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a. Correlation of the operating envelope conditions to which each material will be subjected (i.e.; temperature, loading frequency, max and min cyclic stresses, steady and vibratory stresses, etc.) through the test environment and usage;

b. a parts listing with the corresponding materials and manufacturing processes;

c. identification of mechanical properties that should be generated for each material/part;

d. test specimen configuration;

e. the source of material data;

f. number of tests to be conducted for each material property curve needed for each part;

g. quality control actions or vendor substantiation test requirements that will be utilized to ensure at least minimum mechanical properties will be attained in finished parts through the production run; and

h. risk assessment and abatement plan for use of any advanced materials and processes.

Existing data obtained through earlier tests can be used during initial design only when the manufacturing processes are similar (i.e.; same methods of producing billets, forgings, heat-treat processes, machining, surface treatment, etc.). Final definition of structural capability should be based on the material property curves generated by testing specimens fabricated from the "as produced" parts, from parts produced by equivalent practices, or from parts sufficiently similar in processing and size to verify material properties relative to different locations on the part, as appropriate, based on screening tests or historical data. Material properties should be defined for each material/part source (i.e., material and manufacturing vendor).

The number of tests conducted for each curve or condition should be adequate to establish minimum material properties used in design or to establish the correlation between the data obtained from specimens cut from parts and the database within the calibrated design methodology.

High cycle fatigue

Material allowables should be based on the combination of mean and vibratory stress amplitude, or equivalent quantities for multiaxial stress conditions. Material allowables for a uniaxial stress state, presented in the form of a constant life Haigh diagram (commonly referred to as, "a Goodman diagram"), should be based on actual data. Straight line extrapolations from fully reversed (R = –1) loading data to the quasi-static yield or ultimate stress are *not* acceptable for engine full production release status. Further, for high values of mean stress greater than one-half the static yield stress, maximum stress (= mean stress + alternating stress) should be used as the material allowable in addition to vibratory stress to establish a factor of safety. For materials that exhibit time-dependent deformation (e.g., titanium at room temperature), strain accumulation should be considered in the establishment of material allowables. Time-dependent deformation should include effects due to: (a) static creep—deformation accumulated on the basis of time spent at stress levels above which static creep can occur, and (b) cyclic creep or creep ratcheting—deformation accumulated on a cycle by cycle basis due to hysteresis in the stress-strain response of the material. Cyclic-dependent creep or creep ratcheting should be considered when plasticity effects are more pronounced, specifically for larger stress ratios, R, above 0.7. At high stress ratios, the use of K_{max} as well as ΔK should be considered for damage tolerance.

Material capability in the presence of a damage state as described in section A.4.6 should be verified. The following damage states should be verified as follows:

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Low cycle fatigue

Cracks which might form due to low cycle fatigue loading and grow to an inspectable size should be considered in high cycle fatigue analysis. It should be demonstrated that the largest crack which might be present just before inspection should not propagate due to high cycle fatigue loading if present, based on a threshold stress intensity applicable to the specific crack size considered. For cracking below the inspection limit, it should be shown that high cycle fatigue would not lead to failure within twice the inspection interval.

In regions of contact (i.e., blade to disk), it should be assumed that a crack of depth $2a_0$, normal to the contact surface, can develop during service. It should be demonstrated that such a crack will not grow to a catastrophic size during a time corresponding to twice the inspection interval in regions where inspections are performed to detect cracks of depth $2a_0$, or for twice the design service life for regions where such an inspection cannot be (is not) performed. Crack detection is defined as the ability to find a crack with a reliability of 50 percent and a confidence limit of 90 percent. When the growth rate is established, the bulk stresses in the component—including both LCF and vibratory stresses—should be considered, in addition to contributions from the local stress field due to the contact loads. Where vibratory stresses are present, the threshold stress intensity should be used to insure HCF propagation will not occur (see HCF section, above) unless it can be shown that crack arrest will occur after further crack growth.

Note: a_0 is determined from the El Haddad formulation (see A.2.3) which relates threshold ΔK to $\sigma_{endurance}$ and ensures growth behavior is outside the "short crack" regime, so anomalous short crack behavior does not have to be considered.

$$a_0 = \frac{1}{\pi}\left[\frac{\Delta K_{th}}{F\Delta\sigma_{end}}\right]^2$$

where ΔK_{th} is the threshold stress intensity range, $\Delta\sigma_{end}$ is the endurance limit stress range, and F is the geometry factor for the specific crack being considered. For a through edge crack, F = 1.12, while for a thumbnail crack in a smooth bar, F = 1.12(2/π) = 0.713. In general, F is defined from the stress intensity solution for any crack as

$$K = \sigma F \sqrt{\pi a} \quad .$$

Fretting fatigue

When used to describe wear of two contacting surfaces in a turbine engine, fretting fatigue is typically a misnomer. The scientific definition of fretting fatigue puts limits on the amount of relative displacement (typically no more than 100 μm). Unfortunately, the type of damage usually termed “fretting fatigue” in a turbine engine experiences significantly more relative displacement than 100 μm. The more accurate term for most contact damage in an engine would be “contact fatigue.” However, for the sake of consistency with previous ENSIP editions, we will use the term “fretting fatigue” to describe all possible wear modes.

a. Design of contacting parts, specifically the blade/dovetail attachment, needs to account for the high local stresses and spatial stress distribution near the edge of contact. Standard finite element analyses do not resolve these stresses accurately or quickly enough to be a part of the design process. Industry needs to incorporate additional methods, such as numerical solutions to the equation presented by Hills and Nowell [1] that is used to describe the contact-induced stress state from a random indenter, to account for the edge of contact stress concentration. Finite element analyses may be sufficient to determine damage propagation stresses at a depth below

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that affected by the edge of contact (typically 100μm). As a general design guide, the contact stresses due to the geometry and vibratory modes of the component should be designed to be as small as possible, while other component properties, such as Low Cycle Fatigue life, are maintained. For example, changing the length of the dovetail post may change the contact stresses of a dovetail slot on a bladed disk. However, this change will also change any number of component responses at other locations, such as the stress concentration at the dovetail fillet or the modal response of the bladed disk. In this example, the post length should be designed such that contact stresses are within the design allowable and it can be shown that failure in either the blade or disk will not occur in normal operation.

b. In the case of contacting parts that experience changes in contact force (normal pressure) that is dependent on engine operating condition, the design process needs to account for load interactions due to the mission cycle. Also, coefficient of friction (averaged over the contact surface) evolution over the mission cycle needs to be accounted for in stress analysis. The coefficient of friction will change with repeated cycling and this change can have a significant impact on contact stresses.

c. Stress analysis needs to account for material similarity or dissimilarity, material isotropy or anisotropy. Additionally, component life calculations may account for the beneficial effects of residual stress surface treatments or palliative coatings. Up to 100 percent of the Goodman allowable may be used for components with surface enhancements, such as Laser Shock Peening or Low Plasticity Burnishing. This is an option and not a requirement. In order to do this, the retention of residual stresses needs to be demonstrated such that the predicted component failure rate does not exceed 1×10^{-7}/EFH. Similarly, taking a benefit from coatings may only be done if the coating can be demonstrated to be durable enough to survive one inspection cycle (assuming the coating will be stripped and reapplied at each inspection).

d. Stress states near the contact zone, which result from combined steady and vibratory loading, should be shown to be below that which would propagate a crack with a 10:1 aspect ratio and a surface length defined as $2a_o$ in section A.5.6. In the cases where this criterion cannot be met, stress states that result from combined steady and vibratory loading should be shown to be below that which would produce high cycle fatigue failure in a contact region with the high stress gradients typically found in fretting fatigue. The fretting fatigue damage state and the resultant degradation of the fatigue limit should be determined from the predicted vibratory loads and interfacial conditions where fretting fatigue may occur. The reduced fatigue limit should be determined under conditions representative of the fretted region, including consideration of contact and friction stress interaction with applied mechanical stresses.

e. A reduction in design allowables is not required for high cycle fatigue if it can be shown that the stresses and other conditions in the interface region have a negligible effect on the high cycle fatigue limit compared to the undamaged material.

f. Recent research has indicated the design for fretting fatigue has to account for simultaneous application of steady and vibratory loads (HCF/LCF interaction). Additionally, the value of coefficient of friction used in the design analysis needs to be demonstrated to be applicable to the geometry and material selection of the final component. Lastly, the contacting surfaces in fretting fatigue have very complex damage features such as pits, gouges, re-welded material, and small cracks. Any design methodology based on the morphology of damage at the design surface needs to be demonstrated in a relevant environment.

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Reference

[1] Hills, D.A., and Nowell, D., *Mechanics of Fretting Fatigue*, Kluwer Academic Publishers, Dordrecht, Netherlands, 1994.

Foreign object damage (FOD)

High cycle fatigue material allowables should be determined based on the probability of the type and severity of FOD damage. Type of severity of FOD should be determined from prior field experience, analysis and testing. A statistical assessment of the size and occurrence of FOD should be used to supplement field experience, analysis and testing. The statistical assessment should include, but not be limited to: type of impactor, probability of impact, capture area of probable damage, angle of incidence and velocity based upon engine and aircraft speed, damage size, damage shape, impact-induced residual stresses and conditional probability of FOD given FOD on a neighboring blade. A fatigue notch factor (K_f = fatigue strength in smooth bar/fatigue strength in notched bar) criterion may be used in preliminary design for guidance to assess FOD capability (K_f = 3 for preliminary design). However, for final design of components, statistical data should be used to determine a distribution of impactors and impact conditions. These conditions can be used to predict the damage shape and size on the component in question. Predictions of damage shape and size will be done using a combination of realistic impact testing and computational modeling of the impact event and resulting damage. Knowledge about damage characteristics can be combined with mean and vibratory stresses from all modes of interest in the engine operational envelope. For non-inspectable components and all integrally bladed rotors (also known as blisks), the combination of stress state surrounding the damage site, damage geometry, damage location and probability of occurrence should be such that the component failure rate is below the parameter established by section A.4.13.3. (below $1x10^{-7}$ per EFH per stage). For inspectable, non-blisk components, the combination of damage shape and stress state near the damage site should be such that a crack emanating from a FOD site would not grow to failure within two inspection cycles at the appropriate component level. The size of the crack, or lack thereof needs to be determined from the assumed impact parameter using a combination of testing and computational analysis. If these requirements cannot be met in the manner specified above, a FOD/DOD detection system may be installed as a means to reduce the risk of failure to an acceptable level. Blend limits for FOD and criteria for removal or repair should be based on the FOD analysis as required in the preceding paragraph. Material allowables for material which has been repaired, such as by blending or welding, should be adjusted as necessary to account for any degradation of the fatigue limit due to the blend or repair operation. The effect of any redistribution of internal stress by prior processing should be considered in the development of safe blending limits.

Other Damage States

Material allowables should be established based on any other damage state as described in section A.4.6 under the “High Cycle Fatigue” heading.

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VERIFICATION LESSONS LEARNED (A.5.6)

Many durability problems can be traced to the selection of unsuitable materials. The need to reduce engine weight has forced gas turbine engine contractors to make compromises in the selection of materials or higher-risk designs. Characterization and choice of materials should be closely monitored by the Procuring Activity. Material selection should be reviewed under a risk assessment or management plan.

The contract specifications (i.e.; Contract Data Requirements List [CDRL] or elsewhere, as appropriate) should require all data generated under the contract be supplied to the Air Force Research Laboratory (AFRL/ML, ATTN: MR. THEODORE NICHOLAS, 2977 P STREET, WRIGHT-PATTERSON AFB OH 45433-7817) for inclusion into the Damage Tolerance Handbook, MCIC-HB-01.

A.4.7 Parts classification.

All engine parts, components, controls and externals, and expendables will be classified for criticality.

REQUIREMENT RATIONALE (A.4.7)

Parts, components, and expendables need to be classified to ensure the appropriate design requirements are applied.

REQUIREMENT GUIDANCE (A.4.7)

For all applications, the classifications should be fracture critical, durability critical, and durability non-critical. Fracture-critical parts may be further classified as safety- and mission-critical. The engine manufacturer should provide the classification summary.

A failure mode and effects criticality analysis (FMECA), a safety/hazard assessment, or other engineering analysis should provide the basis for classification.

The intent is to apply damage tolerance requirements only to fracture-critical components. Damage tolerance requirements should not, in general, be applied to components in which structural cracking will result in a maintenance burden but not cause inability to sustain flight or complete the mission; i.e., durability-critical parts. However, damage tolerance requirements should be applied to durability-critical parts to: (1) identify components sensitive to manufacturing variables and pre-damage which could cause noneconomical maintenance (e.g., blades), or (2) aid in the establishment of economic repair time or other maintenance actions.

Component classification may be affected by aircraft/engine configuration; i.e., single engine or dual engine. Components for dual engine systems will only be classified as fracture critical if failure would likely cause aircraft loss or if the mission could not be completed. Components for single engine systems should also be classified as fracture critical if failure would result in inability to maintain sustained power. An example is a large blade that would be contained; but, due to progressive damage, the engine-sustained power capability would be insufficient to maintain flight. Controls and accessories should be included in the evaluation and classification of components as fracture critical. Historical records and experience gained during development tests should be used to classify components. Component classification should be established early and should be identified in the contract specifications. The fracture-critical parts list should be updated as required during the development phase based on experience gained during analysis, engine test, and/or flight operations.

REQUIREMENT LESSONS LEARNED (A.4.7)

It has been difficult to determine, in advance, engine parts or components whose failure would have resulted in secondary failure which would have lead to loss of aircraft or essential mission

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capability. Therefore, continued evaluation has been required as the subsystem was being defined. An example is the high pressure hydraulic or pneumatic pressure vessel which, by itself, may not be safety-of-flight critical, but if located on or close to the primary airframe structure could precipitate a failure of the airframe, if it were to fail.

The number of critical parts can be significantly larger for an engine model used in a single engine configuration as opposed to a dual engine configuration. For example, the numbers of critical parts for the F100 engine in the F-15 and F-16 are 45 and 70, respectively, excluding externals.

A.5.7 Parts classification.

The requirement of 4.7 will be evaluated by analysis, inspection, and test.

VERIFICATION RATIONALE (A.5.7)

Evaluation of classification is necessary to ensure the appropriate design requirements are applied.

VERIFICATION GUIDANCE (A.5.7)

A FMECA, in addition to other engineering analyses, could be conducted to evaluate the results. Parts should be classified as fracture critical, safety critical, mission critical, durability critical, or durability non-critical, depending on the application (USA, USAF, USN). The parts classification summary for the same engine may vary with application or use. For example, single-engine versus multi-engine aircraft will have different lists.

VERIFICATION LESSONS LEARNED (A.5.7)

None.

A.4.8 Damage tolerance.

Fracture-/safety- and mission-critical engine parts will be able to maintain adequate damage tolerance in the presence of material, manufacturing, processing, and handling defects for the design service life and design usage specified in 4.3 and 4.4.

REQUIREMENT RATIONALE (A.4.8)

The requirement protects fracture-, safety-, and mission-critical parts from potentially degrading effects of handling damage and/or material, manufacturing, and processing anomalies which could result in premature engine failures and loss of aircraft. This process ensures proper material choices, control of operating stress levels, use of fracture-resistant design concepts, manufacturing and process controls, and the use of reliable inspection methods during production and in-service maintenance. Attainment of engine damage tolerance is achieved through application of detail requirements in the functional areas of design, materials selections, manufacturing control, and inspections.

REQUIREMENT GUIDANCE (A.4.8)

Damage tolerance will be achieved by proper material selection and control, control of stress levels, use of fracture-resistant design concepts, manufacturing and processing controls, and the use of reliable inspection methods. The design objective will be to qualify components as in-service noninspectable to eliminate the need for depot inspections prior to achievement of one design lifetime. As a minimum, components will be qualified as depot- or base-level inspectable structure for the minimum interval.

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Damage tolerance can be achieved by performing crack growth evaluation as an integral part of detail design of fracture-critical engine components. Initial flaws (sharp cracks) should be assumed in highly-stressed locations such as edges, fillets, holes, and blade slots. Imbedded defects (sharp cracks) should also be assumed at large volume locations such as live rim and bore. Growth of these assumed initial flaws as a function of imposed stress cycles should be calculated. Total growth period from initial flaw size to component failure (i.e., the safety limit) is thus derived. Trade studies on: (1) inspection methods and assumed initial flaw size, (2) stress levels, (3) material choice, and (4) structural geometry can be made until the safety limit is sufficiently large such that the need for in-service inspection is eliminated or minimized. Damage tolerance design procedures which account for distribution of variables that affect growth of imbedded defects are permitted (e.g., probability of imbedded defects associated with the specific material and manufacturing processes). Specific requirements on initial flaw sizes, residual strength, critical stress intensities, inspection intervals, damage growth limits, and verification are contained elsewhere in this document.

Damage tolerance requirements may be applied to durability-critical parts to: (1) identify components sensitive to manufacturing variables and pre-damage which could cause non-economical maintenance (e.g., blades), or (2) aid in the establishment of economic repair time or other maintenance actions.

A Damage Tolerance Control Plan should be prepared to identify and schedule each of the tasks and interfaces in the functional areas of design, material selection, manufacturing control, and inspection of fracture-, safety-, and mission-critical parts. The tasks to be identified in the plan include:

a. design concepts/material/weight/performance/cost trade studies;

b. a damage tolerance analysis, development testing, and proof of compliance testing tasks;

c. a parts list that identifies fracture-, safety- and mission-critical parts, locations, and special controls required to meet damage tolerance requirements (e.g.; material specification controls, quality assurance requirements, etc.);

d. zoning of drawings for fracture-, safety- and mission-critical parts to identify critical locations and associated quality control requirements, defect locations, orientation, inspection method, and acceptance standards. The use of alternate procedures for identification of critical locations, etc., on drawings may be proposed.

e. basic materials fracture data (e.g.; K_{IC}, K_C, K_{ISCC}, da/dn)

f. identification and control of fracture toughness and crack growth rate properties in the material procurement and manufacturing process specifications

g. traceability requirements on all tiers of procurement, processing, fabrication, and assembly for fracture-critical components; Serialization or time coding requirements for tracking operational exposure of individual components.

h. quality control requirements during component manufacture. Identification of procedures for certifying and monitoring subcontractor, vendor, and supplier inspection and quality control. Nondestructive inspection requirements for use during depot- and base-level inspections, including supporting manuals (technical orders) and equipment needs.

An example of damage-tolerant design principles (criteria, design, analysis, and substantiation) is contained in AFWAL-TR-81-2045.

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REQUIREMENT LESSONS LEARNED (A.4.8)

There have been numerous Class A incidents on Air Force aircraft and engines due to structural failures caused by material defects, manufacturing defects, or fatigue-induced cracks. These defects grew in size due to repetitive cycles of maneuvers or throttle excursions until such time as the residual strength of the component became less than the applied load and failure occurred. Causes have been: (1) use of high strength, low fracture toughness materials, (2) improper detail designs which result in high stress levels and structural discontinuities, and (3) lack of adequate quality control requirements (both in production and depot maintenance). Also, past review of commercial engine experience reveals noncontained failures of blades, disks, and spacers, due to structural cracking. When compared against the total number of parts and flying hours, these occurrences are low. However, the demonstrated consequences of failure on USAF systems have been high: lost crewmembers and aircraft. Many of the incidents could have been avoided by proper material selection, control of stress levels, use of fracture-resistant design concepts, manufacturing and process controls, and use of reliable inspection methods during production and in-service maintenance.

Recent examples of optimized part designs following the ENSIP/DTD guidelines include the F109-GA-100, F100-PW-220 ILC, F100-PW-229 IPE, F110-GE-129 IPE, and F119-PW-100 ATF engine designs. These design configurations have shown that damage tolerance requirements can be met with small or modest increases in overall engine weight, will have little impact on engine performance, and will provide greatly-improved engine durability while weapon system life cycle cost is significantly reduced.

Most of the tasks to be contained in the Damage Tolerance Control Plan have been accomplished by engine manufacturers in legacy development and production programs. However, the durability and damage tolerance requirements established here impose tighter controls and more interface involvement between the functional areas.

A.5.8 Damage tolerance.

Damage tolerance of fracture-critical engine components will be in accordance with 4.8. Verification will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.8)

Damage tolerance analyses are needed to support damage tolerance design concepts, material selection, maintenance requirements, performance, cost, and weight impacts. Damage tolerance tests are required to support material selection and trade studies, obtain early evaluation of allowable stress levels and chemical/thermal environment spectra, and to verify analysis procedures and damage tolerance characteristics.

VERIFICATION GUIDANCE (A.5.8)

Early analysis will enable identification of structurally-sensitive areas which do not meet the desired crack growth intervals, and design changes can be introduced early with minimum impact. Emphasis on conducting early analysis will minimize occurrence of deficiencies in later development and proof-of-compliance testing, and facilitate attainment of important System Development & Demonstration Phase test milestones.

Attainment of damage-tolerant parts is achieved through application of detail requirements in the functional areas of design, materials selections, manufacturing control, and inspections. Most of the tasks to be contained in the damage tolerance control plan have been accomplished by engine manufacturers in past development and production programs. However, the damage tolerance requirement discussed in this handbook imposes the need for new tasks as well as tighter controls and more interface involvement between the functional areas.

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Other methods of compliance to damage tolerance requirements are fracture screening, leak-before-burst for a specified interval for plumbing and control housings, the use of redundant structures or controls, and, as a last resort, remote probability may be used, but only with Using Service approval.

Damage tolerance analysis should be conducted on each component classified as fracture critical by A.4.7 of this document. Each location of the component will be surveyed to determine the most critical locations for the assumed initial flaw; such features as edges, fillets, holes, blade slots, and other high-stressed areas will be considered. Stress/environment spectra will be developed for each component and location to be analyzed. Imbedded defects will be assumed to exist within large volume locations such as live rims and bores. Damage tolerance analysis that addresses imbedded defects can be based on probabilistic methods that account for the distribution of variables. Interactions between assumed initial flaws at different locations on a component need not be considered. Average (B50) fracture mechanics properties can be used in the crack growth and residual strength calculations. Stress intensity will be based on the structural geometry and assumed flaw geometry. The critical stress intensity will be based on the required residual strength load level and temperature conditions that exist at the component location being analyzed. Limiting stress intensity will include consideration of the allowable stress intensity and account for the effect of vibratory stresses. Certain minimum levels of vibratory stress; e.g., 10 Ksi; should be assumed to exist on each fracture-critical part to identify sensitive components. The requirement for a damage tolerance analysis and the associated schedule should be contained in the contract specifications.

Early testing should include tests of simple specimens, small elements, and subscale components that represent critical structural details and materials, and full-scale components such as disks. Evaluation of component damage tolerance characteristics during full-scale engine test may be required to demonstrate proof of compliance under realistic environments.

Specimen and element tests should be conducted on representative structural details and materials. Representative fracture-critical components will be selected for evaluation by test. These components will either be preflawed or contain natural flaws and will be cycled to evaluate flaw growth characteristics. Preflaws should be sharpened via precycling, vibration, scratching with a razor blade or other sharp instrument, etching, electro-discharge machining (EDM), or tackwelding of the surface, or other means to ensure flaw growth for evaluation/correlation of analyses. Some cycling of parts may be required to "grow" the preflaw to the necessary size prior to actual testing. Electro-discharged machined preflaws are "clean" (sharply defined). Tackweld preflaws are "dirty" (difficult to determine crack length and to differentiate crack lines from thermal cracks).

Components previously cycled to evaluate low cycle fatigue should be used for damage tolerance testing. Test results will be correlated with predictions of crack growth intervals and critical flaw sizes. The damage tolerance test program will be of sufficient scope to verify fracture-critical parts. Deletion of verification of certain fracture-critical parts can be proposed based on similarity of materials and structural configurations and demonstrated knowledge of the applied stresses. The scope of the damage tolerance design development test program and associated schedule should be contained in the ENSIP Master Plan or contract specifications. After contract award, the test plan should be finalized and submitted to the USAF for approval. The test plan will be revised and maintained current during full-scale development. Information such as rationale for selection of scope of tests, description of test procedures, loads, and duration of tests should be included in the test plan. Sufficient tests to evaluate allowable stress levels and to support material selection should be scheduled for completion prior to the Critical Design Review (CDR). Component tests will be scheduled for completion prior to an ISR decision.

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The amount of full-scale engine damage tolerance testing that is required is dependent on the extent damage tolerance is demonstrated by earlier component tests and other full-scale testing (i.e., number of cracking incidents and subsequent crack growth that occurs during accelerated mission tests).

VERIFICATION LESSONS LEARNED (A.5.8)

None.

A.4.8.1 Residual strength.

The residual strength will be equal to the maximum stress that occurs during design usage conditions. Residual strength requirements will be established for all damage-tolerant-designed parts and components. Associated static and dynamic loading conditions for these parts and components should be included.

REQUIREMENT RATIONALE (A.4.8.1)

The load-carrying capability of fracture-, safety-, and mission-critical parts, with "damage" present, need to remain above some minimum value during part design service lives and unrepaired service usage.

REQUIREMENT GUIDANCE (A.4.8.1)

The static and dynamic loading conditions which should be considered are:

a. maximum limit maneuver loading,

b. maximum pressure loading,

c. maximum speed loading, and

d. maximum temperature effects.

The engine contractor should provide the residual strength requirements for all parts. The minimum residual strength for each part (and location) should be equal to the maximum stress that occurs during design usage conditions. Normal or expected control system overspeed (e.g., 105 percent) and engine deterioration will be included. Burst margin overspeed conditions should be excluded.

Analytical studies have shown that not every part location will be limited by a crack growing to a calculated critical stress intensity equal to the material's fracture toughness. Some part locations will in fact be life-limited by cracks growing to a predicted vibratory threshold DK_{th} HCF.

Where, $K_{max\ allowable}$ LCF = ΔK_{th} HCF/(1 - R)

and R = $(\sigma_{steady} - \sigma_{vibratory}) / (\sigma_{steady} + \sigma_{vibratory})$

σ_{steady} = maximum operating stress neglecting vibratory stress

$\sigma_{vibratory}$ = ½ (peak to peak vibratory stress)

and ΔK_{th} HCF = f(R, temp) .

Overspeed residual strength requirements need not be considered for those part locations limited by cracks reaching a calculated vibratory threshold. One overspeed cycle occurring at a crack size equal to the vibratory threshold creates less damage (change in crack size) than additional LCF/HCF crack growth from the vibratory threshold to a maximum stress intensity (K_{CRIT}) defined by the material fracture toughness.

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For those locations not limited by vibratory stress concerns, the part's maximum allowable crack size should be limited to a size that will survive the maximum design stress that occurs on the last cycle of the calculated safety limit.

REQUIREMENT LESSONS LEARNED (A.4.8.1)

None.

A.5.8.1 Residual strength.

The requirements of 4.8.1 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.8.1)

Evaluation of residual strength by analyses and tests is required to ensure less-than-critical-size flaws will not grow and cause failure due to the application of the required residual strength load.

VERIFICATION GUIDANCE (A.5.8.1)

Analyses should assume the presence of flaws in the most unfavorable location with regard to geometry stress and material properties and should show that at the end of the required damage tolerance operational period, the strength requirement can be met for this flaw configuration and the required load.

The testing should be conducted in accordance with A.4.8.

VERIFICATION LESSONS LEARNED (A.5.8.1)

None.

A.4.8.2 Initial flaw size.

Initial flaws will be assumed to exist as a result of material, manufacturing, and processing operations. Assumed initial flaw sizes will be based on the intrinsic material defect distribution, manufacturing process, and the nondestructive inspection (NDI) methods to be used during manufacture of the component.

REQUIREMENT RATIONALE (A.4.8.2)

This requirement is necessary to establish the probable flaw size that can exist in a part after manufacture. Damage tolerance, as applied in ENSIP, assumes the presence of a flaw in each fracture-, safety-, and mission-critical part, at the highest stressed location with an orientation most unfavorable with respect to the stress field. Tolerance to these initial flaws needs to be designed into the part.

REQUIREMENT GUIDANCE (A.4.8.2)

Flaw sizes are intended to represent the maximum damage that can exist in a part after manufacture. Assumed initial surface flaw sizes can be based on the NDI methods to be used during manufacture. Assumed initial imbedded flaw sizes will be based on the intrinsic material defect distribution or the NDI methods to be used during manufacture. The initial flaw sizes are used to determine the safe crack growth period (i.e., safety limit). Initial flaw sizes should be proposed by the manufacturer and subject to approval by the Using Service. Demonstration of flaw size detection reliability should be required.

If an NDI method reliability of 90 percent Probability of Detection (POD)/95 percent Confidence Level (CL) is assumed, the minimum initial flaw sizes for crack growth analysis should be as listed in tables XVI and XVII.

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TABLE XIVI. Minimum initial flaw sizes for crack growth analysis with NDI method reliability of 90 percent POD/95 percent CL.

INSPECTION METHOD	MATERIAL	FLAW TYPE	FLAW SIZE (DEPTH X LENGTH)
Manual	All	Surface	0.035 X 0.070 inches
Manual	All	Corner	0.035 X 0.035 inches
Manual	All	Imbedded	0.035-inch diameter
Semi-Auto	All	Surface	0.020 X 0.040 inches
Semi-Auto	All	Corner	0.020 X 0.020 inches
Semi-Auto	All	Imbedded	0.035-inch diameter
Automated	Ti	Surface	0.010 X 0.020 inches
Automated	Ni	Surface	0.007 X 0.014 inches
Automated	Ti	Corner	0.010 X 0.010 inches
Automated	Ni	Corner	0.007 X 0.007 inches
Automated	Ti	Imbedded	0.025-inch diameter
Automated	Ni	Imbedded	0.032-inch diameter

If an NDI method reliability of 90 percentPOD/50 percentCL is assumed, the minimum initial flaw sizes for crack growth analysis should be:

TABLE XVII. Minimum initial flaw sizes for crack growth analysis with NDI method reliability of 90 percentPOD/50 percentCL.

INSPECTION METHOD	MATERIAL	FLAW TYPE	FLAW SIZE (DEPTH X LENGTH)
Automated	Ti	Surface	0.010 X 0.020 inches
Automated	Ni	Surface	0.005 X 0.010 inches
Automated	Ti	Corner	0.010 X 0.010 inches
Automated	Ni	Corner	0.005 X 0.005 inches
Automated	Ti	Imbedded	0.017-inch diameter
Automated	Ni	Imbedded	0.020-inch diameter

The initial flaw size detectability requirement of 90 percent POD/95 percent CL should be used for all manual and semi-automated NDI methods. The 90 percent POD/50 percent CL requirement can be used for some automated NDI methods based on the NDI process being in control. The 90 percent POD/95 percent CL was originally used because manual Fluorescent Penetrant Inspection (FPI) was the most common method of inspection and was highly operator-dependent. Operator variability is the most influential single variable on reliability demonstrations/testing. With the introduction of enhanced automated eddy current inspection

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systems, the POD/CL requirement was changed to 90 percent POD/50 percent CL to reflect the reduced/removed operator variability. However, demonstration of flaw size detection reliability should be required to ensure the system is a controlled process. For detectability requirements of 90 percent POD/95 percent CL, there may be a larger test matrix, more specimens, and thus higher cost necessary to achieve the statistical 95 percent CL.

It is recommended that initial design and sizing of components be based on .040-inch surface flaws or .020-inch by .020-inch corner cracks. The basis for this recommendation is twofold: (1) it establishes an initial flaw size that will support the use of FPI as the standard NDI method at production and depot, and (2) it provides capability for application of upgraded NDI methods at a few locations when full-scale development results indicate the need due to higher than anticipated stresses/usage.

Initial design should also account for .020-in.-diameter imbedded flaws in large surface areas. The imbedded flaws in weldments should have a diameter equal to 20 percent of the thickness of the weld. These initial flaw assumptions provide some margin when analysis and test results indicate that stresses are higher than anticipated.

REQUIREMENT LESSONS LEARNED (A.4.8.2)

A review of aircraft and engine experience reveals that unexpected premature cracking occurs at high-stressed areas. Initial conditions have included material and manufacturing defects (voids, inclusions, machining marks, scratches, sharp cracks, etc.).

Flaw detection capabilities of the various nondestructive testing (NDT) methods are affected by a wide variety of variables. Key to high reliability in NDT methods is high proficiency of the inspection personnel. Also important is the availability of recorded inspection data for evaluation of the characteristics of various sources of initial damage. Redundant inspections also improve the reliability of detecting flaws; i.e., independent applications of the same NDT method or use of different methods. Experience has identified preferred processing methods or key processing parameters as indicated below.

Fluorescent Penetrant Inspection (FPI) - The preferred process utilizes a high sensitivity post emulsified penetrant with a hydrophilic (water soluble) emulsifier. A nonaqueous or water soluble developer is preferred over the dry powder or wet suspendable type of developer. Surface preparation is very important and should include an etch to remove smeared metal, dirt, combustion products, etc. In-process FPI preceded by a heavy etch that removes considerable surface material (.001 to .002 inches) is recommended for each fracture-critical rotating component. Etch of the finished component prior to FPI is also recommended for each fracture-critical component. However, the selection of an etchant for the finished component needs to be evaluated thoroughly to ensure no detrimental life effects are caused when adequate surface material removal is achieved (.0001 to .0002 inches).

Fluorescent Penetrant Inspection - The preferred process uses automated processing, and special fixturing, when necessary.

Principle:	Liquid penetrant is drawn into surface flaws by capillary action. Flaws are revealed by a dye.
Applications:	Applications include surface cracks, laps, porosity, shrinkage areas, laminations, etc., that are open to surface.
Advantages:	This process is inexpensive, portable, and sensitive. It is not dependent on magnetic or electrical properties of a material.
Disadvantages:	Flaws on the surface of the part. Not usable on porous and rough surfaces.

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Eddy Current Inspection - The preferred process uses automated scanning and automated data recording, and special fixturing, when necessary.

Principle: The coil induces a current in a metal then detects the resulting current that fluctuates with a change in electrical property of the part.

Applications: Applications include surface cracks, laps, porosity, shrinkage areas, laminations, etc., that are open to surface. Detects variations in the metal and microstructure of parts.

Advantages: This process is very sensitive. It is not necessary to contact part.

Disadvantages: This process is often too sensitive to unimportant properties.

Ultrasonic Inspection - The preferred process uses automated scanning and may use more than one mode (i.e.; longitudinal, shear, surface, lamb, etc.), where appropriate.

Principle: Sound waves are transmitted through material and reflected by flaws.

Applications: Applicable to subsurface flaws such as cracks, laminations, and bonds where the principle plane is perpendicular to the sound source.

Advantages: The process is able to detect flaws that exist deep in a material. The capability to produce images of the flaw exists.

Disadvantages: Flaws that are parallel to the source of sound are undetectable. Inspection is to be performed by trained personnel.

Radiographic Inspection - The preferred process requires selection of proper kilovoltage and exposure geometry.

Principle: X-rays and gamma rays are sent through the metal and strike a film. The existence of flaws is seen as dark shadows on the film.

Applications: This process is applicable to subsurface flaws in castings and weldments with the principle plane of flaw parallel to radiation beam.

Advantages: Flaws are detectable at any depth.

Disadvantages: Flaws perpendicular to radiation beam are undetectable. Inspection may be hazardous and is to be performed by trained personnel. Inspection is expensive.

Magnetic Particle Inspection - The preferred process provides that adequate magnetic field strengths are introduced in the part and at critical locations in the part.

Principle: Discontinuities distort an applied magnetic field and cause leakage fields that attract iron powder.

Applications: This process is applicable to cracks, inclusions, and other discontinuities on or near the surface of parts.

Advantages: The process is inexpensive. It is suitable for extremely large objects.

Disadvantages: This process is limited by depth of flaw and coatings. Personnel who perform inspections are to be trained to interpret results.

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A.5.8.2 Initial flaw size.

Material controls, manufacturing process controls, and in-process nondestructive inspection (NDI) will be performed on each fracture-critical component to ensure the requirements of 4.8.2 are met.

VERIFICATION RATIONALE (A.5.8.2)

Initial defect sizes depend on the detail NDI method and/or manufacturing process controls to be employed in production. Particular values selected for design have significance only when evaluated by demonstration programs. Evaluation of the initial flaw sizes is necessary to ensure flaw sizes greater than the those sizes assumed do not exist in finished parts.

VERIFICATION GUIDANCE (A.5.8.2)

Controls and inspection methods will be established through the damage tolerance control plan. Damage tolerance of many parts is achieved by providing minimum flaw growth intervals based on initial flaw sizes. The initial flaw size values selected for design only have significance when production NDI capability is confirmed by the demonstration programs. Demonstration programs, in the absence of existing data, should be performed to ensure flaws greater than the design flaws of A.4.8.2 will not occur in finished components. Subsequent to successful completion of these demonstration programs, the selected inspection methods and processes will become a part of the production requirements and may not be changed without approval of the Procuring Activity.

It is recommended that initial flaw size based on NDI methods be demonstrated to have a probability of detection and confidence level of 90 percent/95 percent.

VERIFICATION LESSONS LEARNED (A.5.8.2)

In past programs, inspection capabilities have been quoted for generic areas and processes with no real basis in reality. For example, an eddy current capability was quoted as .010 x .020 inch for Titanium broach slots. However, eddy current could not detect this or any reasonable flaw size within .100-inch of the edge of the broach slot due to the geometry signal. Today, detection of flaws in edges is possible with certain probes, but the inspection time is three times slower than an inspection which does not include the edge. This was only discovered with an NDI demonstration program.

A.4.8.3 In-service inspection flaw size.

The flaw size, which will be presumed to exist in a component after completion of a depot-, intermediate-, or base-level inspection, will be specified.

REQUIREMENT RATIONALE (A.4.8.3)

This requirement is necessary to establish the probable flaw sizes that can exist in a part after a depot-, intermediate-, or base-level inspection. In-service inspection flaw sizes need be specified to establish part-life limitations, and the maintenance capability requirements.

REQUIREMENT GUIDANCE (A.4.8.3)

Although this paragraph establishes a requirement that applies to a post-EMD activity, the information (given up-front) is relative to the logistic requirements for the engine. The probable flaw sizes assumed to exist in a part after completion of a depot-, intermediate-, or base-level inspection need be consistent with NDI capability used during in-service inspections. It is not essential for the assumed flaw sizes following depot-, intermediate-, or base-level inspections to be the same as those following production inspections, as long as inspectability is insured. However, in-service inspection flaw sizes should be larger than or equal to those flaw sizes

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detectable through current NDI methods. Flaw sizes for in-service inspectable flaws and in-service noninspectable flaws should be based on the NDI methods incorporated into the life management plan. The re-occurring inspection interval will be based on the assumed flaw size after completion of the initial depot-, intermediate-, or base-level inspection.

Flaw size detection capability versus inspection method should be the same as that specified in A.4.8.2, provided the component is removed from the engine and completely inspected with procedures which provide the same degree of confidence and sensitivity as those performed during production. Where etching or other necessary surface preparation is not practical or possible on in-service components, FPI should not be used.

REQUIREMENT LESSONS LEARNED (A.4.8.3)

See A.4.8.2 Lessons Learned.

A.5.8.3 In-service inspection flaw size.

The requirements of 4.8.3 will be evaluated by analysis, inspection, demonstration, and test.

VERIFICATION RATIONALE (A.5.8.3)

This requirement is necessary to establish the probable flaw sizes that can exist in a part after a depot-, intermediate-, or base-level inspection. In-service inspection flaw sizes need be specified to establish part-life limitations, and the maintenance capability requirements.

VERIFICATION GUIDANCE (A.5.8.3)

The Using Service should be able to demonstrate the necessary NDI reliability. The NDI reliability should provide for a probability of detection at the lower-bound confidence level consistent with A.4.8.3. Although this evaluation paragraph seems to apply only to a post-EMD activity, it actually provides information (up-front) relative to logistic requirements. In-service inspection flaw sizes should be larger than or equal to those flaw sizes detectable through current NDI methods. The engine contractor should include the in-service inspection requirements (methods and intervals) in the life management plan. Logisticians may not be able to provision for detection of unreasonable flaw size values (smaller than current NDI capability).

In-service inspection flaw size detection capability should be demonstrated on parts with a significant amount of prior engine operation time. This will provide a better understanding of in-service inspection flaw size capability.

It is recommended that initial flaw size based on NDI methods be demonstrated to have a probability of detection and confidence level of 90 percent/95 percent.

VERIFICATION LESSONS LEARNED (A.5.8.3)

Reliability assessment methodologies can provide very different POD results based on the same data sets. MILITARY-HANDBOOK-1823 provides a reliability assessment procedure to establish POD and should be used for all POD and process quality control assessments. Software based on MIL-HDBK-1823 to standardize POD calculations was developed under contract with the University of Dayton Research Institute (UDRI) and should be used for POD calculations.

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A.4.8.4 Inspection intervals.

The frequency of inspection in terms of the required design lifetime will be specified in terms of

a. in-service noninspectable—once at the end of one design lifetime, or

b. depot- or base-level inspectable.

REQUIREMENT RATIONALE (A.4.8.4)

The design objective is to eliminate the need for in-service inspections to achieve damage tolerance. However, the weight penalty incurred to achieve a safety-limit or damage growth interval sufficiently large to preclude the need for in-service inspections may be prohibitive on some components. Therefore, in-service inspections will be allowed on some parts, subject to justification.

These requirements are intended to provide the minimum information necessary to show basic maintenance functions have been considered in the design of the engine.

REQUIREMENT GUIDANCE (A.4.8.4)

Inspection intervals should be compatible with the overall weapon system maintenance plan. The inspection intervals may be proposed by the contractor and approved by the Using Service. The in-service noninspectable period should be at least one times (1 X) the design service life specified in A.4.3. The minimum depot-, intermediate-, or base-level inspection interval should be equal to the hot parts life of A.4.3.1. The inspection intervals, when accepted by the Using Service, should be contained in the contract specification.

Parts are usually designated "in-service noninspectable" because: (1) inspection capability precludes detection of flaws (i.e., parts contain imbedded flaws or inaccessible flaw regions) or (2) the part is not intended to be inspected during its design life. Parts designated as depot-, intermediate-, or base-level inspectable are classified as such because inspection capability exists such that they can be readily inspected (i.e., surface flaw inspection).

REQUIREMENT LESSONS LEARNED (A.4.8.4)

The design objective of damage tolerance is to qualify parts as "in-service noninspectable". However, the weight penalty incurred to achieve a flaw growth interval sufficiently large to preclude the need for in-service inspections may be prohibitive for some parts.

A.5.8.4 Inspection intervals.

The requirements of 4.8.4 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.8.4)

Evaluation of inspection intervals is required to ensure the flaw growth interval of A.4.8.5 is of sufficient duration to preclude failure between inspections.

VERIFICATION GUIDANCE (A.5.8.4)

The test should be conducted in accordance with A.4.8.

Inspections after engine testing should provide data that substantiates the flaw growth interval of A.4.8.5. This data should be compared to the time between inspections to ensure that the appropriate inspection interval has been chosen.

VERIFICATION LESSONS LEARNED (A.5.8.4)

None.

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A.4.8.5 Flaw growth.

The initial flaw sizes specified in 4.8.2 will not grow to critical size and cause failure of the part due to the application of the required residual strength load within two times (2 X) the specified inspection interval.

REQUIREMENT RATIONALE (A.4.8.5)

The flaw growth interval (i.e., safety limit) should be specified to ensure the assumed initial flaw will not grow as a function of usage to critical size that would cause unstable growth and fail due to application of the required residual strength load.

REQUIREMENT GUIDANCE (A.4.8.5)

The flaw growth interval is also known as the safety limit. It is recommended that the flaw growth intervals be twice the inspection intervals specified in A.4.8.4. Flaw growth interval margins, other than two, can be used when individual assessments of variables (i.e.; initial flaw size, da/dN, K_{IC}, etc.) that affect flaw growth can be made (e.g., to account for observed scatter in crack growth during testing). The following should be considered in treatment of variables which can affect the calculation of the flaw growth interval:

a. No life credit should be taken of any beneficial effects of residual stresses or surface treatments such as shot peening or coatings, except when the following conditions are met:

(1) The beneficial effect of residual stresses or a surface treatment can be verified through analysis and testing for the duration of the service interval and under the actual anticipated usage and maintenance conditions such as polishing for eddy current inspection.

(2) Quality control procedures for application and in-service integrity are demonstrated to a satisfactory level of reliability.

These beneficial effects should be verified and the extent of life "crediting" should be approved by the Procuring Activity.

b. Damage in a primary structure may result in load increases in the secondary structure. The analysis of such secondary structures should account for this.

c. Continuing damage should be assumed at critical locations where the initial damage assumption does not result in failure of the part (e.g., the case of a free surface at a bolthole). The following assumptions of initial damage and location should be considered with the limiting condition used to establish safety limits and inspection intervals:

(1) When the primary crack and subsequent growth terminates prior to component failure, an initial flaw equal to or greater than that which is demonstrated to be inspectable in 4.8.3 should be assumed to exist at the opposite side of the feature after the primary crack has terminated. The stress gradient assumed at the opposite location should be based on the boundary conditions that exist when crack growth has terminated at the primary location. The safety limit for this condition should be the sum of the crack growth at the primary location and at the opposite location.

(2) Growth of an assumed initial flaw at the location opposite the primary location should be verified as an initial condition.

d. The effects of vibratory stress on unstable crack growth should be accounted for when the safety limit is established. Threshold crack size should be established at each individual sustained power condition (Idle, Cruise, Intermediate) using the appropriate values of steady stress and vibratory stress. The largest threshold crack size should be used as a limiting value in calculation of the safety limit if it is less than the critical crack size associated with

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the material fracture toughness. An analytical approach to defining the effects of vibratory stress is based on a maximum stress intensity allowable, $K_{max\ allowable}$ LCF (as a function of stress ratio, R), which is predicted from appropriate material high cycle fatigue vibratory threshold DK_{th} HCF (as a function of stress ratio, R) properties at steady-state operating conditions. This relationship is as follows:

assume	$K_{max\ allowable}$ LCF	=	ΔK_{th} HCF/(1 - R)
where,	R	=	$(\sigma_{steady} - \sigma_{vibratory}) / (\sigma_{steady} + \sigma_{vibratory})$
	σ_{steady}	=	maximum operating stress neglecting vibratory stress
	$\sigma_{vibratory}$	=	½ (peak to peak vibratory stress)
and	ΔK_{th} HCF	=	f(R, temp) .

ΔK_{th} HCF versus R-ratio material property curves used in this verification at various temperatures should be developed during material characterization, as necessary.

e. The ΔK allowable for threshold crack growth rate should be based on the crack length under consideration; the maximum allowable crack extension which will not produce failure, instability, or measurable change in dynamic response characteristics; and number of high cycle fatigue cycles between inspections for damage-tolerant components, or full fatigue life (per section A.4.13.3) for durability-limited components.

It should be demonstrated, through analysis or testing, that the limit in the number of allowable high cycle fatigue cycles should not be exceeded as specified above for the values of da/dN_{th} and ΔK_{th} chosen for the definition of the threshold condition.

For conditions where da/dN versus ΔK is not known (for example, in the small crack regime or where a crack has not formed to a measurable or deterministic size), then the threshold condition should be based on a fatigue limit (as specified in section A.4.13.3) for the required number of cycles. In general, the threshold crack growth rate condition, da/dN_{th}, *and* the fatigue limit that corresponds to both the appropriate number of high cycle fatigue cycles required by the design condition and to the maximum allowable crack size or damage state should be met.

f. Galling/fretting limits (i.e., permissible depth of surface damage) for all contact surfaces should be defined based on $K_{max\ allowable}$ LCF (as a function of stress ratio, R) or stress analysis that demonstrates that the fatigue limit is not exceeded for the specific part under the applied steady and vibratory stresses.

g. Calculation of hold time crack growth. Assessments of material/design acceptability do not typically account for the flaw growth under constant load conditions. The duty cycle is compressed such that only the cyclic content is preserved. If the constant load duty cycle content is neglected, the rate of growth of cracks can be underestimated. Early assessments of material/design candidates should include the effects of hold time crack growth under representative load and temperature conditions. Initial screening and subsequent re-screening as the duty cycle matures should be performed to pre-empt service failures.

REQUIREMENT LESSONS LEARNED (A.4.8.5)

Since average fracture properties have been used in analysis, parts made from materials with scatter factors greater than two have failed prior to their inspection interval. Thus, for materials with large scatter factors (i.e., greater than two), factors of safety greater than two, on residual life, should be considered. In addition, the USAF has experienced several disk post (lug)

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failures attributed to high stress gradients arising at the Edge of Contact (EoC) between the blade and disk. Conventional lifing assessment practices did not address the susceptibility of the material/design to prevent the cracks from propagating under hold time crack growth conditions.

A.5.8.5 Flaw growth.

The requirements of 4.8.5 should be verified by analyses and tests.

VERIFICATION RATIONALE (A.5.8.5)

Verification of flaw growth is necessary to ensure initial flaws will not grow to critical size and cause failure due to the application of the required residual strength load.

VERIFICATION GUIDANCE (A.5.8.5)

The test should be conducted in accordance with A.4.8.

Analyses should demonstrate that the assumed initial flaws will not grow to critical size for the usage, environment, and required damage tolerance operational period. The analyses should account for repeated and sustained stresses, environments, and temperatures, and should include the effects of load interactions. Analysis methods should be verified by test, utilizing engine and rig testing.

VERIFICATION LESSONS LEARNED (A.5.8.5)

None.

A.4.8.6 Composites.

Composite parts will be damage-tolerant with defects resulting from material quality, manufacturing processing, and handling damage.

REQUIREMENT RATIONALE (A.4.8.6)

Damage tolerance of composites need be specified since they are a special type of fracture-, mission-, and safety-critical part. This paragraph is needed to establish a means for composites to comply with damage tolerance requirements.

REQUIREMENT GUIDANCE (A.4.8.6)

Damage-tolerant design of organic matrix composite (OMC) parts is extremely complex. Composites exhibit near-linear stress-strain characteristics up to failure, while most metals display some ductile deformation. Hence, composites are less tolerant to overload than metals. Composites generally exhibit good resistance to tension fatigue and are susceptible to local delaminations which result from compression fatigue.

Because of the multi-phase nature of the materials used in composites, a substantially higher number of defects may exist in a composite part than would occur in a metallic part.

Handling damage to composites includes scratches, gouges, delamination, and fiber breakage. In these instances, delaminations and fiber breakage are usually the result of impact damage. The extent and type of damage which results from impact on composites depends on the energy involved in the impact.

Defects in composites due to manufacture are usually of two types: (1) those produced during the preparation and production of the composite and (2) those produced during machining, processing, and assembly of the final component.

Some composite materials are known to absorb moisture and lose strength over time.

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REQUIREMENT LESSONS LEARNED (A.4.8.6)

Damage-tolerance requirements for composite parts have been derived from the work done for the Aircraft Structural Integrity Program (ASIP). (See MIL-HDBK-1530.)

A.5.8.6 Composites.

The requirements of 4.8.6 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.8.6)

Tests and analyses need be performed to evaluate the damage tolerance of fiber composites.

VERIFICATION GUIDANCE (A.5.8.6)

The following should be transferred, verbatim, into the specification paragraph: "Composite parts should be subjected to impact damage equivalent to 100 ft-lbs (74 N-m), using a one-inch (2.54 cm)-diameter spherical impactor, then operated for the periods specified in A.4.8.4.

Composite parts should be subjected to impact damage to evaluate damage tolerance to handling-/maintenance-induced impact damage. The contractor should propose other means to evaluate the damage tolerance of organic matrix composite (OMC) parts."

VERIFICATION LESSONS LEARNED (A.5.8.6)

Methods for evaluation of damage tolerance of composites is derived from the work done for the Aircraft Structural Integrity Program (ASIP). (See MIL-HDBK-1530.)

A.4.9 Durability/economic life.

The durability/economic life of the engine will not be less than the required design service life when subjected to the design usage.

REQUIREMENT RATIONALE (A.4.9)

Durability requirements need be applied since engine durability is a primary design requirement. Durability requirements need be applied to minimize cracking or other structural or material degradation which could result in functional impairment or excessive in-service maintenance problems and costs.

REQUIREMENT GUIDANCE (A.4.9)

Durability of engine components will be obtained by proper material selection and control, control of stress levels, detail design, and use of protection systems. A durability control plan should be prepared to identify and schedule each of the tasks and interfaces in the functional areas of design, material selection, manufacturing control, and inspection of engine parts and components. The tasks to be identified in the plan should include:

a. design concepts/material/weight/performance/cost trade studies

b. life analysis, development testing, and proof of compliance testing tasks

c. a list that identifies parts, locations, and special controls required to meet life requirements (e.g.; material specification controls, quality assurance requirements, etc.)

d. zoning of drawings for parts to identify critical locations and associated quality control requirements, defect locations, orientation, inspection method, and acceptance standards. The use of alternate procedures for identification of critical locations, etc., on drawings may be proposed.

e. basic materials data

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f. identification and control of variables that affect properties in the material procurement and manufacturing process specifications

g. traceability requirements on all tiers of procurement, processing, fabrication, and assembly for durability-critical parts. Serialization or time coding requirements for tracking operational exposure of individual parts.

h. Quality control requirements during manufacture. Subcontractor, vendor, and supplier quality control requirements during manufacture. Identification of procedures for certifying and monitoring subcontractor, vendor, and supplier inspection and quality control.

i. Nondestructive inspection requirements for use during depot- and base-level inspections, including supporting manuals (Technical Orders) and equipment needs.

Most of the tasks contained in the Durability Control Plan have been accomplished by engine manufacturers in legacy development and production programs. However, the durability requirements established here impose tighter controls and more interface involvement between the functional areas.

REQUIREMENT LESSONS LEARNED (A.4.9)

Increased turbine engine performance requirements have resulted in higher thrust-to-weight ratios. This trend has led to higher stresses in engine components which in turn has forced development of specialized high-strength alloys and new manufacturing processes and techniques. As a result, USAF engine systems have experienced early structural cracking and often contain components that have to be replaced one or more times during the operational life of the engine. Experience with past engine systems highlights the fact that the Air Force needs a disciplined approach for design, analysis, test, and improvement of engine components so that in-service maintenance and component replacement needs over the operational life of the engine will be minimized.

The J85-21 engine had a compressor blade flutter problem (high cycle fatigue) which was discovered after the engine had successfully passed qualification testing and after engine production started. Structural failures during one-type engine qualification testing threatened the existence of the weapon system program and raised serious questions about military engine procurement techniques. Fatigue failures in military aircraft engines have caused loss of life and aircraft. Although one Navy fighter had two engines, engine problems resulted in loss of life and aircraft. Cracking of expensive turbine blades in the engine produced serious maintenance support problems through excessive replacement rates during engine overhaul.

A.5.9 Durability.

The requirements of 4.9 will be evaluated by a strength and life analysis, inspection, demonstration, and part, component, and full-scale engine tests.

VERIFICATION RATIONALE (A.5.9)

Attainment of durable parts is achieved through application of detail requirements in the functional areas of design, materials selection, manufacturing control, and inspection. The strength and life analysis report is one of the most important reports submitted by the engine contractor. Special inspections conducted periodically during the durability test programs are essential to preclude part failure and loss of the development engine.

Definitive criteria are needed to judge the success of qualification tests. Problems will likely be discovered during teardown inspection of these engines prior to attainment of the required test durations. Therefore, criteria is needed to establish a course of action in the event problems

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occur, and to define follow-on actions (i.e.; remaining test duration, problem cause, corrective action, operational implications, qualification tests for redesign, etc.).

VERIFICATION GUIDANCE (A.5.9)

The following inspection requirements and success criteria should be transferred, verbatim, into the specification paragraph:

> "The engine contractor will specify inspection procedures, in addition to those for the endurance test engines. Inspection requirements will include in-service design inspections developed in accordance with the durability and damage tolerance requirements of this specification. Inspection requirements will also duplicate the expected field maintenance concept for the engine. Special inspections to monitor the status of critical parts will be included.
>
> Each structural problem (failure, cracking, yielding, wear, erosion, etc.) discovered during endurance testing inspections will be analyzed to determine cause, corrective actions, and operational implications, including development schedule and cost impacts.
>
> A Strength and Life Analysis report should be prepared containing an analysis which defines: (1) the lives in equivalent LCF cycles for all appropriate parts, (2) the LCF duty cycle for the individual component tests and any full-scale engine testing, (3) the cool-down time between cycles, and (4) the total number of cycles to demonstrate the equivalent of the LCF life requirements. The LCF lives of all the appropriate parts and the mission hours equivalency for the selected features in a given test mode should also be defined. The predicted burst speeds of critical rotating disc components should be specified.
>
> The report should identify the allowable size of cracks which would be cause for failure for parts subject to fatigue (thermal or mechanical). The contractor should provide a specific description of the expected failure mode of each part listed.
>
> The Strength and Life Analysis report should define all variables (e.g.; material properties; manufacturing and assembly processes and tolerances; and operating temperatures, pressures, stresses, etc.) that significantly affect the durability and life of the engine. The report should define the sensitivity of parts to variations in these variables and other design assumptions. The limits to variation of these variables should be specified. The report should define how design, processing, manufacturing, assembly, quality control, testing, etc., will be used to ensure all engine durability- and life-affecting variables fall within these limits.
>
> The report should also contain an analysis to establish the accelerated mission-oriented engine duty cycle and the resultant equivalencies for selected features for the AMT/endurance test. These engine duty cycles should be constructed such that the AMT/endurance test produces hot parts damage equivalent to at least the hot parts life of A.4.3. The accelerated engine duty cycles should be derived from the mission cycles of A.4.4. A composite accelerated duty cycle (or cycles) may be an acceptable alternative to individual AMT/endurance test duty cycles. Consistent with the engine duty cycle of *TBD*, a random sequence schedule for the mission duty cycles should also be established with the approval of the Using Service.
>
> Both a pictorial and tabular definition of the accelerated engine duty cycles should be included in the report and in the model specification. There should also be a detailed description of the methodology used to develop the duty cycles, which lists the test acceleration factors and shows appropriate damage factors.

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A fixed level of inlet air temperature and pressure should be established for each AMT/endurance test duty cycle. The inlet air temperature and pressure level determined for each duty cycle should produce the damaging effects on the engine consistent with the actual Mach number and altitude variation defined for each mission and the ambient temperature distribution. The strength and life analysis should also provide the methodology and data utilized to define the engine parts lives (B0.1, B1.0, B10, and B50). Revised or updated reports should include actual data utilized from the development program such as NDI data, etc.

Stress analysis should include such items as engine cases, discs, vanes, blades, mounts, combustion liners, bearing supports, gears, brackets, and tubing.

A strength and life analysis should be performed and a report submitted prior to completion of Initial Flight Release. The report should be updated by change pages by completion of Full Flight Release, Initial Service Release, and Operational Capability Release. A revised report should be issued after completion of Operational Capability Release."

If desired, this paragraph should include a requirement to evaluate problems or failures that occur during endurance testing (i.e., loss of mission capability or requiring maintenance actions) relative to contract/specification reliability and maintainability goals and to use test results to validate the Life Cycle Cost models.

VERIFICATION LESSONS LEARNED (A.5.9)

Complete failure of test engines has occurred in legacy development programs due to undetected damage growth. Many of these failures could have been prevented by use of detail inspection requirements (methods and intervals) developed through the durability and damage tolerance control plans and experience obtained from earlier development testing (e.g., previously-observed deterioration and distress).

A.4.9.1 Low cycle fatigue (LCF) life.

Engine parts will have a minimum LCF life, which is at least equivalent to the design service life of 4.3.

REQUIREMENT RATIONALE (A.4.9.1)

Low cycle fatigue is one of the most severe and costly problems encountered in service. This requirement is an attempt to reduce or eliminate LCF failures.

REQUIREMENT GUIDANCE (A.4.9.1)

Low cycle fatigue is the damage generated in a material by cyclic load reversals, which cause repeated plastic deformation. Rotating parts are subjected to mechanical and thermal stresses during engine operation. These stresses are represented as cyclic loading on engine parts. A full LCF cycle occurs each time the engine is started, run to Maximum power, then shutdown. Partial LCF cycles occur during throttle changes. The severity depends upon the degree of change in engine speed and temperature that accompanies the change. Miner's rule may be used for the summation of LCF cycles to determine the cumulative damage. Cumulative damage from LCF cycles will, eventually, cause a crack to initiate.

Prediction of LCF life is not an exact science and need be determined on a statistical basis. Traditionally, the minimum LCF life of a part is defined as the B0.1 life to crack initiation. This means that 1 in 1000 parts will develop a LCF-initiated crack during the minimum required life of that part.

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Low cycle fatigue design procedures need be used to ensure generalized cracking will not occur before the required design service life is attained. Low cycle fatigue design and analysis procedures, if properly applied, can generate a component configuration that will experience minimal cracking by proper material selection and control of stress levels, detail design to eliminate or optimize stress concentrations, and control of component surface finish and correlation with LCF material allowables. Special attention should be placed during detail structural design to minimize stress concentrations. Those stress concentrations present in the design should be optimized to produce the largest cyclic life possible. Other factors which affect LCF life are: material properties, subsurface material flaws, surface flaws, operating temperatures, and part finishes.

REQUIREMENT LESSONS LEARNED (A.4.9.1)

Most LCF problems did not surface until a number of years after qualification and acceptance by the Using Service.

A.5.9.1 Low cycle fatigue (LCF) life.

The requirement of 4.9.1 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.9.1)

The LCF lives need be evaluated to preclude the occurrence of part failures.

VERIFICATION GUIDANCE (A.5.9.1)

Low cycle fatigue analyses and testing need be accomplished. Testing should consist of component and AMT testing. The component test should be spin tested in a heated spin pit, as a minimum. Thermal gradients should be applied, if possible. The spin testing of rotors should include dummy blades. The testing should be continued until crack initiation or five times (5 X) the predicted LCF life via analysis, whichever comes first. The following time-at-temperature durability test will be conducted for engines which do not use the AMT of A.5.9.1.1 to evaluate LCF:

For ISR and OCR, the length of the LCF engine test will be equivalent to $^1/_2$ of the cold section life, or $^1/_2$ of LCF life, or 4000 cycles, whichever is longer. For IFR and FFR, the LCF test time needs to be at least twice the proposed/actual flight test time.

The LCF test cycle will be in accordance with table XVIII and the following:

a. inlet test conditions corresponding to tables XVIII and XIX condition(s) selected by Using Service;

b. power settings run to initial shaft power levels from table XVIII condition(s) selected by Using Service;

c. engine controls operating with maximum acceleration/fuel flow schedules and transient load change commands of ½-second or less;

d. output shaft speed operating at maximum operating values from tables XVIII and XIX condition(s) at high power settings decreasing to the minimum self-sustaining speed at idle; and

e. with customer bleed and power extraction.

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TABLE XVIII. LCF engine test cycle turboprop/turboshaft engines.

TIME (min)	EVENT
0.5	Start Engine
2.0	Run at Idle
0.1	Accel to Maximum Power
2.5	Run at Maximum Power
0.1	Decel to Idle
3.0	Run to Idle
0.1	Accel to Max Continuous
2.5	Run to Max Continuous
0.1	Decel to Idle
2.0	Run to Idle
2.1	Shutdown and Cool-Down
15.0	TOTAL

Notes:

1. *Transient power commands are in ½-second or less.*
2. *Power settings are initial rating values of output thrust or power, or specified by Using Service.*
3. *Engine control and fuel schedules are set for maximum acceleration.*
4. *Output shaft speeds are rated values at high power settings and minimum self-sustaining speed at idle.*
5. *with customer bleed and power extractions*

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TABLE XIX. Performance at altitude conditions[4].

Power setting[1]	Alt. (ft)	Amb. Temp.	Mach Num.	Minimum Net Thrust N (lbf)		Maximum SFC (lb/hr/lb)		Maximum Engine Rotor Speed (rpm)		Maximum Gas Temp.[2] (°C [°F])		Maximum Measured Temp[3] (°C [°F])		Total Allow (lb/sec) ± __%		Cust. Bld. Air	Cust. Pwr. Ext.
				New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]		

NOTES:
[1] *Power settings will be as required by the Using Service to cover the operating envelope and to be compatible with mission requirements.*
[2] *Defined at the first-stage turbine rotor inlet location (contractor will parenthetically insert, in column heading, the station designation).*
[3] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[4] *The engine performance values shown will be from the engine computer program of JSSG-2007, 3.2.1.1.*
[5] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

TABLE XX. Turboprop/turboshaft performance at altitude conditions[4] with ____ nozzle.

Power Setting[1]	Alt. (ft)	Amb. Temp.	Mach Num.	Maximum Gas Gen. Speed (rpm)		Maximum Gas Temp[2] (°C [°F])		Maximum Meas. Temp[3] (°C [°F])		Total Airflow (lb/sec) ± ___%		Maximum SFC (lb/hr/kw)		Minimum Shaft Power kw (shp)		Output Torque N-m (lb-ft)		Output Shaft Speed (rpm)		Resid. Jet Thrust N (lbf)		Cust. Bld. Air	Cust. Pwr. Ext.
				New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]	New	Det[5]	New		New	Det[5]	New	Det[5]		

NOTES:
[1] *Power settings will be as required by the Using Service to cover the operating envelope and to be compatible with mission requirements.*
[2] *Defined at the first-stage rotor inlet location (contractor will parenthetically insert, in column heading, the station designation).*
[3] *Contractor will parenthetically insert, in column heading, the measurement plane station designation.*
[4] *The engine performance values shown will be from the engine computer program of JSSG-2007, 3.2.1.1.*
[5] *Deteriorated engine performance is specified for _____ TACs (or hours) of use.*

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Background:

Low cycle fatigue engine testing should be required for engines not tested with AMT duty cycles. These include mainly turboprop and turboshaft engines using the time-at-temperature durability tests from A.5.9.1. For helicopter engines, the LCF engine test is considered more rigorous and demanding than missionized schedules or schedules which simulate partial cycles. Low cycle fatigue engine tests add thermal cycling and fatigue effects and have uncovered problems not found in endurance test runs. The LCF test time should be for $^1/_4$ of the engine LCF design life or $^1/_4$ of cold section design life, whichever is longer.

A Strength and Life Analysis should be prepared and a report should be submitted to the Using Service for approval.

A minimum of three sets of fracture-, safety-, mission-, and durability-critical engine parts, identical to the parts list and configuration of the IFR milestone engine should be subjected to LCF testing as specified below to verify the LCF requirements of A.4.9.1. Low cycle fatigue testing should be conducted in accordance with the requirements of "zero failure verification testing", as referenced in AFWAL-TR-83-2079. This method outlines the number of units to be tested and the amount of time to be accumulated on each unit without failure to verify the life. The underlying failure distribution should be assumed to be Weibull. A confidence level of 90 percent should be assumed. The test plan should meet all the requirements of "zero failure verification testing" to verify the minimum LCF lives. These components should be continued in spin testing until crack initiation has occurred.

The LCF testing in the subparagraphs below should constitute the full effect of strain generated by centrifugal, pressure, and aerodynamic forces as well as thermally-generated strains. Dwell times at thrust settings should be sufficient to accomplish stabilization of strains equivalent to those encountered in a service mission cycle. The actual number of cycles, duty times, and length of cool-down time should be based upon the Strength and Life Analysis Report.

a. One set of fracture-, safety-, mission-, and durability-critical engine parts should be subjected to official full-scale engine AMT/endurance testing which produces LCF damage equivalent to at least one-half the cold parts lives specified in A.4.3.2. These same parts should then be subjected to further testing, either by individual component tests (spin pit) until the minimum LCF lives required by A.4.9.1 are evaluated, or by continued testing in an engine.

b. Two sets of fracture-, safety-, mission-, and durability-critical engine parts should be subjected to LCF testing until the minimum LCF lives required by A.4.9.1 are evaluated. The LCF testing should be achieved in any manner appropriate to the full-scale development program (e.g.; full-scale engine tests, individual component tests, or combinations thereof). Component testing of these selected parts, except for the combustor, should be performed either with high temperature and loads appropriate for engine maneuver load conditions simulation, or with loads adjusted for material properties at the test temperature. The combustor should be tested only at high temperature conditions.

All repairs and parts replacement should be recorded and reported. Test substantiation (i.e., full-scale engine testing) of critical parts may require other parts to receive damage greater than their LCF life. Part replacement or repair may then be accomplished to continue the test. The LCF test should be considered successful and the minimum lives of A.4.9.1 verified, if no units fail within the prescribed test time. Failure is defined as generation of a crack size of A.4.8.2. Test times are derived in accordance with AFWAL-TR-83-2079.

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The test time for a high pressure turbine (HPT) disk is determined in the following example:

ASSUMPTIONS:

a. The failure distribution is Weibull.

b. The shape parameter (ß) is 3.0.

c. The confidence level is 90 percent.

The Weibull distribution can be expressed as:

The reliability goal for the HPT disk is 99.9 percent at 4000 LCF cycles.

$$R(4000) = 0.999$$

This is equivalent to the goal of having a characteristic life of 39,990 LCF cycles.

$$\text{Characteristic Life (eta)} = 39{,}990 \text{ LCF cycles}$$

The characteristic life multiplier can be determined from table 5.1 in AFWAL-TR-83-2079. Three sets of hardware will be tested. Therefore, the characteristic life multiplier for a sample size of 3 and ß = 3.0 is 0.916.

$$(0.916)39{,}990 = 36{,}630$$

Hence, three units need be tested for 36,630 LCF cycles without a failure to ensure the reliability goal of 99.9 percent at 4000 LCF cycles is met. Failure is defined as generation of a crack size per 4.8.2.

VERIFICATION LESSONS LEARNED (A.5.9.1)

Most LCF tests on the complete engine were not conducted for the LCF life of the parts. This would be a very long and expensive program, even if accelerated. The lack of statistical significance of one data point (a single engine) justifies running several identical parts in spin pits. An increase in sample size provides an increase in the statistical significance. Disks, which are expensive to replace, have been tested in a spin pit with as many as five samples. Spin pit testing should be performed in heated spin pits and continued until crack initiation. Spin pit testing has certain disadvantages. Specifically, it only simulates the centrifugal loading. It does not simulate the vibratory, aero-elastic, thermal, or loads from adjacent disks. Of those loads, simulation of the thermal stresses imposed on disks is the most difficult.

A.5.9.1.1 Accelerated mission test (AMT).

An accelerated mission test (AMT) will be performed on the initial flight release (IFR) engine configuration. The test run schedule will simulate the design duty cycle of 5.3. The minimum test duration will be two times (2 X) the initial flight test usage. This test will be completed prior to first flight.

VERIFICATION RATIONALE (A.5.9.1.1)

Accelerated mission test is required prior to first flight to provide assurance the engine can safely accomplish the flight test program.

VERIFICATION GUIDANCE (A.5.9.1.1)

The minimum duration for AMT prior to first flight will be two times (2 X) the usage planned for the initial flight test program (2 X flight test duration) on a single engine. Accelerated mission test will be scheduled so that the duration of testing simulated by ground test is accelerated by a minimum factor of two (2) beyond any flight test engine. Additional guidance regarding derivation of the AMT duty cycle is given in section A.5.9.1.2 of this handbook.

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VERIFICATION LESSONS LEARNED (A.5.9.1.1)

None.

A.5.9.1.2 Full-scale development engine.

An AMT will be performed on the full-scale development engine configuration. The test schedule will simulate the design duty cycle of 5.3. The minimum test durations will be one-half the design service life at full flight release (FFR) and one times (1 X) the design service life at initial service release (ISR).

VERIFICATION RATIONALE (A.5.9.1.2)

Extended duration AMT is required to substantiate the durability/economic life of the engine. Additionally important is that the results of extended tests be available at the time the Air Force is evaluating the suitability of the engine for full production commitment.

VERIFICATION GUIDANCE (A.5.9.1.2)

This AMT should be scheduled so that a minimum of one-half times ($^1/_2$ X) the design service life is accomplished at FFR and one times (1 X) the design service life is accomplished at ISR on a single engine. The results of this AMT will be used to determine if the durability requirements of section 4 of the document have been met.

A review of the ENSIP Military Standard and appended handbook (MIL-STD-1783) with the AIA's Project PC 338-2A Committee in September 1982 did not garner full endorsement of the above guidelines on AMT duration. The main point of difference was AIA concern with the amount (duration) of testing required at each development milestone. Although USAF experience strongly indicates design service life requirements should be demonstrated prior to volume commitments to avoid production changes and retrofits, the AIA has endorsed test durations significantly lower than the above guidelines.

Options to accelerate the AMT should be used when possible. For example, an option is to conduct one-half design service lifetime (one hot parts life) and thereafter change the test cycle to a mission-related LCF test. For the LCF test cycle, dwell time at intermediate power is condensed to the minimum time required to simulate thermal gradient. The dwell times for the LCF test cycle should be based on analysis and thermal survey data. The LCF test cycle can allow a greater acceleration of simulated service usage and reduce test costs.

Additional guidance regarding derivation of the AMT duty cycle is provided as follows:

Accelerated mission test spectrum derivation. Accelerated Mission Tests will be derived initially based on design mission profiles and mix and continually updated based on real usage from the usage program in Task V (lead the fleet with flight recorder data). When it is necessary, altitude and/or ram testing will be included.

Prior to the start of test, the engine control fuel schedule will be adjusted to obtain starts, restarts, and accelerations, and to provide starting and acceleration temperatures all at or above rated or maximum values, as applicable. Deceleration fuel schedules will be preset to provide maximum thermal shock. The customer air bleed will be set with a fixed orifice to provide maximum permissible bleed air flow. The accessory and customer power takeoff pads will be loaded to provide max continuous loads plus transients to maximum allowable loads.

All repairs and parts replacement will be recorded and reported. The actual number of cycles, duty times, and length of cool-down time will be based upon a study that uses thermal survey data and will be that required to obtain at least the same failure and stress rupture damage on the test engine as one operational lifetime predicted in the strength and life analysis. This test substantiation of one life may require certain parts to receive more than one lifetime of damage.

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If this damage on those parts exceeds the parts' design life, part replacement or repair may be accomplished in order to continue the test.

The accelerated mission-related test spectrum will be derived from the following:

a. design duty cycle

b. results of thermal survey

c. latest usage information

d. damage tolerance and durability analysis

e. vibration analysis and strain survey.

The test spectrum derivation will consider at least the following:

a. flight-by-flight mission usage with blocks of ram, alt, and high mach conditions included, as appropriate

b. low cycle fatigue mechanism

number of 0-max/mil-0 throttle excursions

number of idle-max/mil-0 throttle excursions

c. stress rupture and creep failure mechanism - time at and above military power

d. incremental running to interrogate high cycle fatigue failure modes: 10^6 cycles on a one per rev basis

sustained power levels that occur in operation, but not included in normal AMT (idle, cruise, etc.)

known critical speeds (shaft and blades)

e. mission mix

f. Mach Number and altitude, percent time spent at each point in flight envelope

g. hold times and sequence of major throttle settings

h. number of augmentor lights and time in augmentation

i. ground run time and profile; i.e.; trim run, test cell, etc.

j. gearbox Hp extraction and bleed

k. field trim procedures and frequency

l. oil temperature and fuel temperature

m. heated inlet conditions representative of the design duty cycles

n. truncation of both small throttle settings and extended time below military power.

The burner pattern factor and radial profile will be established for each "AMT" test engine.

Calibration. The engine including temperature sensing systems and all controls will be checked on the bench and in the engine prior to test to ensure tolerance limits are met. The engine calibration procedures will be sufficient to establish the performance characteristics through thrust measurements of the complete engine. Calibrations will be made at ambient conditions initially with no customer bleed and no accessory power extraction other than that required for continuous engine operation. Calibrations will also be made with heated inlet conditions which simulate operational use. Calibration data will be sufficient to establish compliance data with the specification requirements for performance ratings, thrust transient times, and starting.

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Recalibration checks at intervals during the test will be made. Extensive recalibration at the end of the test will be conducted to determine performance retention (thrust and specific fuel consumption [SFC]) and control temperature shift at intermediate power for use as an indication of anticipated deterioration rate, and to compare with overall pretest calibration values for the engine, temperature sensing systems, and all controls.

Disassembly and inspection. The engine completing and AMTs will be disassembled in accordance with the procedures contained in the pretest plan. These engine parts will be given a "dirty inspection" for evidence of leakage, oil coking, unusual heat patterns, and abnormal conditions. The engine parts will then be cleaned and a "clean inspection" will then be performed. Engine part measurements will be taken as necessary to inspect for excessive wear and distortion. These measurements will be compared with the engine manufacturer's drawing dimensions and tolerances or with similar measurements made prior to the test. During the "clean inspection" an examination and condition assessment will be conducted. The Procuring Service will be provided all results of nondestructive tests and recommendations for modification or redesign of deficient parts. The Procuring Service will be notified of the inspection commencement date prior to each inspection. The following data will be made available to the Procuring Service during both inspections:

a. Inspection forms filled-out by the engine manufacturer listing all findings

b. Tabulation of all parts found deficient

c. Detailed configuration list of the component or system tested

d. Test logs and list of test events

e. Spectrometric oil analysis report.

As a result of the inspection, parts will be categorized as follow:

a. No repair required

b. Repair required before further use

c. Outside repair limits.

Final approval of the pretest plan will be subject to approval of the Procuring Activity.

VERIFICATION LESSONS LEARNED (A.5.9.1.2)

None.

A.5.9.1.3 Production-tooled engine.

Accelerated Mission Test will be performed on a production-tooled engine configuration. The test schedule will simulate the design duty cycle of 5.3. The minimum test duration will be one times (1 X) the design service life at operational capability release (OCR). Accelerated Mission Test of any proposed design changes will be conducted to a duration of one times (1 X) the design service life at OCR.

VERIFICATION RATIONALE (A.5.9.1.3)

Extended duration AMT of the production-tooled engine configuration including proposed design changes is required to substantiate the durability/economic life of the engine. Additionally important is that the results of extended tests be available prior to initiation of high production rate.

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VERIFICATION GUIDANCE (A.5.9.1.3)

The minimum test duration will be one times (1 X) the design service life at OCR. Accelerated Mission Test of any proposed design changes will be conducted to a duration of one times (1 X) the design service life at OCR. Additional guidance regarding derivation of the AMT test schedule is given in section A.5.9.1.2 of this handbook.

VERIFICATION LESSONS LEARNED (A.5.9.1.3)

None.

A.5.9.1.4 Production-tooled engine configuration.

Accelerated Mission Test will be performed on a production-tooled engine configuration. The test schedule will simulate a service duty cycle that is derived from operational usage data. The minimum test duration will be one times (1 X) the design service life.

VERIFICATION RATIONALE (A.5.9.1.4)

Accelerated Mission Test conducted to a duty cycle based on operational data is required. The results obtained from this test will be used to update the structural maintenance plan, as required. This test is not intended as verification that the durability requirements of section 4 of this document have been met, but rather to define the expected operational life based on measured operational usage of the engine.

VERIFICATION GUIDANCE (A.5.9.1.4)

This AMT is not intended to be a contractual requirement to be completed prior to operational capability release, but to be incorporated as a requirement for subsequent Component Improvement Program (CIP) effort. The minimum test duration will be one times (1 X) the design service life. The maximum test duration will be two times (2 X) the design service life. Additional guidance regarding derivation of the AMT test schedule is given in section A.5.9.1.2 of this handbook.

VERIFICATION LESSONS LEARNED (A.5.9.1.4)

None.

A.5.9.1.5 Inspections.

Major inspection programs will be conducted as an integral part of the AMT programs.

VERIFICATION RATIONALE (A.5.9.1.5)

Special inspections conducted periodically during the test programs are essential to preclude component failure and loss of the development engine. Completion of development milestones rests on maintaining integrity of the development engines.

VERIFICATION GUIDANCE (A.5.9.1.5)

The contractor should propose an inspection policy for the AMT engines. Inspection requirements should include in-service design inspections developed in accordance with the durability and damage tolerance requirements of this handbook. Inspection requirements should also duplicate the expected field maintenance concept for the engine. Special inspections to monitor the status of critical components should be included. Teardown inspection per A.5.9.1.2 should be identified at the completion of test milestones to support the interpretation and evaluation task of A.5.9.1.5.

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VERIFICATION LESSONS LEARNED (A.5.9.1.5)

Complete failure of test engines has occurred in legacy development programs due to undetected damage growth. Many of these failures could have been prevented by use of detail inspection requirements (methods and intervals) developed through the durability- and damage-tolerance control plans and experience obtained from earlier development testing (e.g., previously-observed deterioration and distress).

A.5.9.1.6 Interpretation and evaluation of test results.

Each structural problem, such as failure, cracking, yielding, wear, and erosion, discovered during inspection of the AMT engines will be analyzed to determine cause, corrective action, and operational implications relative to meeting the design requirements contained in this handbook. Specific requirements will be identified.

VERIFICATION RATIONALE (A.5.9.1.6)

Definitive criteria are needed to judge the success of qualification tests, especially the AMTs of A.5.9.1.1. Problems will likely be discovered during teardown inspection of these engines prior to attainment of the required test durations. Therefore, criteria is needed to establish course of action in the event problems occur and to define follow-on actions (i.e.; remaining test duration, problem cause, corrective action, operational implications, qualification tests for redesign, etc.).

VERIFICATION GUIDANCE (A.5.9.1.6)

The contractor will identify the requirement in A.5.9.1.6 to perform interpretation and evaluation of test results. The requirement should identify the policy for remaining test duration in the event failures or problems occur prior to attainment of the required test duration. In general, component failure or other problems that occur prior to attainment of the required test duration will not require retest of the full engine to the full duration but will require additional testing to complete the remainder of the duration that existed at the time the failure or problem occurred. However, redesign and/or retest for the full duration will be required for those components that experience failure or problems that in service would endanger the pilot or aircraft and for those components that have to be replaced due to secondary damage (i.e., over-temperature exposure or domestic object damage), although qualification may not be tied to the ISR and OCR milestones. The approach for qualifying these components (i.e.; redesign, test vehicle, and schedule relative to the development milestones) will be subject to negotiation between the Procuring Agency and the contractor. It is anticipated that the AMT of A.5.9.1.3 will be utilized to qualify redesigns required as a result of problems uncovered by earlier test, components not qualified for the full test duration by earlier test, and other design changes that occur due to cost, manufacturing considerations, etc.

Additionally, A.5.9.1.6 should identify the requirement to evaluate problems or failures that occur during AMT (i.e., loss of mission capability or requiring maintenance actions) relative to contract/specification reliability and maintainability goals and to use test results to validate the LCC models.

The A.5.9.1.6 requirement should identify the need to establish problem cause, corrective action, operational implications, additional redesign, and test requirements, together with schedule and costs.

It is recommended that A.5.9.1.6 contain the following specific requirements:

The AMT engine test will be considered to be satisfactorily completed when the engine has: (a) completed the test duration per the pretest plan, (b) the test engine and components are operating satisfactorily at the end of the test, (c) recalibrations reveal performance retention to be within limits specified in 4.11 and in the Primary Specification, (d) not experiencing any

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catastrophic failures, (e) not experiencing in-flight shutdown events, and (f) assessment of failures and impending failures and establishment of corrective actions.

In the event of catastrophic failures of nondiscretionary in-flight shutdowns in the single engine configuration, penalty running requirements will be established by the Procuring Service after a review of the circumstances. Impending failures will be used to set inspection intervals and evidence of impending failure will be included as maintenance and reliability factors for verification of compliance.

VERIFICATION LESSONS LEARNED (A.5.9.1.6)

None.

A.4.9.2 Life design margin.

A life margin will be applied during design of engine components.

REQUIREMENT RATIONALE (A.4.9.2)

Life design margin is needed in early development due to uncertainties in usage, environmental conditions, and quality of the finished part.

REQUIREMENT GUIDANCE (A.4.9.2)

The recommended margin for LCF life design is 100 percent, based on minimum material properties. The weight impact of having a LCF margin can be small and, in many cases, LCF of parts will meet the 100-percent margin based on design stress levels set by other structural requirements such as high cycle fatigue, creep and stress rupture, burst strength, and stiffness requirements. Optimization of stress concentrations can increase LCF life by an order of magnitude and more than offset any weight increases incurred by imposition of a LCF margin. Also, the inverse exponential shape of LCF design curves often enables considerable increase in LCF life by only a modest decrease in nominal stress.

It is not so clear that a significant life margin can be achieved on engine hot parts relative to stress rupture capability. The goal is that a design life margin be achieved on hot parts wherever possible to protect against environmental uncertainties (internal temperatures, deterioration, etc.) or to increase the usable life beyond the minimum specification requirements. The contractor will recommend other design margins to apply to engine hot parts based on consideration of other pertinent failure modes and knowledge of the internal environment.

The LCF design margin may be reduced if measured data on usage of a similar system is available or if other conservative measures are to be utilized in the design. Trade studies should be identified in the durability control plan to identify cost (weight, performance, life cycle cost, etc.) as a function of durability/economic life to establish realistic life design margins.

REQUIREMENT LESSONS LEARNED (A.4.9.2)

None.

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A.5.9.2 Life design margin.

Attainment of the life design margin will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.9.2)

Life design margin need be evaluated to ensure compliance with the requirement.

VERIFICATION GUIDANCE (A.5.9.2)

See A.5.9.1.

VERIFICATION LESSONS LEARNED (A.5.9.2)

See A.5.9.1.

A.4.9.3 Corrosion prevention and control.

The engine will operate satisfactorily without detrimental material degradation in the environmental conditions specified in 4.5 through 4.5.3 for the design service life.

REQUIREMENT RATIONALE (A.4.9.3)

Corrosion prevention and control is important to avoid material degradation that will cause an uneconomical maintenance burden and affect operational readiness.

REQUIREMENT GUIDANCE (A.4.9.3)

Repair or replacement of corrosion prevention systems is permitted where engine experience shows that the protective treatments become ineffective prior to attaining the required design service life under realistic environments. However, a minimum period of unrepaired service usage will be specified and this period will be equal to or greater than the depot or base level inspection interval of A.4.8.4.

REQUIREMENT LESSONS LEARNED (A.4.9.3)

None.

A.5.9.3 Corrosion prevention and control.

The corrosion resistance of the engine materials, processes, and protection systems will be evaluated as follows: _____.

VERIFICATION RATIONALE (A.5.9.3)

A plan is needed to identify protective treatments and evaluation methods based on experience obtained with previous engine systems.

VERIFICATION GUIDANCE (A.5.9.3)

A corrosion prevention and control plan will be prepared. The contractor will identify the protective treatments to be used in the engine. Prior experience with use of these treatments will be provided as well as identification of tests to qualify proposed new treatments. The contractor will identify problems that have occurred on past engine systems related to corrosion or other environmentally-induced material degradation and will provide justification that the proposed engine configuration will not be susceptible to these problems.

VERIFICATION LESSONS LEARNED (A.5.9.3)

None.

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A.4.10 Strength.

The engine will meet all the requirements of the specification during and after exposure to limit loads, singly and in combination. The engine will not experience catastrophic failure when subjected to ultimate loads, singly and in combination. In addition, the engine will meet the following strength criteria.

REQUIREMENT RATIONALE (A.4.10)

Limit and ultimate loading need be addressed since the engine may be exposed to limit and ultimate loading conditions during operation.

REQUIREMENT GUIDANCE (A.4.10)

The limit load conditions of A.4.3 will occur over the life of an engine system. Therefore, the structure need be able to react to these loads without incurrence of detrimental permanent deformation or degraded performance so that operational capability is maintained. Also, stresses greater than design limit load values can occur as a result of inadvertent operation of the engine and/or weapon systems. Stresses greater than material allowables can occur due to variation in material properties (i.e., castings). Therefore, it is required that the engine structure have a margin of strength to withstand without failure externally-applied forces that exceed the limit load conditions of A.4.3.

The engine should incorporate fail-safe design objectives to eliminate catastrophic failure including, but not limited to, the following considerations:

a. Compressor and turbine disks will be protected by having blades fail first under overspeed or overtemperature malfunctions.

b. A main rotor shaft bearing or lubrication system failure will not cause parting or decoupling of the shaft(s).

c. In the event of shaft decoupling, the disks will be designed such that the burst speed will be at least 5 percent greater than the maximum predicted free rotor overspeed or the turbine blading will contact the turbine vanes to minimize a turbine overspeed, or an overspeed trip system will be installed to control turbine overspeed.

d. In the event of a rotor bearing failure, the structures supporting the rotating masses will be designed to minimize the probability of gross misalignment of the engine rotating parts.

e. All areas of the rotor that could puddle oil will have appropriate drains.

Specific guidance to establish factors of safety and strength requirements is contained in A.4.10.1 through A.4.10.10.

REQUIREMENT LESSONS LEARNED (A.4.10)

None.

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A.5.10 Strength.

The requirements of 4.10 will be evaluated by structural analysis and part, component, and full-scale engine tests.

VERIFICATION RATIONALE (A.5.10)

Structural analyses and tests are required to evaluate the engine and its components' ability to meet strength requirements.

VERIFICATION GUIDANCE (A.5.10)

Specific guidance on structural analysis requirements for A.4.10 and A.4.10.1 through A.4.10.10 are consolidated within this guidance section. The contractor will describe the extent of structural analysis to be performed. Design analysis methods to be used to demonstrate ability to meet operational requirements of A.4.10.2 through A.4.10.10 should be identified. The schedule for the analysis should be identified and should meet the milestone guidance provided in table II.

Structural modeling techniques to be used should be described for each major class of components (e.g.; static structures, disks, shafts, airfoils). Detailed modeling for these structures is required. It is recommended that the analysis approach employ direct utilization of the thermal heat transfer model required to establish stress as a function of flight conditions. Final analyses should utilize finite element breakups or comparably-precise methods to establish stress concentrations and gradients at structural discontinuities (bolt holes, rim slots and posts, radii, blade shrouds and dovetails, etc.).

The contractor should describe the extent of component strength tests performed. Specific tests to be used to demonstrate the ability to meet operational requirements should be identified. The schedule for these tests should also be identified.

Strain gauges should be utilized during component strength tests to verify analysis methods relative to nominal stresses and peak surface stresses at concentration details. It is recommended that strain gauges be utilized on each component strength test of static structure. Strain gauge data on rotating structures should be obtained from core and full-scale engine testing.

Older technologies, which are still valid for placement of strain gauges and the understanding of the stress field in complex areas, are stress coat and photoelastic modeling. Stress coat techniques can be used to help establish locations for strain gauges. Photoelastic modeling and test is suggested for a better understanding of maximum stresses and gradients at complicated structural details (e.g., blade dovetails and rim slots).

VERIFICATION LESSONS LEARNED (A.5.10)

None.

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A.4.10.1 Factors of safety.

Factors of safety will be applied to design usage induced loads to establish limit and ultimate conditions.

REQUIREMENT RATIONALE (A.4.10.1)

Sufficient factors of safety (table XXI) need be determined to ensure adequate safety margins exist in designs.

REQUIREMENT GUIDANCE (A.4.10.1)

Table XXI should be used to establish factors of safety.

TABLE XXI. Factors of safety.

LOAD TYPES	LIMIT	ULTIMATE a	ULTIMATE b
Externally-applied Loads	1.0	1.0	1.5
Thermal Loads	1.0	1.0	1.5
Thrust Loads	1.0	1.2	1.0
Internal Pressures	1.5	1.0	2.0
Aircraft Flow Field Loads	1.0	1.0	1.5
Crash Loads	N/A	1.0	1.0

NOTES:

1. *For all castings, a factor of safety of 1.33 will be applied to the limit and ultimate load factors specified above, unless the castings have been fully characterized.*
2. *Two combinations (a & b) will be used to establish ultimate loading conditions required in other paragraphs in this section.*

REQUIREMENT LESSONS LEARNED (A.4.10.1)

Catastrophic failures of cast parts and pressure vessels have occurred due to porosity and poor manufacturing processes. These parts were designed with a 1.5 factor of safety for ultimate load conditions. A more graceful (i.e., non-catastrophic) failure would have occurred if the hardware had been designed with a 2.0 factor of safety.

A.5.10.1 Factors of safety.

The requirements of 4.10.1 will be evaluated by analyses and tests.

VERIFICATION RATIONALE (A.5.10.1)

Factors of safety need be evaluated to ensure adequate safety margin.

VERIFICATION GUIDANCE (A.5.10.1)

Strain gauges and other instrumentation should be used during tests to evaluate analysis methods. It is recommended that tests be conducted progressively to ultimate load conditions.

VERIFICATION LESSONS LEARNED (A.5.10.1)

None.

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A.4.10.2 Blade and disk deflection.

The blades and disks will not contact any static parts of the engine other than seals and shrouds, during all phases of engine operation including surge and stall occurrences. Seals and clearances will remain effective under all internal and external operational loads.

REQUIREMENT RATIONALE (A.4.10.2)

Sufficient rigidity need be provided so the engine can operate to the limit loads and repeated loads that occur within the flight envelope without detrimental damage.

REQUIREMENT GUIDANCE (A.4.10.2)

High thrust bearing loads cause rotors to shift, thereby increasing clearances or causing detrimental damage to static or rotating hardware. This happens more in the turbine where high temperature increases creep and thermal stress. The resultant rub opens the blade clearances or can damage hardware where rubs were not intended to occur.

REQUIREMENT LESSONS LEARNED (A.4.10.2)

Rotor shifts have caused blading to contact vanes and led to subsequent blade failure.

A.5.10.2 Blade and disk deflection.

The requirements of 4.10.2 will be evaluated by the analyses and tests.

VERIFICATION RATIONALE (A.5.10.2)

Structural analyses and tests are required to evaluate that blade and disk deflection does not result in contact with any static parts of the engine other than seals and shrouds. Blade and disk rigidity need be evaluated to ensure flight safety against blade failures and titanium fires.

VERIFICATION GUIDANCE (A.5.10.2)

The test should be conducted in accordance with A.4.10.

VERIFICATION LESSONS LEARNED (A.5.10.2)

X-ray photography has been used to determine that seals and clearances are effective under all operational loads.

A.4.10.3 Containment.

Uncontained failures will not cause fire or catastrophic damage to engine external systems or aircraft systems, or injury to personnel.

REQUIREMENT RATIONALE (A.4.10.3)

Uncontained failure of rotating components can cause extensive damage to external engine components, lines, and wiring harnesses. Beyond engine systems, secondary damage can occur to aircraft lines, fuel tanks, and critical systems. Damage is severe enough in many of these cases to cause loss of the aircraft either by loss of functionality of critical systems or by uncontained fire.

REQUIREMENT GUIDANCE (A.4.10.3)

Containment is an interface requirement and should be allocated from the weapon system requirements. The engine will have full engine containment unless there is a split of allocation of this requirement between the engine and the airframe. The full allocation requirement for containment should be to ensure failure of a rotating component poses a minimum catastrophic hazard due to secondary damage, or probability of failure for a component is low enough to be considered impossible. The impact on system weight, performance, cost, and risk need be

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weighed and evaluated per specific program constraints and requirements when possible design solutions are provided.

Failure modes for rotating components need be evaluated to determine if the consequence of failure is uncontained failure and probable loss of the aircraft. In those cases where loss of the aircraft is not remote, components, engine systems, aircraft systems, or some combination needs to be designed to maximize the probability of safe aircraft recovery. Where containment or aircraft shielding is selected, design procedures should be based on past experience that includes engine blade failures and test data that supports/establishes material containment capabilities, and this needs to be incorporated in the Interface Control Document (ICD).

Blade manufacturing anomalies and the variability of FOD/DOD levels and design capabilities significantly limit the ability to design for low probability of failure. As a result, design considerations for fan blade failures should focus on containment versus noncontainment of the entire blade (airfoil, platform, and attachment). Considerations include full containment within the engine case, uncontained but low energy penetration (such that no significant damage occurs to the aircraft), uncontained in combination with aircraft shielding to prevent catastrophic airframe damage, and non-uniform tailored containment (e.g., engine case containment in the upper case only to protect aircraft systems). As part of any solution that allows uncontained failure, external engine and aircraft systems (fuel, hydraulic lines, etc.) need be routed to reduce exposure and limit secondary damage.

The engine should be designed to contain high pressure compressor (HPC) and turbine blades (airfoil, platform, and attachment). These components are subject to more severe environments, material variabilities, complex geometries, and tighter tip clearances than fan blades. These factors combine to reduce the credibility of any attempt to calculate probabilities of failure. In addition, the consequence of case penetration by hot parts followed by hot gas has a greater probability of leading to external engine fires.

Containment of larger (increased energy) rotating components such as disks and seals should be evaluated versus alternative methods listed under fan blades. In addition, designs should be considered that reduce the probability of failure so that the hazard is assumed to be non-existent. Low probability of failures for critical parts may be achieved by selection of appropriate structural integrity criteria and guidance (A.4.6 and A.4.10).

REQUIREMENT LESSONS LEARNED (A.4.10.3)

Experience has led to current designs that allow uncontained fan blades. In association with these designs, critical external engine components are kept off of the fan case to the maximum extent possible. Those components that need to be placed on or in the fan case are grouped in as few circumferential locations as possible, limiting the probability that an uncontained failure will impact a critical component.

The decision to contain fan blades should take into consideration total system survivability. Full containment may maximize the probability of safe recovery of the aircraft following a failure for a multi-engine aircraft, but it might not be the best approach for single-engine aircraft. For example, containment of a fan blade within the engine virtually ensures massive damage to the engine and near complete loss of thrust. This scenario ensures loss of a single-engine aircraft unless a suitable airfield is available very nearby. In contrast, allowing a fan blade to exit the case and be contained by aircraft shielding increases the chance that internal secondary damage will be limited to an extent that allows the engine to continue to operate at a reduced power and permit safe recovery. This has been shown in eight cases of uncontained fan blade failures where single-engine aircraft were able to get home.

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A.5.10.3 Containment.

The requirements of 4.10.3 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.3)

Evaluation of engine and component containment capability is necessary to avoid the occurrence of uncontained failures.

VERIFICATION GUIDANCE (A.5.10.3)

The engine contractor should perform a blade containment analysis which relates the released blade kinetic energy to the energy required for containment. The containment analysis should be provided at the initiation of EMD to provide confidence that the design will contain certain failures. The analysis should be substantiated/correlated with containment tests. Prior failures on identical structures can be discussed in the analysis and used for substantiation of the containment test. Containment tests are necessary since many uncertainties exist with the various containment analysis procedures (e.g.; dynamic considerations of pressure loading versus ballistic loading, effectiveness of containment structure due to varying geometry, material capability forces, etc.).

The tests should be conducted at or above the maximum allowable rotor speeds and maximum operating temperatures. The blades selected for the test (those blades determined to be the most critical) should be modified to fail at a predetermined speed. The test should be considered satisfactorily completed when all damage is contained.

The engine contractor should also provide component containment analyses and/or conduct component testing. The analyses/tests should be conducted to ensure all engine components which utilize rotating parts will contain any rotating part failure at maximum transient speed.

VERIFICATION LESSONS LEARNED (A.5.10.3)

Early use of containment criteria and analysis can avoid a redesign. It is very desirable, for safety of personnel and equipment considerations, to have all blade failures contained within the engine.

A.4.10.4 Blade out.

Subsequent to a single blade failure, with resulting secondary loss of another blade in the same stage at maximum allowable transient speed, the engine will not experience uncontained fire; catastrophic rotor, bearing, support, or mount failures; overspeed conditions; leakage from flammable fluid lines; or loss of ability to shutdown the engine.

REQUIREMENT RATIONALE (A.4.10.4)

The engine need possess adequate structural integrity after blade loss so that a stable time period exists without uncontained catastrophic destruction to allow time for pilot awareness and appropriate action. In addition to the requirement for containment of the failed blade, the secondary failure modes that would result in catastrophic failure need be avoided.

REQUIREMENT GUIDANCE (A.4.10.4)

The following should be used to tailor the specification paragraph:

a. Blade loss loads for conventional blades should be based on the imbalance equivalent to fracture in two blade attachments at the minimum neck section above the outermost retention feature.

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b. Blade loss loads for integrally-bladed rotors should be based on the imbalance equivalent to liberation of two airfoils, including the fillet material down to the rotor rim diameter.

Blade out conditions should also address the possibility of interactive blade/disk vibration modes which result from imbalance or acoustics.

Design for blade failures should include the fan, compressor, and turbine rotors, individually. A single blade failure results in blade out loads equivalent to two blades out due to subsequent secondary damage. Furthermore, adequate damping need be provided so that a single blade failure does not cause engine operation at a critical speed which would cause further failures of other hardware. Blade out loads are needed for proper design of aircraft engine mounts.

REQUIREMENT LESSONS LEARNED (A.4.10.4)

Blade failures have caused uncontained fire and catastrophic rotor failure in compressors that use titanium for blades and cases. An imbalance condition caused by loss of blades can lead to bearing and rotor support failure. Interactive blade/disk vibration modes have resulted in the failure of a blade retainer and subsequent loss of a fan rotor assembly.

A.5.10.4 Blade out.

The requirements of 4.10.4 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.4)

Structural analyses and tests are required to evaluate that blade failure does not cause the engine to experience uncontained fire; catastrophic rotor, bearing, support, or mount failures; overspeed conditions; leakage of flammable fluids; or loss of ability to shut-down the engine.

VERIFICATION GUIDANCE (A.5.10.4)

Evaluation of blade out requirements should include analyses of the fan, compressor, and turbine sections of the engine. Evaluation of the most critical rotors should be accomplished by an engine test. Blade out testing could destroy a complete engine. Hence, it may be permissible to perform part testing instead of full-scale engine testing. This may be done in conjunction with the containment evaluation.

Failure should be assumed to occur at the maximum transient rotor speed (i.e.; the maximum normal operating speed plus adjustments to account for deterioration, control and measurement tolerance, engine-to-engine variations, and Idle-to-Intermediate acceleration overshoots).

VERIFICATION LESSONS LEARNED (A.5.10.4)

Evaluation by engine test during development is very rare. As part of Federal Aviation Administration (FAA) certification, the GEAE-manufactured CF6 was required to demonstrate compliance with blade out requirements via a full-scale engine test.

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A.4.10.5 Overspeed/overtemperature.

The engine will meet all the requirements of the specification during and after overspeed and overtemperature conditions.

REQUIREMENT RATIONALE (A.4.10.5)

These requirements are needed to provide an operational margin for rotor structural integrity while allowing continued use after overspeed and overtemperature events.

REQUIREMENT GUIDANCE (A.4.10.5)

a. engine rotor speeds of __(a)__ percent of the maximum allowable steady-state speed at the maximum allowable turbine temperature or first-stage turbine rotor inlet gas temperature limit (which includes adders for maximum transient temperature and uncertainties in temperature due to sensors and analytical determinations), for five minutes

b. __(b)__ temperature of at least __(c)__ °C (__(c)__ °F) in excess of the maximum allowable temperature or above the first-stage turbine rotor inlet gas temperature limit (which includes adders for maximum transient temperature and uncertainties in temperature due to sensors and analytical determinations), and at maximum allowable steady-state rotor speed for five minutes

c. engine component rotor speeds of __(d)__ percent of the maximum allowable steady-state speed for five minutes

d. engine component fuel, lube, and hydraulic inlet temperatures of __(e)__ °C (__(e)__ °F) above the maximum allowable operating fluid temperature at the maximum allowable steady-state speed for five minutes.

e. The power turbine shaft speed for turboprop/turboshaft engines will not exceed __(f)__ percent of the transient speed limit, or the predicted speed attained following loss of load with the engine at Intermediate power and the power turbine running at the highest rated speed, whichever is greater. The predicted maximum speed will be specified herein.

Requirement “a.” provides for an overspeed condition while operating at normal maximum temperature conditions. Requirement “b.” provides for an overtemperature condition while operating at normal maximum allowable speed conditions. These two requirements are needed to provide an operational margin for engine rotor structural integrity while allowing continued use (should not yield); as opposed to the requirement of A.4.10.6, which is intended to represent the ultimate strength of the material (should not burst).

Requirements “c.” and “d.” are specified to provide an operational margin for component rotor structural integrity while allowing continued use.

The following values should be used:

(a): a value of 115, minimum

(b): insert HPT blade metal, HPT inlet, measured, or gas

(c): a value between 42°C (75°F) and 45°C (81°F)

(d): 115

(e): 25°C (45°F)

(f): a value of 115, minimum.

The temperature description (e.g.; HPT blade metal, HPT inlet, measured, gas, etc.) should be consistent with the performance rating temperature description.

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REQUIREMENT LESSONS LEARNED (A.4.10.5)

Past experience indicates the engine can overspeed or exceed design temperature due to control system malfunctions or other engine operating anomalies.

A.5.10.5 Overspeed/overtemperature.

The requirements of 4.10.5 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.5)

Overspeed/overtemperature of the engine need be evaluated to ensure structural integrity of rotating parts.

VERIFICATION GUIDANCE (A.5.10.5)

An analysis that depicts the overspeed and overtemperature capability of the engine should be provided. Overspeed and overtemperature tests should be conducted to substantiate/correlate the analysis.

For the overspeed test, all rotors should be subjected to engine operation for a stabilized period of at least five minutes duration at (value specified in A.4.10.6) percent of maximum allowable steady-state speed at the engine's maximum allowable temperature. Following the test, parts and assemblies should be within allowable dimensional limits and there should be no evidence of imminent failure. If a cold spin pit is used for hot flow components, speed should be added to compensate for temperature effects (with Using Service approval).

Upon successful completion of the overspeed test, the same engine should be operated at a (HPT blade metal, HPT inlet, measured, gas) temperature of at least (value specified in A.4.10.6) in excess of the maximum allowable temperature and at no less than maximum allowable steady-state speed for five minutes. Following the test, parts and assemblies should be within allowable dimensional limits and there should be no evidence of imminent failure.

For engines with more than one rotor system, the test may be performed separately for each rotor system. Rig tests may be performed with Using Service approval.

VERIFICATION LESSONS LEARNED (A.5.10.5)

None.

A.4.10.6 Disk burst speed.

The minimum loaded disk burst speed of the complete disk assembly will be greater than or equal to the overspeed requirements of 4.10.5.

REQUIREMENT RATIONALE (A.4.10.6)

This requirement ensures adequate margin against the risk of a disk burst in service.

REQUIREMENT GUIDANCE (A.4.10.6)

The minimum loaded disk burst speed of the complete disk assembly should be 115 to 122 percent of the maximum allowable steady-state rotor speed or 5 percent above maximum transient rotor speed, whichever is greater when the disk is subjected to the maximum temperature gradient and maximum material temperature that will occur for that part.

The 122 percent represents a factor of safety of 1.5 (centrifugal stresses vary as the square of speed). The loaded disk burst requirement is necessary since stresses on the disk are obviously greater when it is loaded with blades. The material properties and stress distributions

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are more severe when subjected to the maximum temperature gradient and maximum temperature conditions for the part.

Proposed values for disk burst speeds may provide a compromise between crack growth capability and tensile strength.

Generally, "damage tolerant" materials provide better crack growth capability, but lack high tensile strength (burst) capability.

REQUIREMENT LESSONS LEARNED (A.4.10.6)

The FAA uses 120 percent of maximum allowable steady-state rotor speed. The USN and USAF have allowed minimum disk burst speeds of 115 to 117 percent of maximum allowable steady-state rotor speed. For titanium fan disks, the USAF has required minimum disk burst speeds of 130 percent of maximum allowable steady-state rotor speed.

A.5.10.6 Disk burst speed.

The requirements of 4.10.6 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.6)

The required disk burst speed need be evaluated to prevent the occurrence of a catastrophic failure.

VERIFICATION GUIDANCE (A.5.10.6)

The Strength and Life Analysis of A.4.9.1 should include a detailed evaluation of the operating environment and stress levels seen by each engine disk. The analysis should provide an initial evaluation of the burst capability of each disk. This information should be substantiated/ correlated with disk burst spin pit testing.

Disk burst testing should be conducted to evaluate whether the burst margin requirement of A.4.10.6 can be met with a minimum tensile strength disk (based on the minimum properties specified in A.4.6). Disk burst testing should be conducted on all engine disks. As a minimum, disk burst tests may be conducted on the most limiting rotor (disk with the minimum burst capability) of each module.

Disks should be operated at burst speeds no less than those of A.4.10.6 while exposed to the maximum temperature gradient and maximum material temperature that would occur for that part. Maximum test speed should be sufficient to demonstrate that a minimum tensile strength component (–3 Sigma) can meet the burst margin requirement based on the specific ultimate strength capability of the test component. These conditions should be maintained for a minimum of 30 to 60 seconds. The test should be considered successfully completed if there is no evidence of imminent failure.

Since the blades may actually fail before the disk, substitute blades (dummy blades) may be used in lieu of actual disk blades during the evaluation.

VERIFICATION LESSONS LEARNED (A.5.10.6)

None.

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A.4.10.7 Output shaft torque limits.

The maximum allowable steady-state delivered shaft torque (mechanical) limit for turboprop and turboshaft engines will be at least _(a)_ percent greater than the rating value.

REQUIREMENT RATIONALE (A.4.10.7)

A limit is required to provide a margin of torque to prevent catastrophic component failure.

REQUIREMENT GUIDANCE (A.4.10.7)

The following should be used to tailor the specification paragraph:

(a): 20.

REQUIREMENT LESSONS LEARNED (A.4.10.7)

Past experience indicates that the engine output shaft torque can increase because of a malfunction or other anomalies. Some engine contractors have asked for a deviation. They felt they had a good control of output shaft torque and torque limits could be lower. Other systems (helicopters) with torque limits depend upon pilot action for limit observance.

A.5.10.7 Output shaft torque limits.

The requirements of 4.10.7 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.7)

Evaluations by analysis and tests are needed to ensure the engine will not be degraded by shaft torque operation up to the limit.

VERIFICATION GUIDANCE (A.5.10.7)

Background:

The test can be conducted on the total engine or only on the affected component. The torque should be conducted for a minimum time of 5 minutes.

VERIFICATION LESSONS LEARNED (A.5.10.7)

None.

A.4.10.8 Output shaft speed limits.

The maximum allowable steady-state delivered shaft speed (mechanical) limit for turboprop and turboshaft engines will be at least _(a)_ percent greater than the rating value. The shaft will be able to operate at this speed for at least _(b)_ and function satisfactorily thereafter. Following loss of load, the output shaft speed will not exceed the maximum shaft speed predicted with the engine at Intermediate power and the output shaft running at the maximum attainable rotor speed.

REQUIREMENT RATIONALE (A.4.10.8)

An output shaft speed limit is required to provide a margin of speed to prevent catastrophic failure.

REQUIREMENT GUIDANCE (A.4.10.8)

The following should be used to tailor the specification paragraph:

(a): a value of 15, minimum

(b): a value of 5 minutes, minimum.

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REQUIREMENT LESSONS LEARNED (A.4.10.8)

Past experience indicates the engine output shaft speed can increase because of a malfunction or other anomalies. Engine contractors have asked for a deviation on the speed requirement since they believe the newer control systems will prevent any overspeed.

A.5.10.8 Output shaft speed limits.

The requirements of 4.10.8 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.8)

Evaluations by analyses and tests are needed to prevent catastrophic component failure.

VERIFICATION GUIDANCE (A.5.10.8)

Background:

This test can be conducted in conjunction with the rotor overspeed test. The test should prove structure integrity and parts are not stressed to yield.

VERIFICATION LESSONS LEARNED (A.5.10.8)

The evaluation was conducted by analysis or similarity on some engines in the past.

A.4.10.9 Pressure vessel/case.

All engine cases and pressure-loaded parts and components will withstand the ultimate loading conditions defined in 4.10.1. The cases need to remain intact, although permanent deformation and distress which require repair or replacement is permitted. Engine cases will not fail due to combustion process burning or erosion.

REQUIREMENT RATIONALE (A.4.10.9)

Pressurized vessels, cases, and components require internal and external load safety margins to preclude failure (e.g., burst and hazardous venting conditions). Also, pressurized vessels, cases, and components need be protected from degradation caused by combustion processes or erosion.

REQUIREMENT GUIDANCE (A.4.10.9)

The pressure vessels should be designed to meet the ultimate load capability of two times (2 X) the maximum operating pressure plus one and one-half times (1.5 X) the maneuver loads plus one and one-half times (1.5 X) the thermal loads. The engine should withstand the combined effects of these loads without catastrophic failure.

REQUIREMENT LESSONS LEARNED (A.4.10.9)

Experience has shown that if a case is designed to a factor of safety of 2, problems of rupture, LCF, and burn-through are reduced. One turbofan engine burner case was initially designed close to yield and exhibited problems in service due to LCF and burn-through. After the engine was redesigned to eliminate these problems, the factor of safety on rupture was checked and found to be approximately 2. A factor of safety of 2 was mandated in one engine program. This engine was found during test to be able to tolerate a burn-through without rupture. Another engine program required a factor of safety of 2. Subsequently, that engine-type has experienced several burn-throughs without rupture.

A.5.10.9 Pressure vessel/case.

The requirements of 4.10.9 will be evaluated by analyses and tests.

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VERIFICATION RATIONALE (A.5.10.9)

The engine pressure vessels and gas-pressure-loaded components need to be able to withstand the combined operating ultimate loads without catastrophic failure to ensure that the engine has been designed with appropriate safety margins, and that it can operate satisfactorily for the required design usage and service life.

VERIFICATION GUIDANCE (A.5.10.9)

The engine contractor should provide an analysis of all pressure-loaded parts and components. The analyses should show that all pressure-loaded parts and components can meet the requirements of A.4.10.10 when constructed with minimum-strength materials, as defined in A.4.6. The analyses should be substantiated/correlated with pressure vessel/case testing.

All pressure-loaded parts and components should be tested to at least two times (2 X) the maximum operating pressure in combination with the external ultimate loads based on the external loads encountered during engine operation. These tests should be conducted at the maximum allowable temperature or at a test pressure adjusted to account for the differences between operating and test temperatures.

The above tests are qualification tests to demonstrate that the design meets the strength requirements. It is also recommended that the production acceptance/quality control requirements include proof pressure test of each pressure-loaded component to 1.33 to 1.50 times (1.33 to 1.50 X) the maximum operating load pressure. Proof tests of each article are required to screen detrimental porosity, crack/void, below blueprint tolerances, or other detrimental anomalies that would reduce the life of the component.

VERIFICATION LESSONS LEARNED (A.5.10.9)

None.

A.4.10.10 Pressure balance.

The engine thrust bearings will provide sufficient thrust load to ensure satisfactory bearing operation without skid damage during the design service life.

REQUIREMENT RATIONALE (A.4.10.10)

This loading requirement is necessary to ensure problems due to inadequate engine pressure balance design do not arise. Transient engine loads on output shaft components will be "taken-out" by engine structure.

REQUIREMENT GUIDANCE (A.4.10.10)

The contractor should ensure that his practices for bearing design include requirements for pressure balance to ensure that load and direction are adequate to achieve satisfactory bearing operation. The requirement is not intended to restrict the thrust load to any one direction or even any specific minimum magnitude, but rather to provide a pressure balance system which maintains bearing loads sufficient to ensure adequate bearing life. It is necessary that under any steady-state operating condition a minimum thrust be maintained on the thrust bearing in order to prevent skidding. Standard design practices are to design the pressure balance system to provide the minimum required bearing thrust load at the worst (minimum load) steady-state condition and then check the highest thrust load on the bearing by adding the change in engine-generated loads to this minimum load. If the thrust load is required to be unidirectional, the maximum bearing load will be greater than if the thrust load is allowed to pass through zero during some transient condition. Restricting the contractor to unidirectional loads could, therefore, result in shorter bearing lives than if bearings were allowed to be null loaded. Trade studies should be performed on sizing bearings to take unidirectional loads versus allowing

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crossover. Null loaded bearings can lead to rotor vibrational problems within the operating envelope which can lead to bearing failure or reduced life. If crossover is allowed to occur, the loss of loading on the bearing can result in transient rotor dynamic vibration. Transient rotor dynamic vibration in the operating range may result in pilot discomfort and distraction, and cause durability problems.

Engine/airframe system interactions can be excited by these types of vibration and cause similar problems. These interactions are typically difficult to predict and are usually not found until late in the integration effort.

REQUIREMENT LESSONS LEARNED (A.4.10.10)

None.

A.5.10.10 Pressure balance.

The requirement of 4.10.10 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.10)

The engine pressure balance system need be evaluated to ensure compliance with the requirement of A.4.10.10.

VERIFICATION GUIDANCE (A.5.10.10)

The engine contractor should provide an analysis of the engine pressure balance system. The analysis should show that loads imposed on the engine bearing(s) are of sufficient magnitude to ensure adequate bearing operation without skid damage. The analysis should be substantiated/correlated with engine testing.

An engine should be suitably instrumented and tested to demonstrate that the loads imposed on the engine bearing(s) are of sufficient magnitude to ensure adequate bearing operation without skid damage at all power settings throughout the engine operating envelope. This test should be conducted in an altitude test cell to simulate altitude and ram conditions representative of operational use.

VERIFICATION LESSONS LEARNED (A.5.10.10)

None.

A.4.10.11 Gyroscopic moments.

The engine will meet all the requirements of the specification at maximum allowable steady-state engine speeds when subjected to the rotational velocities and accelerations within the flight envelope and the gyroscopic moment conditions.

REQUIREMENT RATIONALE (A.4.10.11)

Engine flight loads are increased due to rotations and accelerations that occur during aircraft rolling, pitching, and yawing maneuvers. The engine need be designed to resist these loads at the limiting conditions.

REQUIREMENT GUIDANCE (A.4.10.11)

The gyroscopic moment conditions should be as specified below:

a. a steady angular velocity of __(a)__ radians per second around any axis in a plane perpendicular to the rotor axis, combined with a ±1g vertical maneuver load for a total of __(a)__ seconds

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b. a steady angular velocity of __(b)__ radians per second in any axis in a plane perpendicular to the rotor axis for a cyclic life of 10^7 cycles at all load factor conditions within the flight envelope.

The following should be used to tailor the specification paragraph:

(a): This requirement is principally a spin departure criteria requiring a high angular velocity for a short period of time. Angular velocities as high as 3.5 radians per second for fighter aircraft and 1.5 radians per second for bomber and cargo aircraft for 15 seconds are appropriate. Angular velocities of 2.5 radians per second for 30 seconds is appropriate for rotary wing aircraft. The 15-second and 30-second durations are considered to be cumulative exposure time.

(b): This requirement is principally a maximum angular velocity that may be experienced numerous times for long periods such as tight turns or numerous gust-induced nacelle oscillations of a pylon-mounted engine. This angular velocity value is approximately 1 radian per second in pitch or yaw for fighter aircraft. The angular velocity should be the vector sum of the angular velocities in pitch and yaw. Therefore, a steady angular velocity of 1.4 radians per second in any axis in a plane perpendicular to the rotor axis for a cyclic life of 10^7 cycles at all load factor conditions within the flight envelope is recommended. A steady angular velocity of 0.9 radians per second in any axis in a plane perpendicular to the rotor axis for a cyclic life of 10^7 should be used for rotary wing aircraft.

Alternately, for rotary wing aircraft, use 0.4 radians per second for 24 percent of duty cycle life, 0.9 radians per second for 20 percent of duty cycle life, and 1.4 radians per second for 1 percent of duty cycle life.

REQUIREMENT LESSONS LEARNED (A.4.10.11)

A USAF study showed that a bearing load could be increased by as much as 15 times by a 3.5 rad/sec aircraft turn rate. The force on an individual rotor blade could be three times (3 X) the magnitude of the corresponding aerodynamic force. Gyroscopic forces are cyclic in nature, thus tending to reduce the cyclic life of rotor blades. The vector sum of pitch and yaw rates is very important from the standpoint of the gyroscopic moments produced on the engine.

The fighter/attack aircraft and the lighter-weight engines with tighter clearances will make the gyro requirement even more essential. Simulation data from one-type fighter aircraft produced yaw rates of less than 0.5 rad/sec and pitch rates less than 1 rad/sec. Intentional departure stall maneuvers have been used as a "last chance" evasive action against surface-to-air missiles (SAM's). This action results in high yaw rates and gyro loads. Yaw rates of 2.7 rad/sec were demonstrated years ago, in flight, during accelerated departure stall maneuvers, with the attack aircraft. Yaw rates of 3.1 and 3.2 rad/sec were measured on two models of fighter aircraft. One military aircraft mishap was reported as yawing in excess of 200 degrees per second (3.5 rad/sec). The aircraft had violently departed controlled flight due to asymmetrical positioning of the leading edge flaps. Test of one-type military engine on the gyro rig showed significant mechanical interference occurred at 2 rad/sec. The manufacturer was able to correct the problem without major redesign.

A.5.10.11 Gyroscopic moments.

The requirements of 4.10.11 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.11)

Aircraft are exposed to gyroscopic moments during normal operation and the ability of their engines to withstand those conditions need be evaluated.

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VERIFICATION GUIDANCE (A.5.10.11)

The following should be transferred, verbatim, into the specification paragraph:

"Prior to installation on the test stand, the engine will be assembled with special emphasis placed on measuring and recording clearances between blades and cases and radial and axial rotor clearances. Rub probes will be installed around compressor and turbine cases at symmetrical locations and at blade tip locations as designated in the pre-test data. Instrumentation will be sufficient to permit measurement of rotor deflection and shift under gyroscopic loads. Strain gage instrumentation will be provided to measure stresses at critical locations. Sufficient instrumentation of the oil system will be provided to evaluate the oil system's ability to scavenge and function properly during the test.

The engine will be operated with an inlet configuration and exhaust nozzle as specified in the pre-test data. Prior to the test, the engine will be subjected to a performance calibration.

The test will be conducted with the gyroscopic rig operated in incremental steps of 0.5 rad/sec from 0.5 rad/sec up to and including 3.5 rad/sec. At each step, the engine will be operated as follows:

a. Idle for one minute.

b. Accelerate from idle to maximum allowable rotor speed in 30 seconds.

c. Dwell at maximum allowable rotor speed 10 seconds or time sufficient to record data.

d. Decelerate from maximum allowable rotor speed to idle in 30 seconds.

e. Stop rig and engine for visual check of rub.

NOTE: At gyro loads above 1.5 rad/sec, snap accelerations and decelerations may be made to reduce time exposure. The total time at 3.5 rad/sec gyro load will not exceed the time specified in A.4.5.3.

The above test will be conducted with the gyroscopic rig rotating in one direction and then the test will be repeated with the rig rotating in the opposite direction. At the completion of the test, the engine will be subjected to a post-test performance calibration and then disassembled for inspection.

The test will be satisfactorily completed when, in the judgment of the Using Service: (1) the post-test calibration reveals no significant loss in performance; (2) the engine and its systems operated properly during the test; (3) structural loads were within acceptable limits; and (4) teardown inspection reveals no evidence of excessive blade rubbing or evidence of impending failure.

The engine contractor should provide a gyroscopic moments analysis. As a minimum, the analysis should discuss engine mounts, bearings, and bearing support structure capability while exposed to the gyroscopic moment conditions specified in A.4.10.11. The analysis should be substantiated/correlated with a gyroscopic moment engine test."

VERIFICATION LESSONS LEARNED (A.5.10.11)

None.

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A.4.10.12 Main mounts.

The engine mounts will have adequate strength to retain the engine, including retained fluids and externals, at all flight, takeoff and landing, and ground conditions.

REQUIREMENT RATIONALE (A.4.10.12)

Engine mounts are necessary to attach the engine to the aircraft. Elastic limit and ultimate tensile strength load limits need be specified to ensure that both engine and aircraft designs are sufficiently strong for normal operations and safe for a limited range of crash landings throughout the specified service life. Mounts are required to have sufficient strength to protect the engine against a worst-case single attachment point failure in order to ensure safety of flight (i.e., fuel fire and single engine loss of power) and ground safety (i.e., crashed engines or aircraft) (see Lessons Learned). The number, locations, and descriptions of all engine mounts need be specified to ensure proper engine installation into the aircraft.

REQUIREMENT GUIDANCE (A.4.10.12)

The mounts should withstand elastic limit loads of __(a)__ without permanent deformation and ultimate tensile strength loads of __(b)__ without complete fracture. A total of __(c)__ mounts will be provided which have sufficient strength to prevent __(d)__ when subject to a single attachment point failure at any location at the end of the engine mount service life. The locations and descriptions of all engine mounts will be specified. The mount system will accommodate all off-axis loads when a thrust vectoring nozzle is used.

The following should be used to tailor the paragraph, except when the airframe's engine mounting requirements have been previously established:

(a), (b): The contractor should specify the maximum system limits in units of force and in reference to the engine. The specified values should include, but not be limited to, the effects of the following requirements and specific design characteristics: externally-applied forces (i.e., accelerations) of A.4.5.3, gyroscopic moments of A.4.10.11, all airframe loads which are supported through the engine structure (if such loads exist) and safety factors of A.4.10.1, cyclic fatigue, engine mass, material strength/mechanics, and service life. The contractor should specify the bending moment limits in the axial, vertical, and lateral directions. (NOTE: For competitive engine development programs, the Using Service should ensure the engine and aircraft specifications are compatible with the maximum loads of the worst-case engine and airframe combination.)

(c): The Contractor should specify the number of mounts.

(d): The Using Service should specify that engine mounts have sufficient strength to prevent "a reduction in engine power and change in engine position" for single engine aircraft or "loss of engine retention" for multi-engine aircraft.

The contractor should specify the locations and description of engine mounts to ensure that both engine and aircraft are designed to a common goal. Redundant mounts should also be indicated on a figure.

NOTE: The Using Service should ensure that limited crash loading accelerations are specified and identified on the figures, so the contractor can specify ultimate load limits.

Background:

The vector forces for engines which utilize thrust vectoring should be determined by the engine contractor. These forces will be based upon a particular engine/airframe combination. Because of the higher vertical loads imposed during thrust vectoring, it is essential the engine mount system sustain the worst-case load predictions that the aircraft would obtain. The engine mounting system needs to be suitable for the thrust vectoring nozzle arrangement and the

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mount system needs to be durable enough to withstand the various loads that will be imposed when the nozzle is vectored at selected angles.

Part 33 of the Federal Acquisition Regulation (FAR) requires engine mounts have both elastic and ultimate strength integrity.

REQUIREMENT LESSONS LEARNED (A.4.10.12)

Engine mounts have failed in flight. The right engine (JT8-PW) on a Boeing 737-200 jetliner fell-off shortly after takeoff from O'Hare International Airport on 20 January 1989 (Piedmont flight 1480). According to an eyewitness report, "There was a lot of smoke coming out of one engine, and we saw it leaning, almost falling off, and then it fell off..." (ref.: Washington Post, Jan 21, 1989). Similar incidents which involved 737s were a Southwest Airlines flight leaving Dallas on January 3, 1986 and USAir flight leaving Philadelphia on December 5, 1987 (ref.: “The Washington Post;” January 22, 1989).

The single attachment point guidance for multiple-engine aircraft will likely result in two forward mounts and one aft mount, with an aft mount failure possibly resulting in partial engine separation (e.g., inelastic mount elongation) and engine shutdown. Guidance for single-engine aircraft will likely result in a design with four mounts arranged in a rectangular or diamond-shaped pattern, without loss of engine power or change in engine position (33-percent redundancy factor and stresses in the elastic range) after single attachment point failure.

A.5.10.12 Main mounts.

The requirements of 4.10.12 will be evaluated by analysis and test.

VERIFICATION RATIONALE (A.5.10.12)

Elastic limit and ultimate tensile strength load limits need be evaluated to ensure the engine is sufficiently strong for normal operations and safe for limited crash landings or single point attachment failures throughout the service life.

VERIFICATION GUIDANCE (A.5.10.12)

Engine-mount requirements should be evaluated by analysis of the worst-case engine-mount failures and their consequences, to minimize cost prior to preflight qualification. Engine-mount testing should be conducted after the completion of endurance test cycling and prior to production qualification. See A.4.10.12 Requirement Guidance for load criteria, which should be included in the evaluation.

Thrust vectoring nozzles impose new structural loads upon the engine and the airframe. These loads need be examined carefully by analysis of test data. Determination as to whether the mount system will withstand these new forces is a factor of mount system strength, durability, and mount system life considerations.

The engine mounting system need be suitable to the thrust vectoring nozzle arrangement, and the mount system should be durable enough to withstand the various loads that will be imposed by directing the engine thrust at selected angles.

The load calculations done by the engine contractor during the design phase should be evaluated during sea-level demonstrations and tests.

VERIFICATION LESSONS LEARNED (A.5.10.12)

The most commonly known failure mode for the engine mounting system was metal fatigue of the mounting bolts.

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A.4.10.13 Ground-handling mounts.

The ground-handling mounts will support the engine, including all engine-mounted equipment and externals, components, and operating fluids, under the following maximum inertia load conditions, without deformation to the mounts or damage to the engine: (a) axial, (a) lateral, and (a) vertical acting in combination at the engine center of gravity.

The locations and descriptions for the individual ground-handling mounts will be specified. The arrangement will be compatible with ground-handling equipment specified herein by the Using Service.

REQUIREMENT RATIONALE (A.4.10.13)

Ground handling mounts are required to provide a means to lift the engine during the installation/removal from the aircraft, and for ground transportation and maintenance.

REQUIREMENT GUIDANCE (A.4.10.13)

The following should be used to tailor the specification paragraph:

(a): The Using Service should specify at least 4g axial, 2g lateral, and 3g vertical.

Background:

See AFGS-87233 for more information on handling-mount requirements. The mounts should be designed to be compatible with existing engine transportation and maintenance equipment. Life Cycle Cost studies, in conjunction with operational requirement studies, should be conducted to determine if particular or existing engine handling equipment should be used.

If adapters are necessary, they should be designed and provided by the engine manufacturer, as specified in the contract.

REQUIREMENT LESSONS LEARNED (A.4.10.13)

Ground handling mounts and related support equipment have not always been adequately designed and compatible. Engine damage has occurred due to engine support equipment with insufficient shock-absorbing capability, thereby transmitting high "g" forces into the engine. Also, engine ground handling mounts have been damaged or have failed because the mounts are not adequately designed for the loads imposed during engine transportation, maintenance, and installation/removal from the air vehicle.

A.5.10.13 Ground handling mounts.

The requirements of 4.10.13 will be evaluated by analysis, demonstration, and test.

VERIFICATION RATIONALE (A.5.10.13)

The intent of this paragraph is to evaluate whether ground handling mounts provided on the engine are adequate for ground handling, transportation, and maintenance of the engine.

VERIFICATION GUIDANCE (A.5.10.13)

The following should be transferred, verbatim, to the specification paragraph:

"The following procedures will be demonstrated:

a. on-base ground transportation

b. engine installation/removal from the air vehicle

c. engine maintenance tasks

d. other ground handling tasks peculiar to the engine."

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Background:

Tests should be conducted to load levels sufficient to evaluate limit load and ultimate load operational requirements and to evaluate that minimum strength components can meet the load requirements, assuming the test components have average strength capability.

VERIFICATION LESSONS LEARNED (A.5.10.13)

None.

A.4.10.14 Engine stiffness.

The estimated stiffness of the engine in resisting loads and moments applied at the outboard end of the output shaft, relative to the engine mounting points, will be specified herein. The first "free-free" lateral and vertical engine bending modes will be specified herein.

REQUIREMENT RATIONALE (A.4.10.14)

Aircraft maneuvers with turboprop/turboshaft engines cause large moments about the propeller shaft. This may increase propeller shaft and engine/gearbox case deflection thereby requiring better structural rigidity.

REQUIREMENT GUIDANCE (A.4.10.14)

Background:

Engine stiffness is the total deflection of the engine at the output shaft/propeller relative to airframe mounting points.

The loads should include, but not be limited to, the effects of externally-applied forces, gyroscopic moments, safety factors, cyclic fatigue, material strength, and service life.

REQUIREMENT LESSONS LEARNED (A.4.10.14)

None.

A.5.10.14 Engine stiffness.

The requirements of 4.10.14 will be evaluated by analysis, demonstration, and test.

VERIFICATION RATIONALE (A.5.10.14)

The test of the engine stiffness is required to ensure compatibility with the aircraft.

VERIFICATION GUIDANCE (A.5.10.14)

Background:

Engine stiffness should be evaluated prior to preflight qualification since the design may be impacted. See externally-applied forces of A.5.5.3.

VERIFICATION LESSONS LEARNED (A.5.10.14)

None.

A.4.11 Deterioration.

The engine will be able to attain the hot-part design life when operated at temperature conditions which represent a typical rate of performance deterioration. The temperature margin above the production acceptance engine maximum steady-state gas temperature under standard day conditions will be consistent with that required for the engine as stated in the engine specification for the design service life of 4.3.

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REQUIREMENT RATIONALE (A.4.11)

Ability of engine hot parts to meet design life requirements can be significantly reduced due to engine uptrim or other conditions that result in hot-gas-stream temperatures higher than that of the production engine. Some margin above the normal maximum steady-state gas temperature should be used during design of engine hot parts to ensure design life goals or requirements will be met.

REQUIREMENT GUIDANCE (A.4.11)

Aerospace Industries Association PC Project 338-2A members made a consensus recommendation in 1982, that the Procuring Activity not establish a specific temperature margin since this number will vary with engine type, application, and desired TBO. They pointed out that the main interest is that the engine achieve some minimum thrust or horsepower for a specified number of hours and that hot section parts be designed to account for the required temperature margin to achieve this objective. Recommended values for allowable thrust or power loss ranged from 0 percent (i.e., maintain rated thrust or power) to 5.0 percent. Individual recommendations for temperature allowances of 30°F to 70°F above the production acceptance engine maximum steady-state gas temperature were made. Based on the AIA consensus opinion, the operational requirement for deterioration has been derived as stated in A.4.11. It is recommended that the contractor specify in A.4.11 a usage interval equal to the hot part design life of A.4.3.1. The temperature allowance selected by the contractor based on his engine and experience should be called-out in the ENSIP Master Plan.

REQUIREMENT LESSONS LEARNED (A.4.11)

None.

A.5.11 Deterioration.

Capability of engine components to attain hot section part life under deterioration conditions will be evaluated as follows:

A.5.11.a Analysis.

Analysis of LCF, creep, stress rupture, and erosion capability accounting for the required temperature margin above maximum steady-state gas temperature will be performed.

VERIFICATION RATIONALE (A.5.11.a)

Early analysis that accounts for a margin above the maximum steady-state gas temperature will ensure maximum probability in attainment of the desired structural performance of the full engine. Early analysis will also identify proposed design configurations that are marginal or unacceptable to this criteria so that design changes can be pursued.

VERIFICATION GUIDANCE (A.5.11.a)

The contractor will include results of erosion, stress rupture, and creep analysis during design reviews with the Procuring Activity. These results will also be included in the strength and life reports required by the contract.

VERIFICATION LESSONS LEARNED (A.5.11.a)

None.

A.5.11.b Performance.

Component structural performance during conduct of the several engine tests will be verified.

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VERIFICATION RATIONALE (A.5.11.b)

Final evaluation of the capability of engine hot parts to meet design life requirements needs to be based on results of full engine tests. Such tests are required by this document.

VERIFICATION GUIDANCE (A.5.11.b)

The contractor will identify those engine tests which will be used to evaluate the capability of engine hot parts to meet life requirements under deteriorated conditions. It is recommended that the AMT of A.5.9.1.2 be used for this evaluation. A.4.3.1 of this document will contain the life requirements for hot parts; A.4.11 of this document will state that 100-percent rated engine thrust be achieved at the end of one (1) hot section life; and it is anticipated that the Primary Specification will require that specific fuel consumption will not be greater than 105 percent of the rating point at the end of one (1) hot section life. In accordance with A.5.9.1.6 of this document, the criteria for successful completion of the AMT is that the engine complete the test duration with the engine and components operating satisfactorily and that post-test calibration data reveal that the performance retention requirements have been met. Therefore, the AMT of A.5.9.1.2 is the prime engine test for evaluation that hot section life and deterioration requirements have been met.

It is also recommended that High Energy X-ray (HEX) be performed during full-scale engine testing to better understand component deformations, clearances, and other anomalies that affect performance and deterioration.

VERIFICATION LESSONS LEARNED (A.5.11.b)

None.

A.4.12 Creep.

The engine static and rotating parts will not creep to the extent that acceptable field engine operation is impaired for the operating conditions and the lifetime specified in 4.3. Part creep will not affect disassembly and reassembly of the engine or new part replacement at overhaul throughout the specified life of the engine.

REQUIREMENT RATIONALE (A.4.12)

Component dimensional growth needs to be minimized on static and rotating parts to insure acceptable engine operation is not impaired during service operation and that part replacement is not required.

REQUIREMENT GUIDANCE (A.4.12)

Design stresses should not exceed minimum value 0.2-percent creep strength allowables at the operating metal temperatures and time at temperature based on the design service life and design usage of A.4.3. The required useful life of hot parts and cold parts will be as specified in A.4.3.1 and A.4.3.2. Also, it is recommended design stresses not exceed values associated with utilizing greater than 50 percent of the minimum stress rupture life during the design service life and design usage. Design extreme flight conditions and deteriorated conditions should be included in creep and stress rupture design analysis. Reasonable shop practices, including minor machining or plating repairs, is permissible to maintain build tolerances.

In the event the above recommended guidelines cannot be met, the contractor should provide suggested design guidelines to the Procuring Activity for review and consideration.

REQUIREMENT LESSONS LEARNED (A.4.12)

None.

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A.5.12 Creep.

Creep characteristics of the engine static and rotating parts will be verified per 5.12.a through 5.12.c.

A.5.12.a Analysis.

An analysis will be performed to demonstrate that sustained stress and temperature combinations will not result in detrimental permanent set/growth for the required design service life and design usage.

VERIFICATION RATIONALE (A.5.12.a)

Early creep and stress rupture analysis during design is required to establish stress levels that will ensure critical components can operate satisfactorily for the required design usage and service life.

VERIFICATION GUIDANCE (A.5.12.a)

Analytical prediction of creep and component growth and percent stress rupture life, as a function of design life, will be accomplished on each creep-critical component. Design operating stresses should be established based on past experience that indicates a high probability that satisfactory creep and stress rupture life can be achieved (e.g.; 0.2-percent plastic creep life, 0.005-inch diametrical rim growth, 50-percent stress rupture life, etc.). It is recommended component capability be established utilizing minimum creep strength and stress rupture material properties (e.g., –3 Sigma). Initial creep and stress rupture analysis results should be presented to the Procuring Activity during the Preliminary and Critical Design Reviews.

VERIFICATION LESSONS LEARNED (A.5.12.a)

None.

A.5.12.b Test.

A design development test plan and tests for creep evaluation will be developed and performed.

VERIFICATION RATIONALE (A.5.12.b)

Early verification of creep and stress rupture capability through early development component tests can identify potential problem areas and avoid the need for redesign/qualification efforts later in full-scale development.

VERIFICATION GUIDANCE (A.5.12.b)

It is recommended that component/specimen tests be conducted as early as possible on critical components (e.g., turbine disk rim lugs and turbine airfoils). The scope of development tests to evaluate creep and stress rupture should be identified in an appendix to this document or in the ENSIP Master Plan. Development tests may be waived where the contractor provides adequate, relevant experience.

VERIFICATION LESSONS LEARNED (A.5.12.b)

None.

A.5.12.c Inspection.

Inspection and evaluation of components will be performed subsequent to conduct of the several engine tests detailed by this handbook. These inspections will as a minimum be equivalent to the field and depot inspections.

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VERIFICATION RATIONALE (A.5.12.c)

Final evaluation of creep and stress rupture capability needs to include extended operation of the full engine. Several test engines will be run during full-scale development (e.g.; operability, vibration and thermal surveys, accelerated mission tests, etc.) and inspection of critical components from these engines can verify that creep life is adequate.

VERIFICATION GUIDANCE (A.5.12.c)

Inspection requirements for development test engines should include measurement of critical dimensions (e.g.; snap and rim diameters, bolt circles, bores) prior to test and at each scheduled inspection interval. Evaluation of dimensional growth as a function of test time should be conducted and included as a part of the interpretation and evaluation of test results (see A.5.9.1.6). The scope of evaluation of critical component growths should be identified in an appendix to this document or in the ENSIP Master Plan.

VERIFICATION LESSONS LEARNED (A.5.12.c)

None.

A.4.13 Vibration.

The engine as defined in 3.1.9 should be free of destructive vibration at all engine speeds and thrusts (including steady-state and transient conditions) within the flight and ground envelope.

REQUIREMENT RATIONALE (A.4.13)

Safety and maintenance costs require that the engine be free of destructive vibration for the design service life and design usage. Vibration levels that may cause problems can occur in various segments of the engine-operating envelope so that the total flight envelope needs to be examined.

REQUIREMENT GUIDANCE (A.4.13)

Specific guidance to establish vibrational design criteria is contained in A.4.13.1 through A.4.13.3.3 of this handbook.

REQUIREMENT LESSONS LEARNED (A.4.13)

Vibration-related engine failures have been the cause of many aircraft losses. Approximately 25 percent of all modern, single-engine aircraft losses have been the result of vibration-related engine failures. The resulting inspections, redesigns and retrofits have cost the USAF hundreds of millions of dollars and consumed millions of person hours. These events have been the result of very complex interactions of unsteady aerodynamics (such as vane or blade wakes, bow waves from fixed frames, shock interactions, etc.), damped mechanical responses (resonances, forced responses, shaft dynamics, etc.), and material characteristics (sensitivity to foreign object damage, fretting, edge of contact stresses, inappropriate material properties, etc). These events have been experienced in all parts of the engine including fans, compressors, high pressure turbines, low pressure turbines (LPT), gearboxes, external tubing, main shafts and static structural frames, and in all materials although more have occurred in titanium alloys than ferrous or nickel based alloys. Some events have occurred after the accumulation of millions of engine flight hours on a fleet of engines when a new mission was introduced that involved extended operations in a part of the flight envelope that had previously only been encountered in brief transients. Therefore, the requirement to characterize and demonstrate the vibration characteristics across the entire flight envelope is essential to understand and preclude these events in the future. While the flight envelope may change over time, the engines also change, deteriorating both in performance and structural capability as they age.Performance deterioration may lead to increases in physical speed of the rotors, which consumes the margin

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in the design for avoiding certain vibratory modes or cause certain modes to be encountered more frequently and at higher temperatures. Structural deterioration can also occur as a result of wear mechanisms. The most common is fretting which occurs between two rubbing surfaces such as a blade/disk joint. Fretting can reduce the HCF properties of titanium by 50 percent or more. Dwell time at high temperatures and high stresses has shown to affect the LCF capability of some materials and may affect the HCF properties.

A.5.13 Vibration.

Vibration characteristics of the engine (ref. 3.1.9) should be verified by analysis and test. The variation in vibration characteristics should be validated by probabilistic analysis and test.

VERIFICATION RATIONALE (A.5.13)

Assessment of the vibration characteristics of an engine requires a great deal of analysis which should be verified by testing. The validation of vibration response variation and probabilistic solutions is essential because vibration characteristics are highly sensitive to parameter variations and, if not properly accounted for, can lead to unexpected field failures.

VERIFICATION GUIDANCE (A.5.13)

Specific guidance in how to verify vibrational design criteria is contained in A.5.13.1 through A.5.13.3.3 of this handbook.

VERIFICATION LESSONS LEARNED (A.5.13)

Verification of an engine's capability to resist destructive vibrations has proven to be very challenging. There are many factors that affect the vibratory response of an engine at the component level, the engine level, and the installation level. These factors also change over the operational life of the system. Some of these factors are:

a. Operational Environment

b. Engine to Engine Variation

c. Part to Part Variation

d. Material Variation

e. Installation Effects

f. Aging Effects.

The characterization of any of these aspects in a deterministic fashion has proven to be incomplete as examples can be found where each of these variations has led to fleet events and aircraft mishaps. The verification process is therefore to be accomplished on a probabilistic basis and expressed in the likelihood of encountering a fleet event at a rate that is acceptable to the operator and the acquiring agency. For practical purposes a rate of encountering a critical vibration failure may be expressed as fractions of events per million engine flight hours. The current USAF risk management criteria allows for 0.1 Non Recoverable In Flight Shutdowns per million engine flight hours as an acceptable threshold.

Operational Environment

The design mission and design mission mix prescribed in the system specification is the starting point to assess the sensitivity of the engine to differing engine speeds, inlet total pressures, and inlet total temperatures. However, it has been found that the operator will adjust tactics and operational environments based on the capability of the aircraft, the base location, proximity and topography of the training ranges, etc. Therefore, the verification needs to consider the

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probability distribution of operations across the total temperature and total pressure envelope. Further, probability distributions of inlet distortion, augmentor operation, power extraction, etc., need to be developed to verify the engine capability.

Engine to Engine Variation

The unpredictable nature of vibratory failures in developmental and fielded systems is a result of the adverse combination of the statistical variation in several of the fundamental factors. This variation from engine to engine yields a failure distribution that may be no larger than 1 in 1000 engines. While this seems a small failure rate, it is unacceptable when the consequence is loss of the aircraft or major structural damage. Sources of these variations normally fall in two categories: variation in the damped mechanical response (many potential sources) and variation in the strength of the forcing functions (again, many potential sources). This creates a tremendous difficulty in conducting one or a few tests to demonstrate the engine's capability for vibrational tolerance and leads to a process that uses the test data to validate the stochastic models for the demonstration protocol.

Part to Part Variation

Given the effects of engine to engine variation noted above, the situation is further complicated by the strong, sometimes dominant, effect of part to part variation. Nowhere is this more dominant than with the appearance of a tuned absorber. With lightly damped systems, one element of a part may be coupled in such a way as to respond at much higher levels than the same element of the same part in another engine. Development of mistuning tools which use various approaches to reducing the computational time required for assessment and experimental techniques to identify the range of variations in specific parts are tools to assess the susceptibility of designs to high-level vibratory responses which result from part to part variation. There are also processes that allow the use of part to part variation to mistune a rotor intentionally to reduce the responses. A variation of mechanical mistuning may also be achieved by using unequal spacing of vane passages to prevent the development of traveling waves.

Installation Effects

Inlet designs have changed dramatically with the advent of stealth technologies. Obscured inlet designs have produced very high levels of unsteady pressure distortion which in turn may drive resonant responses in both the rotating and non-rotating structure of the fans. Also, it has been found that the connection of the engine gearbox to the aircraft-mounted accessory drive can be a major source of vibration that can cause failures of the driveshaft, gearbox, and other engine-mounted accessories. Wear can contribute to eccentricities and axial loading of the driveshaft which can drive vibrations throughout the engine systems. Installed vibration measurements, either continuously or at installation, will minimize these failures.

Aging Effects

As an engine ages several separate and distinct changes can occur. Clearances increase due to wear mechanisms; to hold performance constant, the control increases fuel flow and engine-corrected speeds increase. This increase may lead to encountering a mode of vibration that was expected to be beyond the operating range or was rarely encountered in previous operations. Also, age can change the structural properties of the materials used in construction of the engine. Wear can lead to fretting and/or galling of critical interfaces such as joints between the blades and disks, joints between disks and spacers, and contact between dampers and disks or blades. Fretting and/or galling can substantially reduce the HCF capability of the

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blade or disk material. Additionally, long dwell time at high temperatures and high stresses can change the LCF capability of some disk materials. The effect of dwell time on HCF properties is not well established.

A.4.13.1 Engine vibration limits.

Maximum engine mechanical vibration limits should be established as a function of frequency, engine order, and location and direction of measurement. Maximum engine mechanical vibration limits should be based on an acceptable margin of safety for the structural capability.

Any rotor critical speeds which excite detrimental engine system modes existing above or below the engine operating range should have a probabilistic design margin established on speed to account for the variation in speeds for different operating conditions, new engine performance, minimum engine, and fully-deteriorated engine. The system modes should be defined within the expected engine operating envelope from production until being pulled from service for low performance. Adequate damping and appropriate balancing should be provided so that any critical speed existing below maximum operating speed should be traversed safely with smooth engine operation. The variation in speeds based on operating conditions, etc., should be included.

REQUIREMENT RATIONALE (A.4.13.1)

Rationally-determined vibration limits can be a quality control tool in production and a maintenance guide for removal of engines from operation. Initial vibration limits in design are needed to judge when redesigns or changes are warranted. Resonance conditions should be avoided so that amplified response and structural failures do not occur. Margin is required between engine speeds and resonance speeds due to the variation that can occur in engine speeds due to Mach number, deterioration, or hot day conditions, or combinations thereof.

REQUIREMENT GUIDANCE (A.4.13.1)

A probabilistic design margin of rotor critical speed should be specified that meets system reliability limits. When there is insufficient confidence in probabilistic solutions, a deterministic margin of at least 20 percent should be specified for rotor critical speeds that exist above maximum operating speed or below idle speed. It is also recommended a deterministic margin of at least 20 percent be specified between the mounting system resonance and idle speed. Section 4.13.1 should specify the maximum permissible engine vibration limits (overall velocity or displacement limit and true root mean square [RMS]) at each accelerometer location on the engine compressor and turbine cases, accessory gearbox case and, if applicable, internal structure. The overall velocity limit specified for each accelerometer should be applicable up to a frequency of (RPM/60*2.5).

The limits should be specified for the engine in a test stand and for an installed engine. Vibration limits should also be specified for any pad locations for engine-mounted accessories. Specified limits have historically been 0.006-inch double amplitude or less for the production engine installed in a test stand, and a limit of 20 g's for pad locations. New or rebuilt engine acceptance limits should be less than the maximum to be used for field limits. Engine limits should be revised based on the total engine test experience to reach the optimum limits that will prevent frequent rejection of production engines or rejection of field engines prior to the desired service interval or life.

REQUIREMENT LESSONS LEARNED (A.4.13.1)

Mechanically-induced vibrations which result in bearing and structural failures have led to catastrophic engine failures and loss of aircraft in modern engines despite major improvements in balancing procedures and strict vibration limits on new and repaired engines. These failures have resulted in a major expense to the USAF for the purchase and fielding of highly capable oil

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debris inspection equipment and a large expenditure of person hours in conducting rigorous oil analysis. These failures were the result of a design which met the new and field requirements for vibration and had apparently adequate margin for a shaft critical speed. The shaft was relatively heavily damped, with the peak resonance well out of the operating range. The result was a damped response function that had a significant response at high operating speeds—well below the peak response but significant under some operating conditions. This caused one of the main shaft bearings to fail at an infrequent but unacceptable rate (since these failures have led to loss of aircraft). Premature wear and condemnations of cases and structural frames have occurred as the result of reliance on very strict balance requirements to manage shaft critical responses in the operating range. Gearbox bearing and power take-off shaft failures have occurred as a result of installation tolerances that allowed misalignment, coupling spline wear, and lack of routine maintenance requirements.

A.5.13.1 Vibration limits.

Verification of engine vibration limits should be by analysis and test. An analytical dynamic analysis of the engine and accessories should be performed to identify critical engine system modes, potential forcing functions, and resonance conditions. This model should be verified with engine testing. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance is to be provided by verifying that the probability levels for each contributing random variable used to compute probabilistic design margins or probability of failure are within the experimental data range for that variable.

VERIFICATION RATIONALE (A.5.13.1)

The verification of engine vibration limits is necessary to ensure safe operation of the engine in the field. Testing and analysis are necessary to ensure that all detrimental system modes are outside the operating range. The validation of vibration response variation and probabilistic predictions is essential because vibration characteristics are sensitive to parameter variations and, if not properly accounted for, can lead to unexpected field failures.

VERIFICATION GUIDANCE (A.5.13.1)

A dynamic model of the total engine system including rotor(s), bearings, frames, cases, and engine supports should be developed to analyze maneuver loads and synchronous and nonsynchronous vibration responses. The model should include the effects of shear deformations, rotary inertia, multishaft gyroscopic influences, bearing speed effects on stiffness and damping, anisotropic bearing supports, and speed and frequency influences on supporting structural stiffness. Stiffness values used to represent flanges, splines, couplings, joints, and tapered elements of the structure and rotor should be verified by test. The model should be able to handle unbalanced distributions which result from bowed rotors or assembly of shaft components, including residual unbalance plus angular or offset misalignments of these shaft sections. Clearances and bearing loads should be determined at each speed with the most adverse magnitude and phase relations of the unbalance. Results from the models should be used to guide unbalanced engine testing. The models should be updated and verified throughout the development of the engine, as hardware and test information becomes available.

The model should be exercised to determine all system modes. Vibration amplitudes, clearances, and bearing loads should be determined at each critical speed with the most adverse magnitude and phase relations of the unbalance associated with the critical speed mode shape. Parametric studies of design changes should be conducted to determine a way to alter any detrimental dynamic modes which exist in the operating range. Results from the models should be used to guide unbalanced engine testing. The models should be updated

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and verified throughout the development of the engine as hardware and test information becomes available.

A mechanical impedance/resonance search static test plan should be developed and tests performed on external hardware and accessories. The test of the hardware should be excited by a shaker in mutually perpendicular planes throughout a frequency range sufficient to cover an operating range from 0 percent to 125 percent of maximum 1/rev-core speed. Strobe lights in conjunction with video recording and reusable tri-axial accelerometers should be utilized to determine maximum displacement response locations. Resonance frequencies should be determined, correlated to analytical models, and compared with potential sources of excitation such as rotor imbalance.

This type of test should be performed early in engine development with the engine in the test stand and then repeated with the engine installed in the airframe system. The system contractor typically performs the installed test to assure dynamic compatibility between the airframe and engine. Each of the contractor responsibilities during the installed test are usually defined in the Interface Control Document (ICD).

Instrumented engine imbalance tests should be performed to evaluate the dynamic response of individual external hardware and accessory components. Accelerometers should also be attached at the case flanges to estimate total vibratory input to the external hardware and accessories. Instrumentation should include tri-axial accelerometers at predicted high relative deflection locations and strain gages at predicted high pseudo-stress locations.

Tests should be conducted on the engine under ram conditions, with appropriate aircraft inlet conditions, simulated fan distortion, compressor bleed and nonbleed, power extraction, off-nominal guide vane schedules, with stalls and other important variables simulated during each test as appropriate. Effects of rotor imbalance relative to field guideline limits should be evaluated, as well as bowed rotor start transient effects. The rotor(s) should be unbalanced with the most adverse weight placement for the lowest critical speed and a phase angle predicted by analysis of residual unbalance. Magnitudes of total unbalance should be large enough to overcome typical residual unbalances to reach maximum levels found in similar engines prior to overhaul and stated field limits. The engine should be run through the operating range to maximum power (redline speed). A sufficient amount of instrumentation should be installed on load-transmitting structures and internal engine components such as the main bearings to permit measurement of bearing loads, cage rotation, and rotor deflections and case flange loads. External components such as fuel controls, fuel pumps, valves, tubes, brackets, etc., should be instrumented at high deflection locations with accelerometers and high pseudo-stress locations with strain gages as predicted by pre-test analysis. Inlet or exhaust systems and other components that are mounted directly on or supported by the engine when mounted in the airframe should also be instrumented based on analytical prediction.

Analytical models should be validated by comparing measured engine test frequencies and displacements to analytical results. A well-correlated analytical model typically will have a predicted resonant response frequency within 10 percent of the resonant frequencies as measured during the test. Analytically-predicted resonant frequencies that are considered correlated can be used to calculate the approximate mechanical impedance by determining the ratio of the measured peak hold deflection at a node equivalent to the measured location. The ratio can then be used to multiply the assumed pre-test mechanical impedance to increase or decrease it and bring the analytical deflections into agreement with the test data.

It should be noted that most vibratory analytical models of external hardware are linear by necessity. Therefore, the assumed vibratory input and mechanical impedance are linear multiples of the normalized displacement vectors calculated by fundamental modal analysis. In other words, the analytically-assumed input and impedence are coupled in the analysis.

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Therefore, the equivalent displacement might be achieved by multiplication of the analytically-assumed inputs instead of the impedence. The inputs to the hardware will be known for controlled shaker input tests and the input assumptions as well as the impedence assumptions can be adjusted to the test data.

VERIFICATION LESSONS LEARNED (A.5.13.1)

The analysis and test data gathered to demonstrate the engine's capability for mechanical vibration should cover all the expected operational envelope for rotor speeds both absolute and relative for engines with two or more spools and the envelope of total inlet pressure and temperature, as the shaft critical speed and/or skidding of the bearings can occur when an axial load reversal between the shafts occurs. Engines have been rejected at the field test cell after rebuild when the vibrations exceeded the established limits even though these same engines passed the acceptance testing at the manufacturer. Later, it was found the acceptance testing was evaluated only at maximum rotor speed while the overhaul limits were applied across the entire speed range. The acceptance test was inadequate; the installation actually required the limits be met across the entire speed range. The rotor balance was found to be extremely critical and the exchange of seemingly identical components could result in passing or rejecting the engine at test cell. The verification should demonstrate enough capability to meet the specific vibration requirements with tolerance to normal manufacturing variations and balance procedures. Conducting accelerated mission testing with an engine intentionally unbalanced to field limits has shown promise in identification of external components which are particularly susceptible to engine vibration.

The mechanical impedance test is a fundamental validation tool for both airframe/engine dynamic engineers and engine hardware designers. This tool provides information that enables engine designers to determine the dynamic resonant response of engine hardware from engine rotor unbalances and airframe to engine interaction. It is a best practice to have an analytical mode for each component validated by instrumented imbalanced engine tests. However, it should be recognized that test resources are typically limited due to instrumentation channel/equipment limitations and test facility and personnel availability. Therefore, it is a best practice to prioritize the instrumentation need for hardware based upon the analytically derived high cycle fatigue failure risk and the failure consequence effects. The prioritization ensures that data is collected on the highest risk components. This in turn enables the hardware designer with early indications to abate the risk prior to production which enhances the overall engine reliability.

A.4.13.2 Surge and stall.

The engine should operate satisfactorily without structural degradation which would cause the engine to not meet the requirements of this document for the design service life in the event of surges and stalls within the flight envelope.

REQUIREMENT RATIONALE (A.4.13.2)

Safety and maintenance considerations dictate the engine should be tolerant to repeated off-baseline occurrences of conditions such as surge and stall.

REQUIREMENT GUIDANCE (A.4.13.2)

The effect of surges and stalls should be considered in design in terms of frequency of occurrence, length of time involved, expected frequency, and magnitude of vibratory stress. The objective is to determine the extent to which surges and stalls contribute to fatigue, wear,

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and thermal damage over the required life of the engine. Surges and stalls that occur during the full-scale development flight testing and results of subsequent teardown inspections should be documented to demonstrate the requirement is met.

REQUIREMENT LESSONS LEARNED (A.4.13.2)

None.

A.5.13.2 Surge and stall.

Verification of the engine's capability to withstand surges and stalls should be by analysis and test.

VERIFICATION RATIONALE (A.5.13.2)

Analysis and testing are required to predict effectively the engine's response to surge and stall.

VERIFICATION GUIDANCE (A.5.13.2)

The model developed in section 5.13.1 should be used to predict the effects of surge and stall. The models should be updated and verified throughout the development of the engine as hardware and test information becomes available.

Instrumented engine tests should be conducted to verify the dynamic response of the total engine system to surge and stall. Instrumentation should include accelerometers, strain gages, and proximity probes in the vertical and horizontal planes. Effects of rotor imbalance up to the maximum allowable should be verified.

VERIFICATION LESSONS LEARNED (A.5.13.2)

None.

A.4.13.3 High cycle fatigue (HCF).

The probability of failure due to high cycle fatigue (HCF) for any component within or mounted to the engine should be below $1x10^{-7}$ per EFH on a per-stage basis, provided the system-level safety requirements are met.

REQUIREMENT RATIONALE (A.4.13.3)

High cycle fatigue has been a major safety and maintenance problem in jet engines. Proper attention is required to minimize these failures. A requirement based upon probability of failure is most consistent with HCF experience, given the inherent variability of the many factors involved.

REQUIREMENT GUIDANCE (A.4.13.3)

General guidance for HCF design is provided here. Specific guidance for HCF design is provided in A.4.13.3.1 through A.4.13.3.3.

Variations in the endurance capability of the material, determination of dynamic stresses, determination of steady-state stresses (or pseudo-steady-state stresses), the sequence in which combinations of stresses occur, and other factors all affect the determination of HCF probability of failure. Indeed, this observation has led to the establishment of the probability of failure requirement. The HCF design will be a function of probabilistic design margins on frequency, predictions on the variation on alternating stress, and the current threshold-based approach for material capability.

The probabilistic design margin on frequency will give the probability of resonance for a mode at a given operational condition and should be compared to the system-level reliability

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requirement. It is conservatively assumed the probability of resonance equals the probability of failure for modes determined to have significant modal excitation (i.e., low order modes and adjacent upstream engine order excitations). Probabilistic design margins on frequency should be computed on the basis of steady-state operation at low order crossings and all known drivers within two stages either upstream or downstream of the subject component. Variations in resonance conditions should be accounted for through probabilistic analysis or appropriate engine test data distributions for the system design condition. Probabilistic design margins lead to a Probabilistic Campbell Diagram, which follows. Deterministic design margins of 10 percent may be used for preliminary design or when there is insufficient confidence in probabilistic solutions.

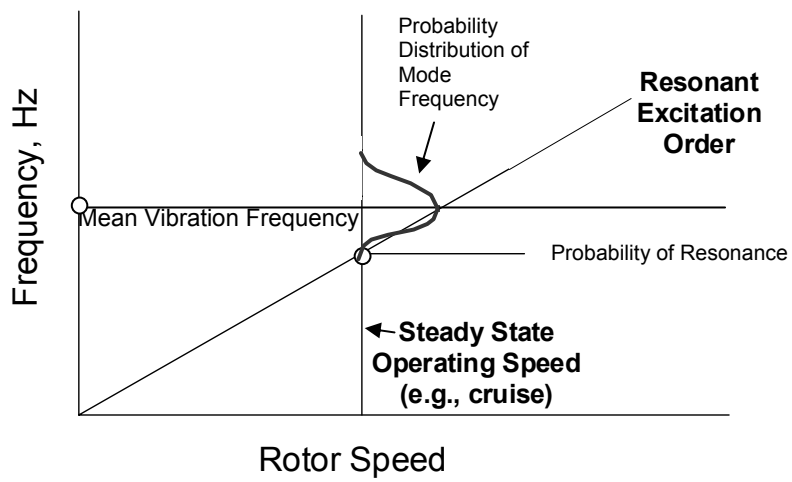

FIGURE 4. Notional Probabilistic Campbell Diagram.

In the event an insufficient probabilistic design margin on frequency exists to meet the probability of failure requirement, the next level of probabilistic design analysis is required to predict the variation in resonant stress response. Such probabilistic design analysis includes physical models of forced response, damping, and mistuning, along with the appropriate probability models for each random variable and correlations between those variables, as appropriate. It should be shown that at the resonant condition that was not avoided with the probabilistic design margin on frequency, the vibratory stress distribution is at a level that meets system reliability requirements when the stress is integrated with material capacity to predict failure. A deterministic design margin may be used when there is insufficient confidence in the probabilistic solution.

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Refer to the figure below for margins on the material capacity and the current threshold-based approach, and to section A.4.6 for definition of margin-related terminology.

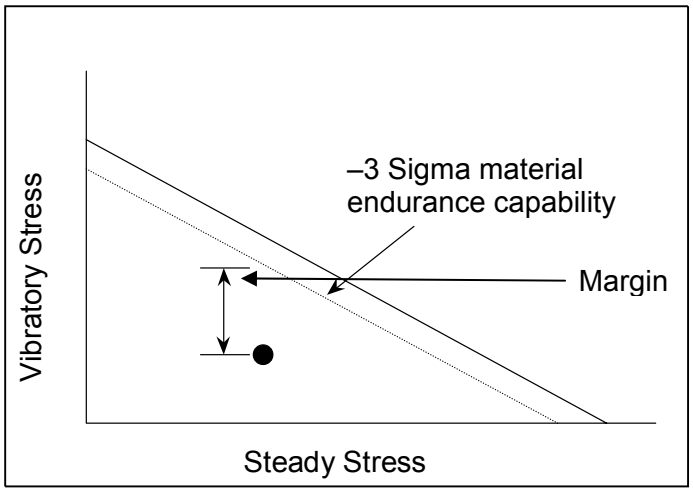

FIGURE 5. Material capacity margins.

Root mean square component vibratory stresses should not be within 60 percent of the minimum material endurance capability (i.e., limited to a maximum of 40 percent of the minimum material endurance capability). If instantaneous peak component vibratory stresses are used, they should not be within 40 percent of the material endurance capability (i.e., limited to a maximum of 60 percent of the minimum material allowance capability). Minimum material endurance capability is defined by S/N fatigue tests of a statistically-significant population of fatigue specimens at various combinations of steady and alternating loads, or R-ratios, representing various levels of local component steady-state, or pseudo-steady-state stress.

All engine parts should have a minimum HCF life of 10^9 cycles. This number is based on the observation that an endurance limit does not exist for most materials. If it can be shown through analysis or test that a given part will not experience 10^9 cycles during its design life, a number lower than 10^9 may be used. Such a condition may be established through analysis of vibrating frequencies and probabilities of a given part being subjected to steady-state or transient vibrations. It should be shown that the total time of exposure to any frequency and amplitude is less than 10^9 cycles or that the amplitude is less than the material allowable at 10^9 cycles. An alternate approach is to use a life of 10^9 cycles based on data obtained at shorter lives, but not less than 10^7 cycles, and a demonstrated valid method to extrapolate to 10^9 cycles to establish an endurance limit. Cycles which have vibratory stress amplitudes less than the endurance limit at 10^9 cycles can be considered to have no detrimental effect on pristine material and can be ignored in damage accumulation evaluation, provided no other damage is present (see section A.4.6).

10^9 Life Estimation

An example of an acceptable method to demonstrate 10^9 cyclic life is the Random Fatigue Limit (RFL) model [1]. This or a similar model can be used with limited 10^9 data to provide fatigue strength at 10^9 cycles. This process is shown graphically in the four following figures.

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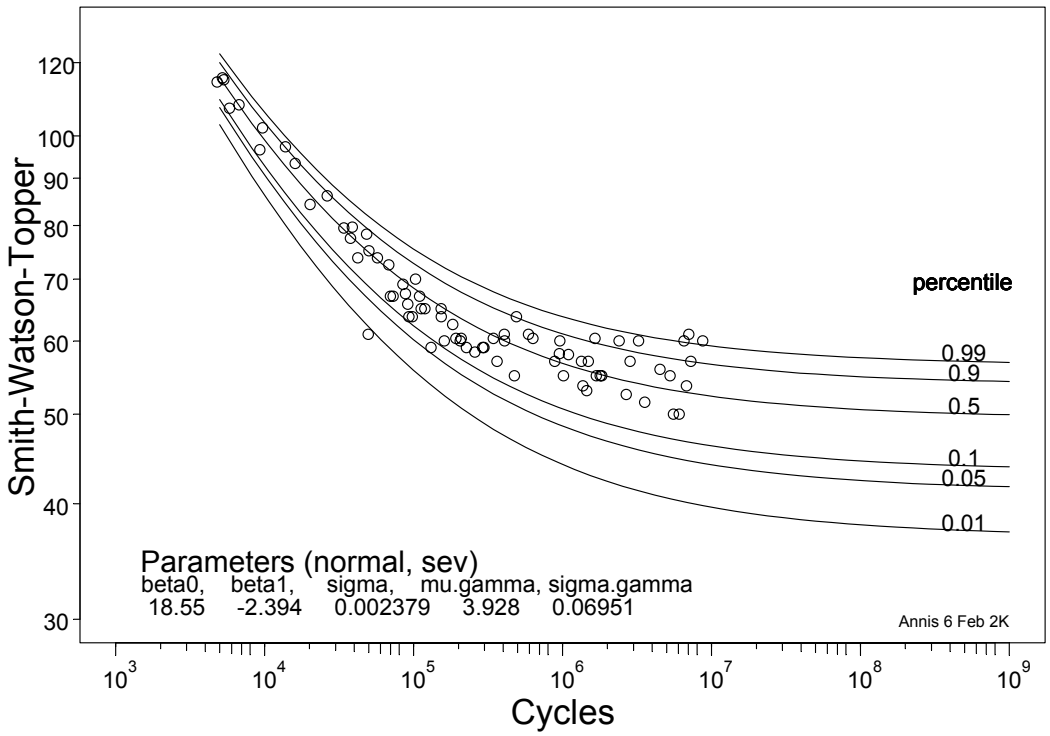

10^9 fatigue strength predicted using RFL model and <10^7 failure data. Percentages indicate the number of cycles likely to occur, statistically, with magnitude > $\sigma_{endurance}$.

FIGURE 5a. Random Fatigue Limit (RFL) model to predict 10^9 fatigue strength.

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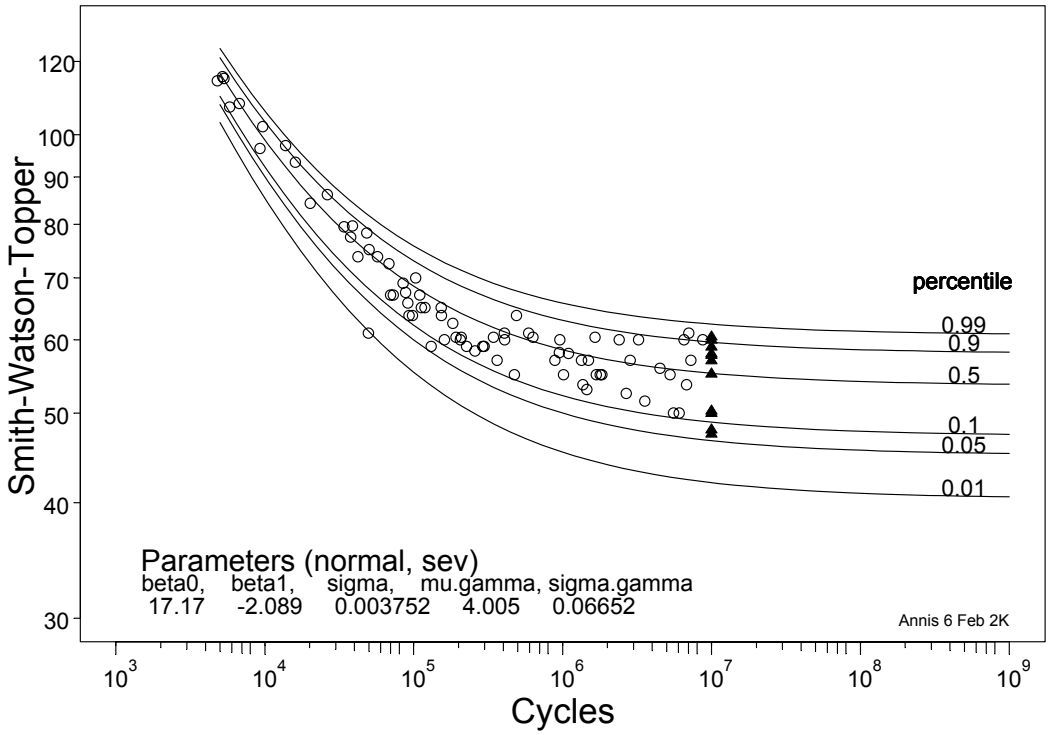

10^9 fatigue strength predicted using <10^7 failure data and 10^7 run-out data. Closed black triangles indicate run-out data.

FIGURE 5b. Random Fatigue Limit (RFL) model using <10^7 failure data.

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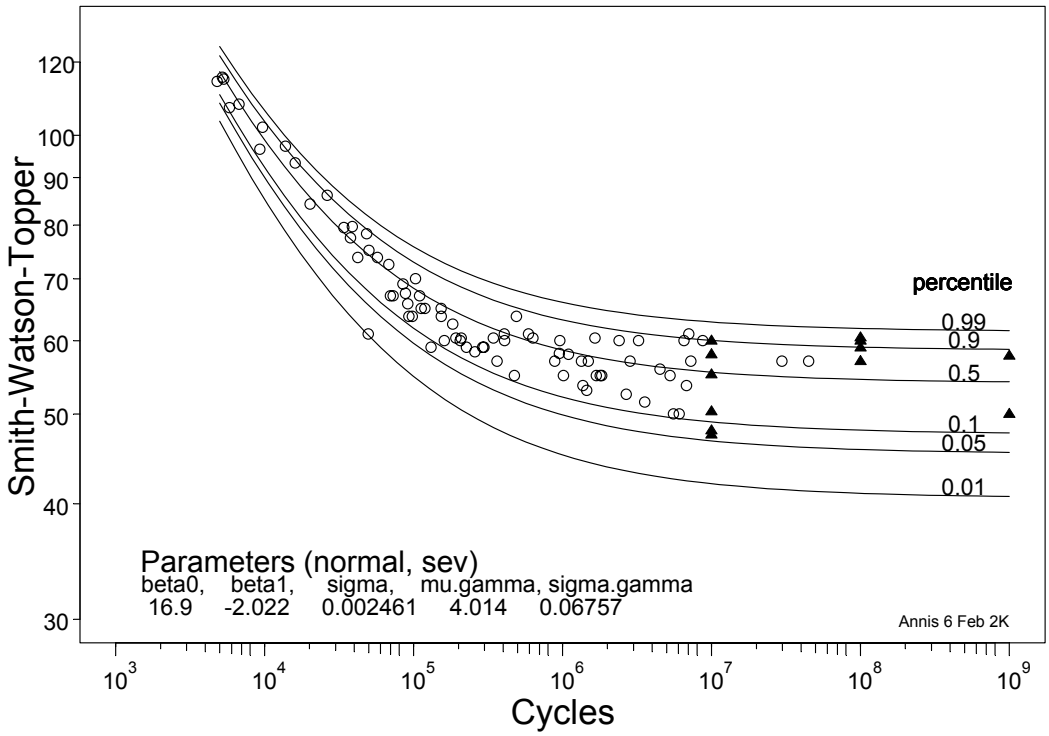

10^9 fatigue strength predicted using previous data and limited >10^7 failure data.
Closed black triangles indicate run-out data.

FIGURE 5c. Random Fatigue Limit (RFL) model using limited >10^7 failure data.

The RFL model overcomes the limitations of previous statistical descriptions of fatigue lives because it assumes each specimen has a random fatigue limit, rather than a set fatigue limit and a random life about that limit. The assumption basically orients the probability density function normal to the 0.5 percentile line. The assumed shape of the probability density function (PDF) is a five parameter Weibull. For higher cycle fatigue, the probability density function is nearly perpendicular to the x-axis (rather than parallel to it). This enables the fatigue limit distribution to have a much smaller deviation than previous models and thus decreases the scatter in the prediction. Orienting the PDF perpendicular to the x-axis also enables run-out data that was censored by other models to be considered. The run-out data is not considered as strongly as failed data, but since the specimens did not fail, it has a pronounced effect on the lower bound of the distribution, and a lesser effect on the upper bound. Additionally, the shape of the 0.5 percentile line is taken directly from the data. Because of this, any radical changes in the shape of the SN curve, such as that depicted in the next figure, that are not shown in the existing data set will not be captured in the 0.5 percentile line. Therefore, caution is necessary when attempting to extrapolate beyond the limits of experimental data.

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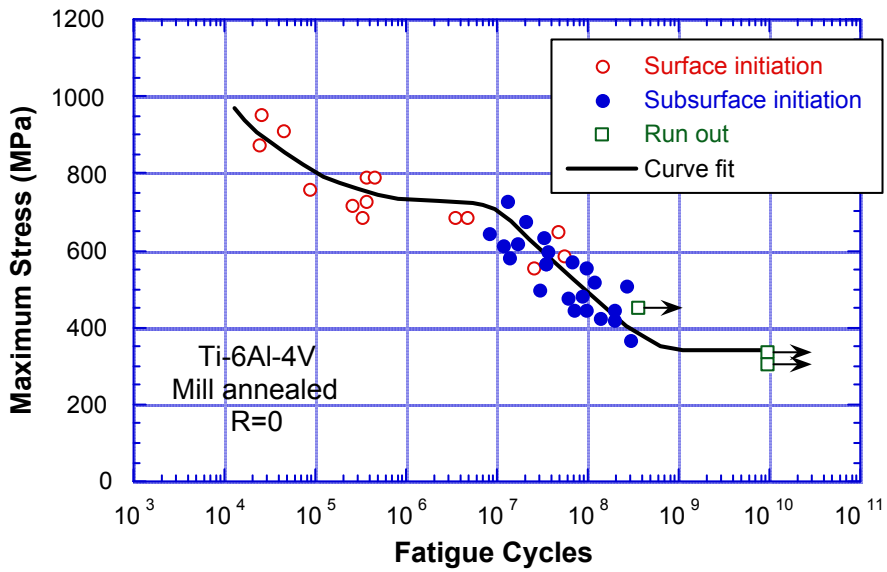

Illustration of change in SN curve shape in very high cycle fatigue [2].

FIGURE 5d. Change in SN curve.

The specifics of the analysis and distribution type can be found in the references below.

References

[1] Pascual and Meeker, "Estimating Fatigue Curves with the Random Fatigue-Limit Model," TECHNOMETRICS Vol. 41, No. 4, pp. 277-302, November 1999.

[2] Atrens, A., Hoffelner, W., Duerig, T.W., and Allison, J.E., "Subsurface Crack Initiation in High Cycle Fatigue in Ti6Al4V and in a Typical Martensitic Stainless Steel," Scripta Met., 17, 1983, pp. 601-606.

One-hundred-percent of the Goodman allowable may be used for components with surface enhancements, such as laser shock peening (LSP), low plasticity burnishing (LPB), and others. These processes are used to desensitize airfoil leading edges to foreign object damage by producing deep compressive residual stresses within a patch or region along the leading edge. The resulting through-thickness compressive residual stress field that is formed along a thin leading edge can produce a large improvement in HCF capability of damaged, surface-treated blades by arresting cracks which emanate from hard-body foreign object damage. The deep compressive stresses can also be useful to mitigate damage caused by other mechanisms, such as fretting fatigue and corrosion. When taking this type of surface enhancement credit, a threshold analysis needs to be conducted in addition to the normal probabilistic life analysis. The upper bound stress level (+3sigma or B0.1 from observed test data) should be used for the assessment.

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Laser shock peening is a surface enhancement process that produces a patch of overlapping laser-shocked spots along the leading edge. This is accomplished by using a high energy pulsed laser beam focused on an opaque coating on the blade surface through a water overlay to create a high amplitude shock wave. Laser shock peening is in flight service in several fighter and bomber engines in fan and compressor leading edge damage tolerance applications. All service applications to date have been in conventional Ti alloys. However, laboratory demonstrations of increased damage tolerance and HCF performance improvements have been successful in Ni-based superalloys, steels, and Al alloys.

The chief technical drawbacks associated with LSP are component distortion and compensatory tensile stresses. The treated region and LSP processing parameters need to be configured such that the compensatory tensile stresses are not of a sufficient magnitude or in a location where they can result in undesirable failure modes. Further, the location and depth of the residual stress field and the compensating far field stresses need to be validated through a combination of testing and analysis and accounted for in the design process.

Low plasticity burnishing is an emerging CNC controlled burnishing process designed to introduce deep compressive residual stresses into metallic components. Low plasticity burnishing was developed as another means of developing deep, thermally stable compression. The LPB process has been studied extensively in titanium- and nickel-based engine alloys and in aluminum and steel structural materials. Low plasticity burnishing is being developed as a cost-effective surface enhancement technique able to mitigate the effects of FOD, fretting damage, and corrosion pitting. This technology has demonstrated the ability to develop through-thickness compressive residual stresses in thin sections (such as blade edges) and depths of compression exceeding a millimeter in thicker components. Depths of compression over 7mm have been demonstrated in very thick-sectioned nickel-based alloy components. As with LSP, the chief reason for performance improvement is the deep compressive residual stress field induced by the LPB process. Since LPB is performed using modified CNC machine tools, the process is adaptable to a machine shop environment, which allows processing during either original manufacture or in overhaul facilities. Initial application is for fretting fatigue mitigation in the propeller hub. Other applications, including several blade edge FOD mitigation applications, are in development. As in the case of LSP, the treated region needs to be configured such that the compensatory tensile stresses are not of a sufficient magnitude or in a location where they can result in undesirable failure modes. Further, as with LSP, the location and depth of the residual stress field and the compensating far field stresses need to be validated through a combination of testing and analysis and accounted for in the design process.

REQUIREMENT LESSONS LEARNED (A.4.13.3)

Deterministic HCF requirements, while somewhat simpler to state, have proven to be ineffective in preventing developmental or field failures due to HCF. In fact, it has been shown that with these requirements as few as 40 percent of the HCF failure modes are found prior to qualification and production. A probabilistic assessment based on understanding all of the variables involved in each potential failure mode is key to minimizing the HCF failures that occur in service. The current risk management strategy for the USAF is based on an integrated risk assessment for each failure mode and is judged on the following criteria:

Field Action Required:	Probability of Failure $\geq 0.5X10^{-6}$ per EFH per failure mode
Field Action Considered:	$0.1X10^{-6}$ per EFH per failure mode $\leq$ Probability of Failure $\leq 0.5X10^{-6}$ per EFH per failure mode
Field Action Not Required:	Probability of Failure $\leq 0.1X10^{-6}$ per EFH per failure mode.

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These values are taken at a 50-percent confidence level. Fleet age distributions are projected based on average engine utilization and failure distributions are based on the average Weibull distribution.

These values, when aggregated to a system level, produce an overall probability of a Non-Recoverable In-flight Shut Down (NRIFSD) occurring approximately once in every 50,000 engine flight hours of fleet operation. This capability has been used successfully by the USAF for the last several years and has been acceptable to all levels of the USAF. The requirement as stated above is intended to produce an engine that does not require field action after qualification. The rate per EFH is used to reflect the effects of fleet size, fleet operating hours, and installations that require one or more engines. The rate per stage is to integrate the potential for several HCF failure modes being a concern in a given rotor stage or static structure, rather than expressing the rate for individual airfoils or components.

Complications exist with the concept of specifying all parts be designed to some discrete specified endurance limit. Some of these are:

a. Prior stressing at a higher stress can cause a lowering of the endurance limit.

b. Stress cycling at gradually-increased cyclic stress can result in an increased endurance limit (this is known as, “coaxing”).

c. Interactions between LCF and HCF can result in either increased or decreased lives; depending upon the magnitude of the loads, the order of the loading, and the material. This is referred to as, “load sequencing.” This phenomenon is evidence that HCF margin determination cannot be defined accurately without consideration of the overall stress-state over time.

d. Installation, handling, and environmental sensitivities can result in significantly higher steady-state and vibratory stresses which will reduce or even have negative margins for HCF capability. Such an example would be external parts which may be sensitive to all of the above. Realistic levels of stress due to these sensitivities should be included when HCF capability is assessed.

Because these complications exist, future efforts should be aimed at components designed to the HCF probability of failure requirement. Future efforts should also strive to integrate HCF-related damage with other forms of damage (e.g.; LCF, creep, etc.) by full consideration of the load sequence and the response of the material to that load sequence.

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A.5.13.3 High cycle fatigue.

Verification of the engine's ability to withstand high cycle fatigue should be through analysis and test. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance is to be provided by verifying that the probability levels for each contributing random variable are within the experimental data range for that variable.

VERIFICATION RATIONALE (A.5.13.3)

High cycle fatigue is a very complex problem. A great deal of testing and analysis is necessary to avoid HCF problems in the field.

VERIFICATION GUIDANCE (A.5.13.3)

When a validated design system is not in place, a method to extrapolate empirical data should be used to demonstrate compliance with 4.13.3 (e.g., integrally-bladed rotors [IBR's] validated with F100 data) Specific guidance for high cycle fatigue is provided in sections A.5.13.1 through A.5.13.3.

Because of the complex nature of HCF, maximum effort should be expended to insure that all information gathered at all steps of the design and verification process are leveraged. This approach is referred to as, "an holistic approach." Models developed during the earliest stages of the design process can be used to assess the sensitivity of mode shapes and frequencies to geometric variations and variations in boundary conditions or operational conditions (referred to as, "influence parameters"). Efforts should be aimed at maintenance of consistent modal characteristics over the range of geometric tolerances and influence parameters the part may experience.

Design models should be carried forward to the verification process. The verification process should include accepted practices for validation of the models such as the modal assurance criteria (MAC) and others. (For more information on MAC, see Shock and Vibration Handbook, Cyril M. Harris, 4th edition, McGraw Hill, New York, 1995, and its cited references.) Once models are verified, they should be used to establish optimal instrumentation locations for subsequent tests. Criteria to define optimum locations include mode sensitivity (the ability of the sensor to detect maximum mode amplitude), mode identification (the ability to distinguish between modes of similar frequency), and other physical criteria such as lead routing, proximity of sensors, and the like. Variations in part geometry can be addressed using sensitivity-based approaches applied to a nominal-geometry model.

After verification, models can be used to define vibratory and steady stress fields for all component locations and at engine operating conditions. These normalized stress fields can be scaled to results derived during experimental test to establish the stress time history (load sequence). Should the component fail to meet HCF design requirements, the verified models can be used to direct redesign efforts.

Verification tests in a lab environment (shaker table), in rigs, and in full-up engines generate large volumes of strain data. All of this data should be archived to establish a database of responses that can be used to assess variabilities and be used in the validation of probabilistic predictions. Further, examination of all relevant data is useful to define the robustness of a given design over the range of variables tested. Part-to-part variability and the variation of responses to influence parameters—like local pressure, temperature, or flow angularity—can be assessed using these data to define statistically what the maximum expected response may be during operational deployment.

Currently, a number of component surface treatments are available that are able to introduce deep compressive residual stresses sufficient to increase dramatically the leading edge damage

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tolerance of turbine engine airfoils. Of these processes, LSP is currently the only process in ongoing production and widespread service. Other processes, such as low plasticity burnishing, had yet to be introduced into service and full-scale production as of March 2004.

It is recommended that the design and evaluation of surface-treated blades, such as those subjected to LSP, be conducted in the following manner:

Design:

The blade design engineer should identify critical areas on the blade leading edge that are sensitive to FOD and should determine the approximate size of the surface treatment induced compressive residual stress field required. It is possible that a number of spanwise leading edge locations may be critical and that deep or through the thickness residual compressive stresses near full length of the leading edge may be needed. Adjustments in processing parameters may be required to prevent leading edge buckling for long spanwise patches.

Measurements of tip deflection and blade twist should be made before and after the application of the surface enhancement process to assess whether the component is within acceptable dimensional tolerances.

Simulated FOD damage for testing should be selected by the thorough analysis of field data and customer desires.

After the desired residual stress field is established, validated analytical/numerical methods should be employed to predict the in-depth stresses both within and outside the surface-treated region. In-depth X-ray diffraction measurements of the residual stresses should be accomplished at several critical locations and correlated with predictions.

Process Sensitivity and Quality Control:

There are many parameters that influence the residual stress state in a surface enhancement treated component. Any change in the processing parameters—such as patch size or configuration, the intensity of the processing, or the sequence in which the process is applied to the patch—may change the resultant residual stress state, and thus the fatigue performance of the surface-treated region.

All airfoil applications to date have been shot peened prior to LSP. If the sequence is to be changed then validation by component test is recommended.

Small laser burns and spallation that can sometimes result from LSP are undesirable. It is recommended that tolerance to LSP-induced defects be demonstrated through component test and analysis.

Testing:

The recommended testing to be used in evaluation falls into two categories: 1.) stair step (also referred to as step testing), and 2.) stair case (also referred to as up and down testing).

The damaged and undamaged blades (no residual stress inducing processing) should be component tested at the desired resonance in a siren or shaker HCF test facility using stair step test methodology: dwell at each vibratory stress level for at least one million cycles before moving to the next higher incremental level until failure occurs. The benefit of using the stair

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step testing is to evaluate a wide variety of test conditions of interest while very few specimens are consumed. Use the stair step data as a guide; stair case testing should be done using one specimen each for a selected vibratory stress level. The blade specimens either failed at this vibratory stress level or run out 10 million cycles. If run out is achieved, another specimen should be stepped to a higher vibratory stress amplitude and the process repeated. The step size should be constant and of sufficiently small size, and the number of tests sufficient, such that existing statistical guidelines to establish the mean fatigue limit strength can be used. This type test eliminates any coaxing effect that may be present in the stair step approach and represents the true HCF strength of the blade at a life of 10 million cycles. The desired amplitude level and frequency of resonance should be maintained by the operator by constantly monitoring the blade tip deflection. Strain gages should be used to establish test stress levels by calibrating to the blade deflection. Gages should be placed at areas of interest and reference locations on the part. If the highest stressed areas cannot be strain gaged, strain gages should be used in conjunction with a finite element model to predict peak stresses.

Surface enhancement processed blades should be damaged at the critical leading edge location and tested in the same manner as the baseline blades.

Quality Control:

Currently, measurement of residual stresses through the thickness or on the surface can be accomplished only via the X-ray Diffraction (XRD) technique. These in-depth measurements are destructive because they involve material removal. At this time, there is no satisfactory non-destructive measurement technique available that can measure in-depth residual stress. Surface measurements do not ensure a satisfactory residual stress profile exists subsurface. Therefore, quality control of production hardware should rely on component test sampling to ensure the desired effect is present before the production parts are shipped.

It is important that the condition of the laser beam in LSP or the process control parameters in LPB or other process be closely monitored for any deviation from desired operating condition.

Turbine Engine Qualification Testing:

A number of LSP'd configurations have been evaluated in test engines. Several severely-damaged LSP blades have been installed in engines and run at resonance under adverse conditions (producing higher vibratory stress than normal service conditions at the damaged location). In another qualification test that involved a single engine aircraft application, a similar resonance test was performed as above, and the same damaged blades were then installed in an AMT test engine and tested to a full engine inspection interval simulating 1,077 flights (4,300 TAC cycles).

VERIFICATION LESSONS LEARNED (A.5.13.3)

Historical approaches to HCF verification have relied almost entirely on experimental results. Design models were generally used only to establish whether the design should fabricated and to define the steady stress field at some assumed critical operating condition. Once fabricated, the locations of maximum vibratory stress, determination of critical locations, and HCF margin were done almost exclusively using experimental methods. These approaches have proven to be unsuccessful and have led to major increases in development time and costs along with major safety, supportability, and cost of ownership issues in fielded systems. Verification of the likelihood of an HCF failure mode on a probabilistic basis has been the process used to resolve field failures and while somewhat immature at this time reflects the current level of understanding of the underlying physics of the phenomena. Experiences with HCF in fans,

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compressors, turbines, shafts, and seals both damaged and undamaged has led to the need to verify the HCF capability on a probabilistic basis. One such event which caused a large fleet action was the result of changing the inlet characteristic without an assessment of the aeromechanical effect on the first stage fan rotor. A three-nodal diameter pattern was induced which aligned with a resonant mode in the fan rotor and resulted in blade vibratory stresses which led to failure in the presence of minor FOD—as small as 0.001 inch in depth. The consequence was a blade release, rotor unbalance, failure of the blade retainer and the release of several blades, then catastrophic uncontained engine failure. Redesign of the blade retainer and surface treatment of the blade to eliminate the FOD sensitivity has resulted in successful field operations. In a different engine, another failure mode was experienced when the resonance occurred only on the deceleration schedule from maximum power to idle. The variable vane schedule was unintentionally different from the acceleration schedule and the vane geometry such that separated flow impinged on the blades as they transited through the resonance and magnified the overall response. The resonant modes were well characterized, but the responses were very low during steady-state operations. Only with an unsteady CFD model was this situation discovered, then verified by test. Changes to the vane geometry and deceleration schedule successfully eliminated the blade/disk failures.

Verifying the HCF probabilistic assessment for a developmental engine is a formidable task that requires the integration of:

a. advanced aeromechanical tools

b. advanced understanding and characterization of manufacturing variations

c. new materials characterization methodology such as the use of Kitigawa diagrams in addition to sound Goodman diagrams

d. advanced instrumentation that can survive and provide needed experimental data across the full range of operating speeds—both steady state and transiently, inlet total temperature, inlet total pressure, and inlet distortion.

All these factors have necessitated an evolution in the verification process to integrate analytical techniques and experimental approaches. It also shows the importance of using probabilistic approaches to predict the variation responses that need to be validated with test and historical databases.

Laser shock peening is a flight-proven technology for increasing the damage tolerance of turbine engine structures. High cycle fatigue failures of blades from FOD on multi-engine aircraft have proved to be very costly because of downstream damage repair and tear down costs, inspection costs, and readiness issues. After seven years of operational use, not one LSP-treated blade has failed in the treated area on at least three separate weapon system fleets. One of these multi-engine systems averaged 13 blade FOD related failures a year prior to LSP.

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A.4.13.3.1 Engine system modes.

The engine should be free of detrimental resonance conditions at all speeds in the operating range. Any rotor critical speeds which excite detrimental engine system modes which exist above or below the engine operating range should have a probabilistic design margin established on speed to account for the variation in speeds for different operating conditions, new engine performance, minimum engine, and fully-deteriorated engine. The system modes should be defined within the expected engine operating envelope from production until being pulled from service for low performance. Adequate damping and appropriate balancing should be provided so that any critical speed that exists below maximum operating speed should be traversed safely with smooth engine operation. The variation in speeds based on operating conditions, etc., should be included. Operational conditions that are not part of the expected operational range but do exhibit significant stresses should be clearly documented to assist the USAF in future assessment of the engine's ability to perform changing missions.

REQUIREMENT RATIONALE (A.4.13.3.1)

Resonance conditions should be avoided so that amplified response and structural failures do not occur. Margin is required between engine speeds and resonance speeds due to the variation that can occur in engine speeds due to Mach number, deterioration, or hot day conditions, or combinations thereof.

REQUIREMENT GUIDANCE (A.4.13.3.1)

A probabilistic design margin on speed is to be defined for detrimental system mode conditions that exist above maximum operating speed or below idle speed. It is also recommended that a probabilistic design margin is to be specified between the mounting system resonance and idle speed. When there is insufficient confidence in probabilistic solutions, it is recommended that a deterministic margin of at least 20 percent be specified. The design margins should be validated by past engine and rig dynamic response performance. The margin needs to show the probability of critical resonance is a value that allows the engine to meet the design goal of 4.13.3. A deterministic margin of 20 percent may be used in preliminary design.

REQUIREMENT LESSONS LEARNED (A.4.13.3.1)

This requirement to avoid damaging resonances is more appropriately focused on limiting the resonant responses to amplitudes that are within the material's capability to withstand the vibratory stresses. In modern jet engines it is not possible to avoid the vast majority of resonant responses. It is likely that one can only avoid excitations that affect the lowest fundamental modes within the engine operating ranges. System modes are those modes with significant interaction between two or more different elements of an engine. The simplest example of a system mode is the coupling between the disk and the blades. There have been many examples of destructive resonances in fans that were driven by inlet patterns and downstream bow waves that resulted in higher nodal diameter responses of the disk/blade systems resulting in service failures. In no case was it possible to remove the forcing function. Rather, in each case, the disk/blade systems were changed to bring the resonant stresses within the material's capability through increased damping, surface treatments, or structural stiffening to de-couple the system mode.

System modes are also found in the critical speeds for the engine shafts and engine-driven accessories. Precision balancing and the use of oil-damped bearing systems have allowed modern large turbofan engines to allow the critical speed for the first mode to be within the operating range: below idle. This does not usually lead to HCF failures but has had other effects. The vibration caused by bowed rotor starts is usually accommodated within the bearing damping system, but in some cases stall margins have been reduced by the eccentric rubs of the bowed shaft. Also, excessive damping can lead to a significant response over a broad

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range of speeds even with the peak well outside the operating range. This is a well-known characteristic of a heavily damped system. The peak response may occur as high as 125 percent of the maximum rotor speed but the response at 100-percent speed may be as high as 50 percent of the peak response. This makes it critical to determine the damped response of the rotor system across the entire speed range, not just at the maximum.

The USAF has also experienced test cell rejections when the vibration acceptance criteria for the depot and field units are different from those used in the production acceptance testing. In one case, the production acceptance test measured vibration only at the maximum rotor speeds. The depot test procedures monitored the vibration from idle to maximum rotor speeds. Some engines showed out of limit vibrations at the low order shaft critical speed, which was between idle and maximum rotor speed. It was found that some shafts were slightly eccentric and depending on the balance of the rotors, these engines would fail depot acceptance testing. It is likely they would have also failed this test when they were originally produced. Resolution required disassembly, restacking, and rebalancing to lower the resonant response to acceptable limits.

Engine-mounted and engine-driven accessories are also subject to system mode vibrations which originate in the tower shaft, gearbox, and power take-off (PTO) shafts. There have been several instances of failures of PTO shafts that were designed with a critical speed in the operating envelope. These had damping that was sufficient when the shafts were new and the couplings well maintained. As the systems aged, the damping was reduced and the shafts or couplings failed. In other cases, the shafts produced vibrations that transferred into the gearbox, which caused premature failure of bearings. This vibration was sufficient to cause failure of engine-mounted clamps and housings, as well.

A.5.13.3.1 Engine system modes.

An analytical dynamic model of the engine and accessories should be performed to identify critical engine system modes, potential forcing functions, and resonance conditions. This model should be verified with engine testing. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance in these margins are to be provided by verifying that the probability levels for each contributing random variable used to compute probabilistic design margins or probability of failure are within the experimental data range for that variable.

VERIFICATION RATIONALE (A.5.13.3.1)

Testing and analysis are necessary to ensure all detrimental system modes are outside the operating range. The validation of vibration response variation and probabilistic predictions is essential because vibration characteristics are sensitive to parameter variations and if not properly accounted for can lead to unexpected field failures.

VERIFICATION GUIDANCE (A.5.13.3.1)

Analytical dynamic computer models should be developed to accomplish dynamic analysis of engine components and the assembled engine as a dynamical system. The models should be updated and verified throughout the development program as hardware and test information becomes available. Probabilistic methods should be applied to quantify the probabilistic distribution of the dynamic response. The probabilistic distribution should be validated with historical databases. Validation need continue after the development program and transition into operational tracking.

The probabilistic model developed in section 5.13.1 should be exercised to determine all system modes. Variations in vibration amplitudes, clearances, and bearing loads should be determined at each critical speed with the most adverse magnitude and phase relations of the unbalance

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associated with the critical speed mode shape. Sensitivity studies of design changes should be conducted to determine a way to alter any detrimental dynamic modes which exist in the operating range. Results from the models should be used to guide unbalanced engine testing. The models should be updated and verified throughout the development of the engine as hardware and test information becomes available.

Instrumented engine tests should be conducted to verify the dynamic response of the total engine system. Instrumentation should include accelerometers, and proximity probes in the vertical and horizontal planes. Tests should be conducted as early as possible on a core engine to determine vibratory stresses and to investigate flutter boundaries. Subsequent tests should be conducted on the full engine. Sensitivity to effects of rotor imbalance up to the maximum allowable should be verified. The rotor(s) should be unbalanced with the most adverse weight placement for the lowest critical speed and a phase angle predicted by analysis of residual unbalance. Magnitudes of total unbalance should be large enough to overcome typical residual unbalances to reach maximum levels found in similar engines prior to overhaul and to reach field vibration limits. The engine should be run through the operating range to maximum power.

VERIFICATION LESSONS LEARNED (A.5.13.3.1)

The verification of the engine's ability to endure the HCF stresses which result from system mode vibration is a comprehensive integration of analytical modeling and extensive component and engine testing to develop an understanding of the vibration environment and sensitivity of the overall engine system. One of the key elements in verification of system mode capability is in system identification. The lack of system identification has been one of the primary causes for system mode vibration that has lead to field failures. The contribution of bearing support stiffness to shaft vibration and subsequent bearing loading and failure is one example of system identification. Most analysis would show the bearing support to be rigid compared to the shaft and not consider its contribution to lowered shaft critical speed. Shaft balancing is accomplished with the part uninstalled in a rigid test fixture that confirms the analysis but may not represent the installation into the engine. New dynamic analysis tools and instrumentation will aid more comprehensive system identification. Reduced-order models can then be applied with some confidence that the system is represented by the models. Advanced vibration data analysis tools can provide valuable insights into system mode capability by gathering and processing data to extract the behavior of specific structural elements such as bearings, gears, and shafts. In situ vibration sensors can also provide valuable data for installation effects such as power take-off shaft and coupling shaft misalignments, etc.

a. Analysis: important but not sufficient
b. Testing: important but not sufficient
c. Understanding of vibratory environment: critical

A.4.13.3.2 Component vibrations.

Engine components should be free of detrimental resonance at all speeds in the operating range. This can be accomplished by intentionally designing modes out of the engine operating speed range or by providing sufficient damping, a probabilistic design margin on frequency and a probabilistic prediction of vibratory stress with respect to steady-state operating speeds, or excitation control to ensure that modes which remain in the running range do not respond detrimentally. A detrimental response is one that exceeds criteria outlined in A.4.13.3.

REQUIREMENT RATIONALE (A.4.13.3.2)

Resonance conditions should be avoided or controlled so that amplified response and structural failure does not occur. Margin, as defined in A.4.13.3, is required due to frequency variations that can occur among a population of engines; or because of changes in operational conditions, deterioration, distortion, or combinations thereof; and other important random design variables.

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Experience has shown that engine structural components operating under combined steady and vibratory stress conditions should be designed to ensure resistance to HCF cracking.

REQUIREMENT GUIDANCE (A.4.13.3.2)

Resonances in the engine operating speed range should not occur at steady-state operating speeds such as, but not limited to, idle, carrier approach, hover, cruise, or maximum. Sufficient frequency margin, as defined in the A.4.13.3, and established during normal operation of a nominal engine at sea-level conditions, should be provided to insure resonances at steady-state operating conditions do not occur elsewhere in the flight envelope or within a larger population of engines. Frequency margin can be provided above or below the steady-state speed. Engine components should not fail due to HCF or a combination of HCF and LCF when subject to the maximum attainable combined steady-state and vibratory stresses at a rate above that stated in A.4.13.3.

Low-order crossings and all known drivers within two stages either upstream or downstream of the subject component should have probabilistic design margins for final design that account for the variation in speeds for different operating conditions. A deterministic 10-percent frequency margin on component modes is acceptable for preliminary design or when there is insufficient confidence in probabilistic methods.

In the event that insufficient margin exists to meet the probability of failure requirement, the next level of probabilistic design analysis is required. Such probabilistic design analysis includes physical models of forced response, damping, and mistuning, along with the appropriate probability models for each random variable and correlations between those variables, as appropriate. Analyses on blades, vanes, disks, and integrally-bladed rotors (IBR's or blisks) should include the effects of response to unsteady aerodynamic pressures, damping, and mistuning. Probabilistic methods should be applied to predict the probability distribution of the dynamic responses. Without these analyses, HCF response of the component may be seriously underestimated.

a. Response to Unsteady Aerodynamic Pressure

Unsteady aerodynamic forces are the dominant cause of high frequency blade vibration that lead to high cycle fatigue. At a given frequency, oscillatory stresses in blades are directly proportional to aerodynamic loads. Errors in unsteady aerodynamic pressure prediction result in proportionally-equivalent errors in stress prediction.

Unsteady pressures estimates should be performed using state-of-the-art techniques current at the time of engine design. The impact of direct and lateral gusts from inlet distortion on internal pressures should be assessed. The impact of vane blade interaction on unsteady flow should also be assessed for use in forced response prediction. The impact of upstream propagation of aerodynamic disturbances should be assessed, where appropriate. Also, the potential impact of aerodynamic disturbances over multiple stages downstream and the impact of vane variability, within tolerances, on unsteady airflow should be assessed.

Tuned aeroelastic analysis depends upon knowing so-called "blade frequencies". While cantilever blade frequencies are sometimes adequate, often they are not. Finite Element Model (FEM) analysis of bladed disk assemblies has clearly shown that higher nodal diameter modes have higher natural frequencies. Thus, it is important to determine exactly what "blade frequency" is needed. The "blade frequencies" can be found from the equation:

$$\left(\begin{bmatrix} K' & -K'' \\ K'' & K' \end{bmatrix} + \begin{bmatrix} M & -M'' \\ M'' & M' \end{bmatrix}\omega^2\right)\begin{bmatrix} \mathbf{u}' \\ \mathbf{u}'' \end{bmatrix} = \mathbf{0} \ .$$

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The value of the eigenvalue λ represents the so-called blade frequency needed for tuned CFD analysis, while the complex mode shape $\mathbf{u}^*$ is the deformation shape of the blade sector at that frequency. Further information on "blade frequencies" including the derivation of this equation may be found in Platzer, M. F. and Carta, F. O., Eds., AGARD Manual on Aeroelasticity in Axial-flow Turbomachines, AGARDograph No. 298, NATO, Advisory Group for Aerospace Research and Development, Neuilly sur Seine, France, 2 Vols. (Vol. 1: Unsteady Turbomachinery Aerodynamics, March, 1987. Vol. 2: Structural Dynamics and Aeroelasticity, June, 1988.) and Fransson, T., and Sieverding, C. H., Eds., Aeroelasticity in Axial-Flow Turbomachines, LS 1999-05, von Karman Institute for Fluid Dynamics, 1999. Tuned aeroelastic analysis should use ω as derived here for the so-called "blade frequency", ς for the damping ratio (material or other damping mechanism results should be used for this value), and a "blade mode shape" of $\mathbf{u}^* = \mathbf{u}' + \mathbf{u}''$. These parameters should be found for each interblade phase angle ϕ.

b. Damping

When operating on resonance, stresses are directly proportional to the resident amplification factor, $Q = 1/\eta$ where η is the modal loss factor. As a result, the impact on stress prediction of errors in understanding of *Q* are proportionally equivalent to normalized errors in understanding unsteady aerodynamics. Thus, it is of paramount importance that damping measurement be performed in an accurate fashion that accounts for the complex dynamics which result from closely-spaced modes and damping non-linearities. Any airfoil mode indicated as a crossing on the Campbell Diagram in which nominal vibratory stresses are expected to equal or exceed 20 percent of the allowable should have a *Q* no greater than 70 under any operating conditions.

Friction damping at shrouds, root connections, and platform dampers should be designed using modern friction models (macro and micro-slip) which account for variability in temperature, normal loads, material properties, surface smoothness, velocity, and displacement dependencies, and any other parameters that can reasonably be expected to impact damping.

c. Structural Mistuning

Structural mistuning is defined as differences in blade-to-blade geometry and/or material properties. To capture the mistuning phenomenon in turbomachinery rotors, one should model the entire structure (rotor and blades), which can be a much more computationally-expensive problem than the tuned case. Mistuning breaks-up the regular natural frequencies and mode shapes of a tuned rotor (nodal diameters and circles) and can lead to localized mode shapes where much of the modal motion is concentrated in a few blades. Furthermore, this localization is found to be severe in structures which have moderate coupling between the blades (the coupling may be aerodynamic or structural through the disk). A strong structural coupling between the blades refers to a relatively compliant hub, which results in a great deal of hub participation in the system modes. As a result of hub participation, dynamic energy is distributed more evenly throughout the assembly, which reduces the potential for mode localization. A weak structural coupling, referring to a relatively stiff hub, can also prevent large responses. This weak structural coupling can isolate blades to such an extent that dynamic energy put into each blade through aerodynamic forcing tends to remain in the blade instead of moving to other blades. The mode shapes in such assemblies tend to be very prone to mode localization. However, one should be careful to separate the localization of a mode from the localization of the forced response. The spatial distribution of the aerodynamic loading tends to be nearly orthogonal to the localized modes. Consequently, the forced response is not likely to become localized unless there is some, but not too much, coupling between the blades. Unfortunately, other restrictions on hub design often tend to place the interblade coupling in a range where the system is prone to localization. An attempt is made here to summarize the

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work being done in the area of forced response of bladed disk assemblies, with a focus on work which details the structural aspects of the problem.

Mistuning of blades can be caused by a number of factors, and can be classified as intentional or non-intentional. It can also be categorized as structural or aerodynamic. Although both are commonly referred to as mistuning, they are completely different phenomena. Structural mistuning is addressed in the this section, while aerodynamic mistuning is addressed in the following section.

Intentional mistuning is often used to reduce system sensitivity to non-intentional mistuning. This is done by generating a mistuning pattern that is dominant relative to nominal, non-intentional, mistuning, or expected future mistuning. Intentional mistuning is generally performed through the variation of blade geometry, although boundary condition (root or shroud) and material mistuning could potentially be used. Geometrical mistuning is the design tailoring of the blades in order to obtain desired frequency separation (or perhaps convergence). Since a great number of blade modes have the potential to be excited throughout the operational range of the engine, great care should be taken to consider the impact of geometrical mistuning on more than the target mode/s. Intentional mistuning has been found, in many cases, to be helpful in extending the flutter envelope. However, it has also often, but not always, been found to worsen the rogue blade phenomenon. Intentional mistuning should not be done without thoroughly considering its potential impact on all system modes within the operational envelope as well as its impact on sensitivity of the bladed disk assembly to non-intentional mistuning for all system modes within the operational envelope.

Non-intentional mistuning is the result of a great number of causes, including but not limited to: material variation, geometrical variation (i.e., variation within geometrical tolerances), FOD, thermally-induced geometrical variation, wear, field repair (especially in IBRs) and material failure (e.g., micro-cracks and debonding). In addition, material variation can include anisotropy, variation in density, and variation in material stiffness properties. As a result of their random nature, the analyses of these phenomena require the use of probabilistic and/or Monte Carlo approaches. The desired application of the Monte Carlo approach is perhaps the single greatest factor driving reduced-order modeling techniques.

Many reports and research papers refer to other types of mistuning. These are generally mass, stiffness, frequency, or mode mistuning. These descriptions should not be confused with causes of mistuning. They are, instead, interpretations of mistuning. For instance, the geometrical mistuning of adding mass to the tip of a blade will generally cause the reduction of the first blade natural frequency, and will increase the modal mass. Depending on the approach taken by the investigators, or the desired insight, this mistuning may be referred to as either, “mass” or “frequency” mistuning. Consider again the geometrical mistuning of thickening a blade near the first (closest to the root) nodal line of the third mode. Such a variation would likely manifest itself as a stiffening and increased mass to the first and second modes. The second mode shape would also change significantly. There would likely be little impact on the third mode. Thus, it manifests itself as the first three consequences of mistuning for the first mode, and all four in the second mode. While this sounds rather complex, the summary is that there are causes of mistuning (geometrical, material…) and there are interpretations of mistuning (generally spoken of in terms of modal parameters).

The result of mistuning can be forced response localization, also referred to as, “the rogue blade(s)” phenomenon. Consider first the unshrouded case. A brief interpretation of what is happening is that ideally, at one extreme, a bladed disk assembly would act as if the hub were a rigid, or fixed, boundary condition for each blade. In that case, the resulting motion of each blade is completely independent of the motion of adjacent blades. In the tuned case, each blade hits resonance at each of the blade natural frequencies; excitation comes from unsteady

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aerodynamics. Each blade, presumed to be identical, will have the same amplitude of motion, and thus, stress. At each natural frequency, each blade deforms in a mode shape identical to each of the other blades at the same natural frequency. The location of peak stress in each mode shape is considered the most likely location for fatigue to initiate. Operating conditions determine which frequencies are excited for the most cycles, and how micro-damage is accumulated, and at what locations. Each blade mode shape has a nodal line at its base, where no motion occurs.

At the other extreme, the hub is considered to be compliant, allowing significant energy to be transferred between blades. The nodal lines of zero deformation shift to the center of the hub, and motion of one blade propagates to the rest of the blades. In this case, the hub dominates the modal motion, with the majority of the motion occurring in the hub. Therefore, the modes are now system modes, and the symmetry of the system results in pairs of modes. The modes, at lower frequencies, come in groups of *n* (*n* being the number of blades) where the predominant blade motion is indicative of a blade mode with a compliant boundary condition. For instance, the first group is often referred to as, "first bending", since the deformations of each blade is in a form similar to that of the first cantilevered bending mode of a blade. Subsequent groups are often "second bending, first torsion", and so on, with the order dependent on the bladed disk design. At some point, the blade motions cannot be considered to be similar to those of a flat plate (for instance, a combination of torsion and bending). At higher frequencies, there is a greater tendency for a spread in the natural frequencies within groups. At this point, the modes are referred to as, "hub dominated", since the mode shapes are predominantly dictated by the hub, with the blades "along for the ride". At some point, the natural frequencies within groups begin to overlap with those of adjacent groups.

Each system mode within a group is comprised of a single type of blade motion (i.e., first bending…) at different amplitudes of motion. With an odd number of blades, *n*, the system has one symmetric mode with all blades moving in the same direction, along with (*n*-1)/2 pairs of repeated natural frequencies for a total of $m = (n-1)/2+1$ unique natural frequencies within the group. With an even number of blades, *n*, the system has one symmetric mode with all blades moving in the same direction, along with (*n*-2)/2 pairs of repeated natural frequencies, plus an additional mode where every other blade is deformed in the opposite direction. The result is a total of $m = 1+(n-2)/2+1$ unique natural frequencies within the group. The modes corresponding to the 2nd through the *m*th natural frequency each have $i-1$ nodal lines where i is the natural frequency number. Within each pair, the nodal lines are rotated 90º/(i -1) relative to one another. For an even number of blades, the final mode has $n/2$ nodal lines. If i symmetrically located excitations exist upstream (or in some cases, downstream) of a rotating bladed disk, then the response of the system is a travelling wave in the form of a linear combination of the two i th modes. Stress is thus evenly distributed amongst the various blades of the disk, just as in the first case. Excitations of more than $n/2$ cycles per bladed disk rotation do not excite system modal responses. That is, the response to more than $n/2$ upstream excitations do not undergo deformation in the form of a mode shape (as defined earlier). However, large responses can and do occur with a deformation periodicity which resembles that of the excitation.

Consequently, tuned forced response analysis can often be performed on the disk assembly using a single Degree of Freedom (DOF) concept where the so-called "blade frequency" may be used as the natural frequency of the bladed disk. In the latter case, the "blade frequency" should be found from a modal analysis of the entire bladed disk assembly. The symmetry of the bladed disk assembly can be used to simplify greatly the analysis in this case. In the past, most CFD work has assumed the former case, with "blade frequencies" sometimes obtained from the latter analysis.

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Real bladed disk assemblies have dynamics that lie between the two cases described. There is some coupling through the hub, but it is often not exceptionally strong. This is the worst of all cases. Each blade can act as a vibration absorber on the system. Dynamic energy can be slowly transferred from one blade to other blades through the hub. If the blades have slightly different natural frequencies because of FOD, DOD, manufacturing variation, or other occurrences, then a single blade or a set of blades may act as a vibration absorber for the entire assembly. The result is exceptionally-high amplitudes for those blades, which leads to blades that accumulate fatigue damage faster than would normally be expected.

In higher groups, when there is mistuning, the concepts of groups eventually breaks down, as the mistuning causes enough disorder in the blade deformation shapes that a mode cannot be defined as belonging to one group or another. For instance, some blades may have a predominantly bending deformation, while others may have a predominantly torsional deformation. In higher modes, the traditional meaning of mode localization breaks down, and instead modal lines and stress patterns look disorganized.

With the existence of shrouds, greater coupling between blades generally exists, with combined stiffness and energy dissipation (dissipation via contact friction). While the stiffness coupling generally enhances blade coupling and alleviates localization, the non-linear effects of the energy dissipation have been shown by at least one researcher to have the potential to induce the rogue blade effect, in theory.

d. Aerodynamic mistuning

Aerodynamic mistuning is the variation of aerodynamic loading, from one blade passage to the next. While structural mistuning is generally represented as variations of the mass and stiffness matrices in a finite element model, aerodynamic mistuning manifests itself as a variation of the forcing function, or blade loads, from the symmetric case. Bladed disk assembly models are generally linear, with the exception of shroud, root, and root dampers. As a result, a doubling of the applied load will cause a doubling of the system response. As a result, aerodynamic mistuning does not in itself have the equivalent potentially-dramatic impact on blade response as structural mistuning does. However, unlike structural mistuning, it will always have an effect, and that effect is proportional to its magnitude. In addition, the location and motion of shock waves which travel along blades can dramatically change blade response. At higher modes, the projection of blade pressures onto the blade modes (which yields blade-modal forces) is dramatically sensitive to both blade modal mistuning as well as distribution of aerodynamic pressure. Clearly, the mistuned analysis is a larger problem than the tuned analysis. Consequently, recent work has focused on ways to reduce the size of the model computationally while the essential characteristics of the forced response phenomenon are retained. While a great deal of effort has gone into deterministically obtaining mode shapes and natural frequencies from prescribed mistunings, the mode shapes of the bladed disk cannot be confused with the operational deformation shape and amplitude. For example, exciting a simple cantilevered Euler-Bernoulli beam at its first natural frequency with a pressure distribution of its second mode will yield zero response. Also, as a result of the close spacing of modes (or repeated frequencies), and mode localization, the greatest potential response may occur slightly off-resonance(s), where the sum effect of being near multiple resonances is greater than the response at a single resonance. Therefore, pure modal responses are not sufficient to determine operational speeds at which the maximum responses will occur, or what that maximum response is. Simulations should include multiple modal responses. Both the modal frequencies and mode shapes should be known well. When rogue blade analyses is performed, simulations should include multiple system modes for a wide-enough set of cases to obtain a statistically-significant peak response distribution over what are defined as allowable mistunings in the bladed disk assembly operation specifications.

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A number of potential reduced-order techniques exist. However, they are not well-tested by third parties. They should be used with caution. They need to be validated for each problem to which they are applied until their capabilities are more fully understood.

REQUIREMENT LESSONS LEARNED (A.4.13.3.2)

The requirement to be free of detrimental resonances within the engine operating range is based on how many developmental and field failures the Services have experienced in the past. Modern high-performance jet engines are designed for light weight, high temperatures, high efficiency, and high engine pressure ratios. The result is lightweight, flexible structures which operate at elevated temperatures and high rotor speeds. All these elements combine to make it impossible to design aerodynamic surfaces (blades and vanes) with resonant frequencies beyond the operating range. The designs now focus on management of the resonant responses to produce stresses within the material's capability from start through idle to maximum power, across the total inlet pressure (P_t2) and total inlet temperature (T_t2) envelope, and transiently from maximum power to idle and back.

The field experience includes instances of blade alone resonance, stator alone resonance, stator/blade interaction, bladed disk coupled resonance, integrally bladed disk coupled resonance, forced response of blades off resonance, forced response of stators off resonance, tuned absorber blade resonant response due to mistuning, disk arm resonant response due to mistuning, spacer resonant response due to duct blockage, etc. The lessons learned for the requirement fall into three broad categories: unsteady aerodynamic forcing functions including shock interactions, damped mechanical response of the structure, and material capability.

One example of a damaging unsteady aerodynamic forcing function occurred in a large fan that resulted in many engines and some aircraft suffering severe damage from blade releases. The inlet presented a strong, three-nodal-diameter pattern at the engine interface plane. This pattern, while predicable based on the inlet characteristics, was not quantified during the development program due to program constraints. Another engine experienced fan blade failures as a result of pressure waves which emanated from a downstream structure. Yet another engine experienced premature wear due to severe turbulence generated by an obscured inlet. Inlets should be characterized for the unsteady aerodynamic pressure and temperature patterns and strengths across the P_t2 and T_t2 envelope early in the design process.

Compressors have exhibited separated flow spillage from the variable inlet guide vanes either during transient operations or at specific operating points within the flight envelope. The advent of curved stators and non-uniform vane spacing are an attempt to reduce the cumulative effect of vane wakes on blades and effectively use the disharmony that results as equivalent to increased aerodynamic damping. Stator resonant failures have been the result of stronger than expected blade passing wakes coupled with reduced mechanical damping. Changes in mechanical damping in stators have generally resulted from design changes made after qualification to reduce manufacturing costs—replacement of an assembly by a one-piece casting is an example.

Blade failures in high-pressure turbines have occurred when shockwaves have been reflected from the downstream stators and produced high vibratory stress in the blades.

Low-pressure turbine blades are by necessity long and slender, which predisposes them to non-integral flutter and amplified resonances. Low-pressure turbine blade failures have occurred in the blade attachment that were the result of a complex combination of an unexpected system-type mode, friction in the attachment, and the orientation of the crystalline structure of the blade attachment. These failures occur on an infrequent basis as they tend to be the result of the low end of the material strength distribution, and the high end of the friction distribution along with some mistuning, which results in a stress at the high end of the vibratory stress distribution.

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They are very difficult to assess by test as they may occur only once in 1,000 to 5,000 blades. But, given the consequence of a catastrophic turbine failure, that is an unacceptable rate. Intentional mechanical damping is a necessary element of most HPT and LPT blade designs.

All of these failure modes can be exacerbated by small material defects—ceramic inclusions in turbine blades, recrystallized grains in single crystal superalloys, undispersed alpha particles in titanium alloys, foreign object damage, and manufacturing processes such as grinding. A process of screening designs for their ability to resist the growth of a small flaw (a =0.005 inch) has shown some success in accommodating more routine manufacturing processes.

Damping

Constrained layer viscoelastic material (VEM) damping systems have been successfully applied to static (non-rotating) engine components in the USAF fleet: specifically, a damping wrap applied to inlet guide vanes (IGV). Damping wraps, which consisted of five layers of VEM and aluminum foil for a total thickness of .016 inch, were bonded to the vanes during overhaul procedures on several fighter engine models starting in 1978 at Oklahoma City Air Logistics Center. Prior to installation of the damping wraps these IGV cases were experiencing high rates of HCF failures in vibratory modes excited at first-stage fan blade passing frequencies, at 28E (28 times engine speed). Maintenance costs associated with these failures were established at $3,500,823 per year. Virtually all HCF failures in the IGVs were eliminated after the implementation of the damping wrap. Five years after fleet implementation was complete, a total cost avoidance savings of $14,778,177 was documented [1]. These damping wraps remained in service until the particular fighter fleet was retired.

The design of the damping wrap involved careful evaluation of the temperature environment, modes shapes, frequencies, effects on engine performance, distortion tolerance, and durability. These issues were addressed in laboratory tests, engine test cell tests, and field evaluations [2]. The temperature range for optimum damping was 0ºF to 125ºF which accounted for 98 percent of the engine operation time in which HCF damage could be accumulated. The damping wrap also demonstrated it could survive 420ºF for the short intervals associated with anti-icing cycles. Modes of concern included the fourth torsion mode at 4000 Hz, fifth bending at 3600 Hz, and third torsion at 3600 Hz [1]. These were nominal frequencies—as they varied with engine model, temperature, and engine-to-engine variations. No significant change in engine performance, anti-icing effectiveness, or distortion tolerance could be measured. Durability of both the damping wrap and engine components proved satisfactory over several years in the fleet. Minor FOD damage to the damping wrap was repaired in the field by application of quick-set, two-part epoxy to exposed foil edges. Ground run-up test procedures were changed so that the anti-icing system was checked at part power rather than full power to avoid over-heating the damping wrap. It was found that checking anti-icing system at full power, with the engine static and no actual icing conditions present, could damage the damping wraps.

Similar damping systems were used on one-type engine inlet extension and another type's IGVs [1].

It has been demonstrated in spin pit testing on rotating fan blades that constrained layer viscoelastic damping systems (CLDS) which feature "floating constraining layers" are likely to fail due to centrifugal loads even if the damping system is encapsulated in a cavity in the blade. Modifications to the CLDS in which one edge of the constraining layer was structurally bonded to the blade were successful [3].

Testing has demonstrated that plasma-sprayed hard damping coatings exhibit non-linear behavior. Both stiffness and damping properties of plasma-sprayed coatings depend on the dynamic strain level in the coatings. This complicates the measurement and modeling of the behavior of these coatings [4-6].

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A.5.13.3.2 Component vibrations.

Verification of model validity, modal characteristics, vibration amplitudes, steady stresses, and all other aspects of the HCF problem should be performed at each step of the design and verification process. An integrated approach where each stage of the design/verification process builds upon the previous one should be utilized. Verification should include numerical verification (sensitivities to key parameters), and data generated in component bench testing, rig testing, engine testing, and, ultimately, operational use. Established methods to compare experimental and analytical results should be employed where possible. Probabilistic design margins and predictions should be validated with bench, rig, and engine test experience in addition to statistical comparisons to operating fleet databases. Assurance is to be provided by verifying that the probability levels for each contributing random variable used to compute probabilistic design margins or probability of failure are within the experimental data range for that variable.

VERIFICATION RATIONALE (A.5.13.3.2)

An integrated approach to verification insures maximum benefit is gained from each effort expended in the course of the design/verification process. Execution of the task at hand with an understanding of what subsequent phases will require maximizes usefulness of information acquired. This ultimately maximizes knowledge acquired during development and reduces overall development and life cycle costs.

VERIFICATION GUIDANCE (A.5.13.3.2)

A methodical systems engineering approach should be taken to understand fully the design and test parameters that should be undertaken to identify and resolve HCF issues within gas turbine engines. Those design and test parameters, as well as a checklist for test protocol item compliance, are presented as follow:

a) Design system: The holistic test and evaluation approach recommended herein begins with the contractor's design system. The manufacturer's design system defines the tools,

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margins, criteria, and material data used for the design of a gas turbine engine. Given that a definition of robustness is insensitivity to variation, the test protocol first recommends that numerical assessments, including probabilistic predictions, be made to bound the range of variation that will potentially be present in a component. This requires that relevant influence parameters be understood. Some influence parameters may be geometric variations, variations in boundary conditions, local environment and body forces (e.g., RPM), etc. Assessments such as these can be performed by “brute-force” or through use of techniques like eigensensitivity analysis. The latter has the advantage of being useful in the identification of model regions where modal frequencies may be especially sensitive to geometric variations. These results can be used to create specific models for parts that are off-nominal if geometric differences are known (by CMM, for example), or to correct/understand discrepancies between experimental results and nominal model results.

The manufacturer should insure structural models used for these studies are sufficiently representative of the actual structure. One way to address this is to perform a mesh density investigation to make sure that computed frequencies for normal modes do not change as a function of model discretization. Utilization of solid elements (isoparametric) with parabolic shape functions is recommended.

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	TEST PROTOCOL ITEM	COMPLIANCE?
I.	Design per Standard Work	
II.	Construct FEM	
	Solid Elements	
	Parabolic Displacements Functions	
III.	Perform Normal Modes Analysis	
	Mesh Density Assessment	
IV.	Sensitivity Assessment	
	Crystal Orientation	
	Geometric Variations (Eigensensitivity)	
	Boundary Conditions	
V.	Define Optimum Sensor Locations	
	Mode Measurement Capability	
	Modeshape Identification	
	Sensitivity to Sensor Misplacement	
VI.	Validate FEM	
	Frequency Comparison	
	Strain Ratio or Relative Displacment Comparison	
	Modal Assurance Criteria (MAC) or similar	
VII.	Compute Normal Modes at Speed	
VIII.	Define Limits for All Component Locations	
IX.	Design Experiment to Maximize Exposure to Influence Parameters	
X.	Test Rig and/or Engine	
	Process All Dynamic Data	
	Transform to Frequency Domain	
	Identify Modes Using Frequency and MAC	
	Apply Limits/Use FEM or FEM Derived Look-up Table	
	Database Results	
	Establish Statistical Variations from Database	
XI.	Assess Robustness	
XII.	Fix as-needed using Eigensensitivity to Move Problem Modes	

Partial ~
Yes ✓
No ✗

The result of a sensitivity conscience design or probabilistic approach is a basic understanding of potential design variations that may exist once the structure is manufactured. Also, this process generates the structural models used to assess uninstrumented locations and perform detailed analysis of experimental results under operational conditions in subsequent phases of the recommended protocol.

b) Component testing: The key focus of component tests generally performed on shaker tables or using stingers is to validate the analytical models generated above. The objective is to insure the finite element model can accurately capture the stress field for steady loads and modal responses. Component tests should be performed to determine experimentally modal frequencies and mode shapes for comparison to analytical results. Impedance tests can be conducted inexpensively to define experimental frequencies. Care should be take to

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insure that boundary conditions and physical characteristics (e.g., crystal orientation of anisotropic materials) of the part being tested are consistent with the model. Experimental and analytical frequencies should match mode-for-mode within 5 percent through the frequency range of interest. If they do not, the designer can utilize sensitivity data generated earlier to understand where problems may exist in the model.

If it is assumed sufficient agreement exists between experimental and analytical frequencies, attention should next be given to rigorous comparison of mode shapes. Utilization of a criteria such as the modal assurance criterion is recommended for this purpose. Accuracy of the MAC is increased as the number of measurements used approaches the number of degrees of freedom in the analytical model. For this reason, utilization of scanning laser vibrometry or other field measurement techniques are recommended. These methods are also non-intrusive. Again, care should be taken to insure that boundary conditions are sufficiently comparable between the model and the test.

Once it is verified that the analytical and experimental frequencies and modes are in sufficient agreement, the model can be trusted to describe the stress field at all locations on the component for any linear loading situation (steady or vibratory). Variations in stress which result from changes in boundary conditions can be assessed and operational effects can be addressed.

If the experimental and analytical results do not agree, every effort should be made to reconcile differences, since failure to do so indicates the analytically-predicted design intent will not be achieved.

Specifically in the case of mistuning assessments, benchtop experimental analysis of bladed disks can be subdivided into two types of testing: forced response and modal analysis.

Forced response analysis is excitation of the disk assembly in a spatial pattern comparable to that expected in operation. This means application of a travelling wave excitation of *m* cycles per revolution, *m* being the number of upstream excitations in operation per revolution (downstream, if back propagation of excitations is being studied). Excitation of this form has been generated in the past using screens, piezo-electric actuators, speakers, air jets, magnets, and fans. Careful control of the timing (for electronically-controlled actuators) or spacing (for spinning excitations) should be maintained to ensure true on resonant excitation. Slight variations can cause significant degradation of the desired results by impacting combined system modal responses. In general, multiple system modes are combined in the system forced response (mode localization could not occur without this happening). These modes are often highly sensitive to mistuning and, as a result, forced response analysis is useful to evaluate specific stress distributions determined analytically but cannot be used as a means of model correction when that model will be used for mistuned analysis, statistically or otherwise.

The term, “modal analysis” is used to represent the traditional systematic identification of a set of system modal frequencies and mode shapes. As discussed in A.4.13.3.1, modes in bladed disk assemblies are often paired with an orthogonal mode with the same natural frequency. Consequently, traditional Single Input, Multiple Output (SIMO) or Multiple Input, Single Output (MISO) modal testing techniques are not applicable in the ideal tuned case (they cannot pick up the repeated frequency and associated mode shape). As a result, Multiple Input, Multiple Output (MIMO) techniques such as Polyreference Time Domain (PTD), Polyreference Frequency Domain (PFD), or the Eigenvalue Realization Algorithm (ERA) are necessary, the latter being freely available from NASA Langley for U.S. citizen use in government, industry, or academia. In the mistuned case, SIMO or MISO techniques

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work in theory; but, in practice, modes are too closely-spaced within a group that they tend to be very difficult, if not impossible, to identify.

Results from the use of “rainbow wheels” to validate damping can be misleading with the existence of modal interaction. Damping itself can be a source of mistuning, and results obtained can be measures of energy transfer to the rest of the system instead of the desired energy dissipation. Ideally, “rainbow wheels” should be avoided. When used, great care needs to be taken to ensure the measured results are interpreted correctly. Damping that is applied to a single blade (i.e., not dependent on adjacent blade motion) can be accurately tested by detuning the blade sufficiently such that the test frequency for the blade is far enough from the system frequencies to avoid modal interaction. For a “rainbow wheel” with a large number of different dampers, this could necessitate a system FEM to determine whether system modes interact with any of the damped blade’s modes. When blade interactive dampers are tested, the band of blades need to be mistuned from the rest of the disk and experimental data need validate that: a) no modal interaction occurs between the group of damped blades, and b) the number of blades used is sufficient that extrapolation to a complete disk is valid. The group need demonstrate at least two special cycles of the same behavior in the middle of the group for such extrapolation to the entire disk to be valid. This corroborates that forward and backward blade interaction is fully developed for the center groups.

c) Subsystem testing: Once a physically-representative analytical model of the structure is available, it can be utilized to define instrumentation locations for use in tests where non-intrusive field measurements are not practical. Definition of these locations can be performed using “brute-force” or by utilizing optimization schemes. Locations selected for instrumentation should provide sufficient detection of responses and should also permit mode delineation by using MAC-like approaches. In order for the latter to be accomplished, more than one sensor should be applied to the component. Locations selected for instrumentation should also be insensitive to sensor application tolerances.

Once sensor locations have been defined, it is necessary to establish experimental limits. These limits serve two functions: to insure test article safety and to establish experimental success criteria. Again, since a physically-representative analytical model is now available, it can be used to assess the various combinations of steady and vibratory stresses at each location represented in the model and compare each of these to a fatigue criteria. Modal responses can be analytically-scaled until the fatigue criteria is met at the minimum margin location. The minimum margin location is easily identified since all locations are monitored. Predicted sensor output can then be “read-off” to define the limit.

Once instrumentation locations, test limits, and success criteria are established, instrumentation can be applied and tests can be performed in rigs or engines. Test conditions should be selected so that ranges of local influence parameters are sufficiently tested. For example, a turbine rig test should seek to test a sufficient combination of pressure and temperature as defined by the local P-T map. Other influence parameters that should be considered include RPM, distortion, inlet profile, or perhaps others.

High cycle fatigue responses should be compared to limit and success criteria defined previously. Data generated to assess component variabilities can now be used to prescribe corrections to the design should limits be exceeded.

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d) Engine test: Engine testing should proceed with the intent of expanding the exposure of each component to its range of influence parameters. Ensure all speed ranges possible are screened for potential detrimental resonances. These should consider variations in blade resonant frequency, variations in variable geometry, engine deterioration, and ambient temperature extremes. Above considerations will ensure both Min (idle governing speed) and Max physical speeds are tested. The sub-idle speed range should also be monitored. The same instrumentation locations used during subsystem testing, or a subset of them, should be used during engine test. Other tools, like the Non-interference Stress Measurement System (NSMS), are being developed to expand visibility despite the need to cover a large number of components during engine test. Also, because multiple components are usually tested concurrently during engine test, optimization techniques may be needed to define a minimum number of test conditions that expose each component to its maximum range of local influence parameters.

The volume of data recorded during engine tests, particularly during simulated altitude ground tests, can be quite extensive. High cycle fatigue failures have occurred which would have been avoidable had all the data acquired been processed and analyzed. Because of this, new technologies and systems that can process, analyze, and database all acquired data should be employed.

New tools developed for analysis of large volumes of engine test data make it possible to assess statistically the range of responses encountered as the local operational range of influence parameters is expanded. These assessments are naturally extendable to operational scenarios and form the foundation of enhancements to the design system, such as development of probabilistic methods. For example, these new statistical tools should be combined with probabilistic models for the system or component to validate structural modeling.

In addition to developmental engine tests aimed at characterizing component HCF margins, demonstration tests should be run. These tests are particularly useful for uncovering HCF problems on engine external components. The value as it relates to internal components is less clear, since there are so many discrete frequencies for each of the parts that it is nearly impossible to insure sufficient exposure to relevant modes. Nonetheless, current practice is to perform an up and down HCF stairstep test prior to the beginning of an AMT, and at the end of the AMT. These stairsteps are typically run at 50 RPM increments with, $5x10^5$ cycles (assuming a 3E driver) at each increment, which equates to 10^6 cycles in each 100 RPM band. In addition, the engine tests should include dwell time (10^7 cycles) at each of the critical speeds above idle, where the response in the rotor system dynamic verification testing shows peak values above the Goodman allowable. The procedures should be repeated for the other critical speeds below maximum speed if these critical speeds lead to maximum response at any point in the engine. Modes above and close to the maximum speed should be checked with the unbalance distribution required to excite these modes. If required, the phase of unbalance distributions should be changed to help determine residual unbalance.

e) Production/operational usage: Databases established during development or component improvement programs, when used in concert with production and operational use experience, have the potential to be used for health monitoring and/or prognostic approach development. All information resulting from “real” use should be used to enrich design and development tools by expanding understanding of response variabilities.

VERIFICATION LESSONS LEARNED (A.5.13.3.2)

The most significant lesson learned for this requirement is that for almost every field failure experienced by the USAF over the last decade, the test data showed the failure should not have

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occurred. This experience conclusively demonstrates that a deterministic approach to verification of HCF capability cannot succeed. One statistical study by a major engine manufacturer estimated a deterministic process (analysis and testing) could at best discover less than 40 percent of the HCF failures that would occur over the life of the program. The process needs to focus—beginning with the detail design and ending with the test of several engines—on development of the distributions of material properties, distribution of component geometries (manufacturing modes), distributions of modes and frequencies including high-order modes, P_t2 & T_t2 effects, etc. These distributions are then combined in models to develop the probability of failure. This approach recognizes the stochastic nature of the material strength, the component behavior, and the operational usage. Many new tools have made it possible over the past few years to develop these distributions—advances in material testing, in optical and eddy current procedures to measure the response of each blade in a wheel, and in scanning laser doppler measurement devices to observe the variations in mode shapes for traveling waves and aerodynamic velocities, along with high-speed processors that accommodate vast amounts of data—contribute to the quality and quantity of data needed to conduct these assessments.

A.4.13.3.3 Non-integral vibrations.

Self-excited vibrations such as flutter, separated flow vibration, or other non-integral vibrations should not occur within the operating range of the engine. In the specific case of flutter, the engine will encounter other limits before flutter. For example, stall boundaries should be encountered before stall flutter.

REQUIREMENT RATIONALE (A.4.13.3.3)

Catastrophic failures have been attributed to non-integral vibrations. While it is possible to limit the amplitude of these vibrations through the use of shrouds or other damping mechanisms, avoidance of them is the most effective approach.

REQUIREMENT GUIDANCE (A.4.13.3.3)

Non-integral vibration such as that encountered during flutter, buffeting, separated flow vibration, or similar forces should be strictly avoided. Sufficient damping should be available so that flutter boundaries are not encountered at any flight condition prior to engine stall.

REQUIREMENT LESSONS LEARNED (A.4.13.3.3)

The requirement to be free of flutter, separated flow vibrations, and other non-integral vibrations has proven to be the desired condition but one that is often not achieved in real engines. Most modern jet engines are subject to one, two, or all three of these conditions either transiently or steady state at some operating condition. These engines are serviceable either because the stress levels are within the material's capability or the exposure time in the region is very small. The fundamental nature of turbine engines provides the opportunity to encounter non-integral vibration in the turbine as the natural flow of work is from the fluid to the structure. Low-pressure turbine blades are normally the component most susceptible to flutter because they are relatively long and slender. Frictional damping is added to both high- and low-pressure turbines to provide stability for flutter. Fans are also susceptible to flutter due to the high mass flow and the need to operate at high stage loadings and efficiencies. Flutter in compressors has been relatively rare, as the operating line will often produce a stall in the aft stages before flutter occurs in the forward stages. Under certain operating conditions, separated flow vibration has developed in the front stages of a compressor and led to blade failures.

Advances in computation fluid mechanics, particularly associated with fluid-structure interaction, have yielded better understanding of the conditions that lead to blade flutter and the recent

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studies of the effect of mistuning on blade stability have expanded the design space for increased performance.

Separated flow vibration has been seen predominantly in two areas: where a fan is subject to a high level of turbulence from the inlet—not just total pressure distortion, and in first-stage compressor blades where the airfoil geometry in the inlet vane allows a separation area to form at certain conditions.

The have been instances when it was suspected that acoustic resonance within a confined space was contributing to an HCF failure. Extensive testing was conducted and although many acoustic resonances were measured, the energy levels were so low as not to be a factor. Destructive acoustic resonance is most often encountered within the combustor and augmentor and often leads to HCF failures on these structures.

Other instances of non-integral vibration have occurred where the precise aeromechanical mechanism for the failures was never identified. Some characteristics of flutter – traveling wave – three nodal diameters but reduced incidence and reduced velocity was in a region predicted to be stable. This has also been noted in other engines but the mechanism was never identified. All rotors tested showed some level of response. None were high enough to fail but some field failures had occurred. A probabilistic approach was used to extrapolate a few blades to a level to experience failure. Some research has shown a major impact on flutter capability can result from small variations in mode shapes, while little impact was seen for frequency variations. The use of similar probabilistic methods used for resonant vibrations for non-integral vibration is warranted.

A.5.13.3.3 Non-integral vibrations.

Verification of sufficient flutter margin should be by analysis and test. To the maximum extent possible, prediction of flutter boundaries using physics based models should be performed prior to engine test. If physics based models are unavailable, empirical models or databases should be used. Analytical predictions should be verified using engine tests under simulated altitude conditions. The effects of mis-rigging and engine deterioration should be evaluated to insure that flutter is not encountered during start-up or at other operational speeds at any flight condition.

VERIFICATION RATIONALE (A.5.13.3.3)

It is currently not feasible to design rotor systems free of self-excited vibration using predictive methods alone. Consequently, use of both analytical and experimental methods is necessary to verify the engine is free of these vibrations and to insure safe operation.

VERIFICATION GUIDANCE (A.5.13.3.3)

Definitions of flutter and flutter boundaries are abundant, but are most often captured using three graphical presentations. These presentations are provided on figures 6 and 7. Each of these figures present the boundaries in different forms and also provide information regarding critical parameters that should be evaluated, either analytically or experimentally, to insure flutter-free operation.

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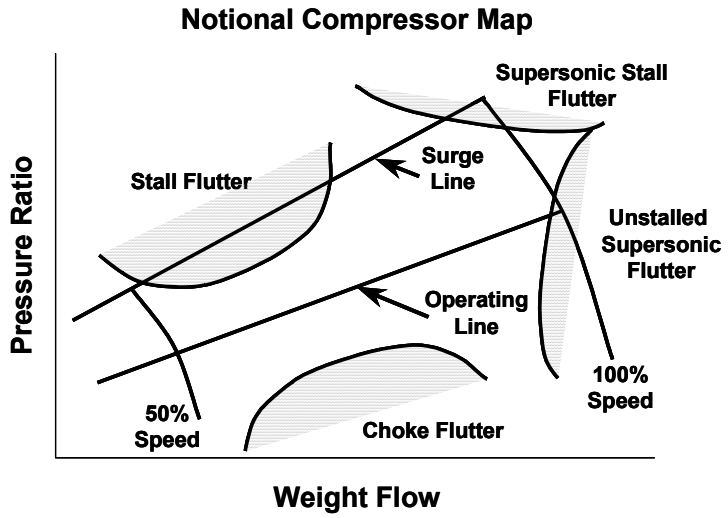

FIGURE 6. Notional compressor map showing flutter boundaries.

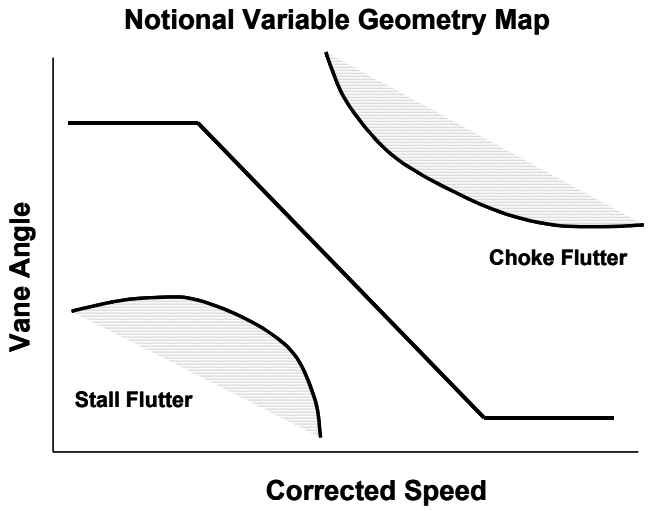

FIGURE 7. Notional variable geometry map.

Verification of flutter margin can be accomplished by investigation of the various influence parameters which establish the boundaries of the preceding figures. This can be done analytically or experimentally. Recent experience has shown that investigation of the effects of engine deterioration and mis-rigging of variable geometries is necessary.

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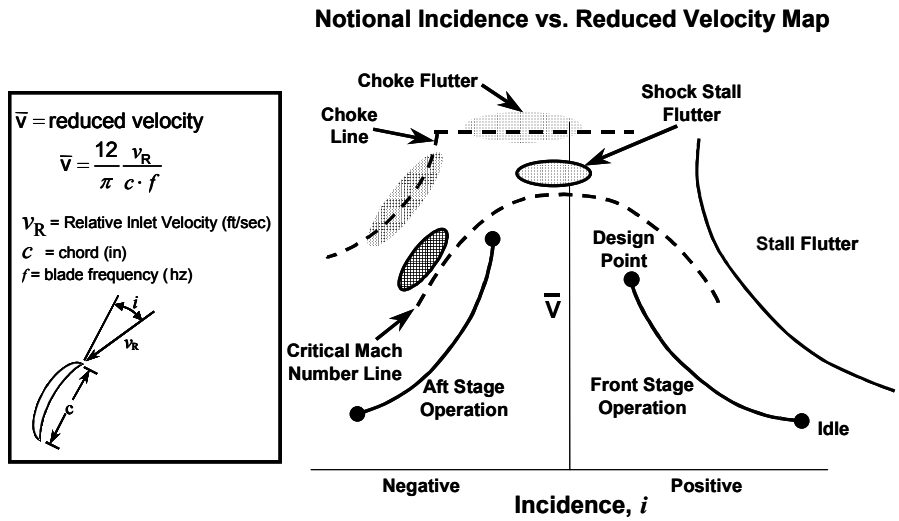

FIGURE 8. Reduced velocity versus incidence map.

The effect of engine deterioration has been shown to produce migrations of the flutter boundaries by shifting work-split among the various stages of a compression system. Work split changes as the engine control attempts to maintain overall compression system performance. The net effect is that stages which do more work have increased incidence and stages which do less work have decreased incidence. The same effect can be realized by changing flight conditions. Flight conditions which produce increased work and consequently increased incidence will produce the same result. The migration described is captured on figure 9. On the incidence versus reduced velocity graph, the result is a movement in the operating line toward or away from flutter boundaries.

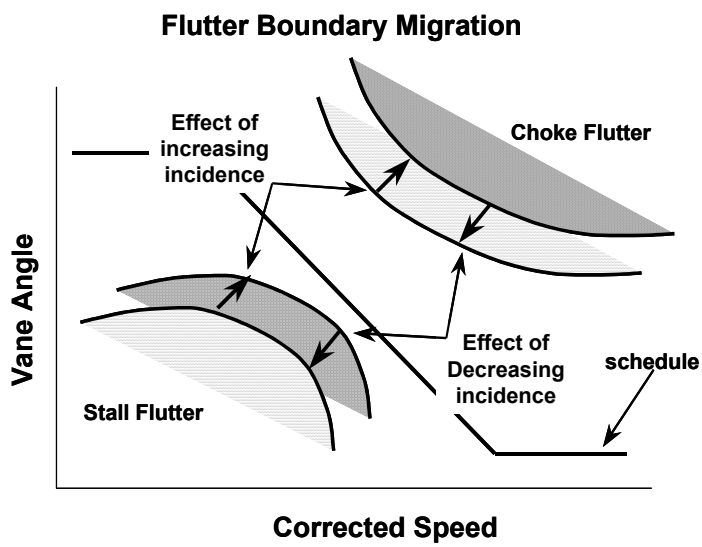

FIGURE 9. Flutter boundary margin.

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VERIFICATION LESSONS LEARNED (A.5.13.3.3)

The traditional process to verify the capability of an engine for flutter, separated flow vibration, and non-integral vibrations (non-classical flutter) using a combination of analysis and limited aeromechanical testing has not been totally successful. Developmental, qualification testing, and field-service failures have occurred. These events, when a root cause was determined, have been found to result from normal variations of mode shape, inlet guide vane schedules (especially during transients), operating envelope, and material capabilities. As with resonant responses, self-excited vibrations need to be addressed in a probabilistic process which considers the range of reasonable excursions in mode shapes, aerodynamic parameters, frictional dampers, and inlet total temperatures and total pressures along with inlet turbulence levels. Integrating these variations with the test process using strain gages and optical or eddy current sensors to establish blade responses and appropriate analytical tools which combine computational fluid dynamics and structural dynamics provide the greatest likelihood these issues will be identified during the development program.

It is now clear that it is not sufficient just to demonstrate a stable operating line within the classical flutter boundaries between stall flutter and choke flutter (either subsonic or supersonic). Rather, it is necessary to demonstrate the susceptibility of the design to normal variations in the relevant parameters and tailor the particularly susceptible modes to retain aerodynamic damping or to insert sufficient damping mechanically to maintain aeroelastic stability.

Flutter boundary migration has proved to be a problem on several fighter-class engines. Significant variations in flutter boundaries have been documented as a function of flight condition. Increased P_2 and T_2 have tended to move stall flutter boundaries toward the variable geometry schedule and, when combined with deterioration, have resulted in loss of all flutter margin. Front stages of high-pressure compressors seem to be most susceptible. Mis-rigging or other drivers can result in non-integral vibrations which have been blamed for some HCF failures near idle speed. This highlights the need to investigate the potential for non-integral vibration low in the speed range.

Verification of the strength of acoustic resonance is particularly difficult, as a simple closed chamber (except for intentional leakages) can have many acoustic modes present simultaneously. The presence of multiple modes may in fact reduce the energy available for any single mode and reduce the interaction with the surrounding structure. With the exception of the combustion chamber and augmentor, verification of the strength of acoustic resonance would require either analytical or experimental evidence that such a resonance would likely result in significant stress on the structure. Specific instrumentation and location are critical to verify the acoustic and combustion instabilities that may be present in combustors and augmentors.

A.4.14 Noise.

The engine will meet the strength and design service life requirements in the presence of the noise environment produced during installed and uninstalled operation at the flight and ground operating conditions consistent with the design usage conditions.

REQUIREMENT RATIONALE (A.4.14)

Maintenance and cost considerations require the engine structure be resistant to sonic fatigue problems. Airframe/engine configuration and test facility acoustics can produce large effects in the acoustic noise levels, and drastically affect the cracking characteristics of exhaust/nozzle components.

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REQUIREMENT GUIDANCE (A.4.14)

Acoustic loads should be considered in the design of exhaust/nozzle components such as stiffeners and fairings. Extra margin in terms of lowered stress levels and increased thickness should be provided where practical.

REQUIREMENT LESSONS LEARNED (A.4.14)

More attention to sources of loading and structural design and evaluation of structures subjected to acoustic fields is required. Exhaust/nozzle component problems have resulted in a significant maintenance burden on some legacy USAF engine systems.

A.5.14 Noise.

The capability of the engine to meet the strength and durability requirements in the presence of the noise environment generated during engine operation will be verified by test. Specific tests required by this document that will be used to demonstrate compliance with the noise requirement of 4.14 will be as follows: _____.

VERIFICATION RATIONALE (A.5.14)

Determination of the magnitude of the various acoustic levels is required to assess adequacy of the design and to determine where design changes may be required. Inspection of engine structure during periodic intervals will allow determination if this operational requirement is met.

VERIFICATION GUIDANCE (A.5.14)

Acoustic measurements should be made during operation in the test cell at various conditions. Analysis of the data should be made to estimate if pressure levels are of sufficient magnitude to cause structural cracking. Inspection of AMT engines of A.5.9.1.1 through A.5.9.1.4 should be used to verify resistance to component structural cracking.

VERIFICATION LESSONS LEARNED (A.5.14)

None.

A.4.15 Foreign object/domestic object damage (FOD/DOD).

The engine will operate satisfactorily when foreign objects/domestic objects are ingested.

REQUIREMENT RATIONALE (A.4.15)

Engines frequently experience damage to fan blades and other airfoils when foreign or domestic objects enter the flow path. Objects include ice, gravel, sand, and nuts and bolts or other retaining mechanisms that come loose. Structures subject to this type of damage need to be able to operate to the next subsequent depot interval to avoid the requirement for immediate teardown when the damage is detected and determined to be within acceptable limits. Therefore, design criteria are required to establish capability of both fracture-critical parts and durability-critical parts to operate with damage present. The previous design requirement for an engine to be able to withstand a FOD incident where the damage size and shape had a $K_f = 3$ has been replaced with a probability of failure criteria. The intent of this change is to encourage industry to design impact-resistant blades to provide engines better protection from failure and to allow industry to take design credit for component geometries inherently resistant to impact.

REQUIREMENT GUIDANCE (A.4.15)

The engine will be able to operate for one depot inspection interval with a probability of failure less than 1×10^{-7} per EFH per stage, based on statistical FOD distributions supplemented by

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analysis, testing, and appropriate component stress state distribution (mean, vibratory, residual, etc.) as described in section A.5.6.

REQUIREMENT LESSONS LEARNED (A.4.15)

Studies by the USAF have shown that for in-service engines, up to 30 percent of FOD sites in the first three stages will have a k_t higher than 3. While k_t does not equal K_f, the two are similar enough to prove a significant percentage of engine blades will experience damage of this size. While previous designs were based on either k_t or $K_f = 3$, most blades do not fail when exposed to this size of FOD in service. This is more than likely due to the lack of sufficient vibratory stresses to propagate damage from those blade locations. In other words, simple geometric damage factors such as k_t or K_f do not account for stresses in the damage zone. Further, the high number of FOD incidents that have a $k_t > 3$ imply this number alone is insufficient to ensure safety of flight. The following figure from AFRL-ML-WP-TR-2001-4159, "Improved High Cycle Fatigue (HCF) Life Prediction," depicts in-service FOD data.

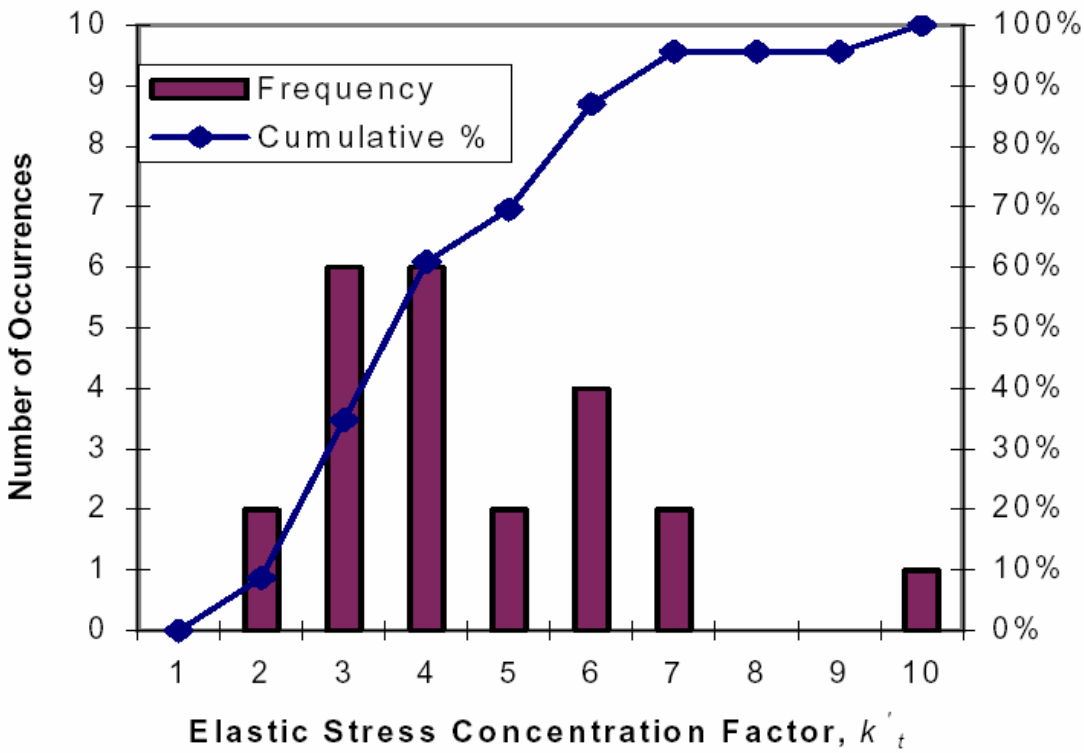

FIGURE 10. Characterization of in-service FOD data.

Foreign object damage sites with K_f less than 3 have been shown to cause failure in in-service engines. It is thought that the failure of these blades was due to residual stresses and damage within the FOD damage zone or sufficient vibratory stresses at the FOD location. Because of the wide variation in damage due to FOD, a criterion such as $K_f = 3$ may not capture the effect of the entire range of possible impact conditions. In order to capture the component fatigue life reduction with sufficient accuracy to enable the use of probabilistic design criteria, it is necessary to perform realistic component impact tests. These tests should be supplemented by computational modeling of the impact event and the resultant damage morphology to enable assessment of the full range of impact conditions.

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A.5.15 Foreign object/domestic object damage (FOD/DOD).

Evaluation of the capability of the engine to meet the foreign object/domestic object damage requirements will be by analysis and test.

VERIFICATION RATIONALE (A.5.15)

Analysis and tests are required to demonstrate that the fan and compressor airfoils can meet the operational requirement of 4.15 and to establish accept/reject criteria for damage detected during flight-line inspections.

VERIFICATION GUIDANCE (A.5.15)

It is recommended that LCF and residual life analysis be performed. A test engine will be subjected to a foreign object damage test to demonstrate compliance with A.4.15. Simulated foreign object damage will be applied to three (3) airfoils of the most critical stage of both the fan and compressor. The damage will be located at the most critical areas susceptible to foreign object/domestic object damage (i.e., at the most limiting vibratory threshold crack size location on the airfoil, when the combination of steady stresses and vibratory stresses that occur at each sustained power condition is considered). The applied damage will produce a minimum fatigue notch factor (Kf) of 3. The engine test will be conducted to an equivalent of two (2) depot inspection intervals which simulate the design duty cycle of A.4.3. No calibration or recalibration will be required for this test. The test will be considered to be completed satisfactorily if no blade separations have occurred during the test.

Subject to approval of the Procuring Activity, the foreign object damage test may be conducted by bench testing or rig testing on full-scale fan or compressor components in lieu of complete engine testing. However, the bench or rig tests need to meet the conditions, duration, and severity of testing equivalent to the engine test described above.

VERIFICATION LESSONS LEARNED (A.5.15)

None.

A.4.16 Structural maintainability.

The engine will be economically maintainable for the design service life and design usage of 4.3. Engine components will fit and function with new components after being operated to the design service life and design usage of 4.3. The function of structural components, elements, and major bearing surfaces will not be degraded by wear, erosion, or corrosion to the extent that performance or structural capability will be impaired. Authorized repairs will be established for critical components that experience detrimental wear, erosion, or corrosion during developmental testing and service operation. The structural life of repaired components specified by the contractor will be equal to or greater than the inspection intervals set forth in 4.8.4. Any repairs need be structurally sound and cost effective.

REQUIREMENT RATIONALE (A.4.16)

It is imperative the engine and its components be designed to be repaired and maintained in the most cost-effective manner. Salvage of the engine and its components from the deleterious effects of wear, corrosion, creep deformation, fatigue cracking, oxidation, erosion, and handling during operation needs to be accounted for in the basic design.

REQUIREMENT GUIDANCE (A.4.16)

Structural design of parts will be such that after one lifetime of use, they will fit and be functional with like, new parts. Components will be designed with allowances for repair when possible and repair life will be defined and be at least capable of two planned depot maintenance periods; e.g., coating systems.

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Repairs will be established for typical modes of deterioration that can be expected to occur during extended operation and for deterioration observed during development testing. Examples of typical repair needs that should be developed are as follow:

Restoring snap diameters

Blade-blending of FOD/DOD

Blade-coating strip and recoat

Removal of hole damage (oversize and bushing)

Restoring compressive surface stresses at areas of galling/fretting by shot peen.

REQUIREMENT LESSONS LEARNED (A.4.16)

None.

A.5.16 Structural maintainability.

Maintainability of the engine will be verified per 5.16.a and 5.16.b.

A.5.16.a Inspection.

Inspection and evaluation of changes in critical dimensions and finish of components after conduct of the several engine tests detailed by this handbook. A maintainability assessment plan will be developed and implemented.

VERIFICATION RATIONALE (A.5.16.a)

Maintainability is best assessed by evaluation of component condition after extended operation in the engine.

VERIFICATION GUIDANCE (A.5.16.a)

Critical component dimensions should be measured before test (during assembly) and after test, and the differences compared to analytical growth predictions. These components should be able to fit and function with parts.

VERIFICATION LESSONS LEARNED (A.5.16.a)

None.

A.5.16.b Test.

Structural life of component repair procedures will be verified by test, as required.

VERIFICATION RATIONALE (A.5.16.b)

Adequacy of repaired components is best assessed by component condition after extended operation in the engine.

VERIFICATION GUIDANCE (A.5.16.b)

Repair should be evaluated by AMT tests for the desired, specified period of time.

VERIFICATION LESSONS LEARNED (A.5.16.b)

None.

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A.4.17 Inspectability.

Critical engine components will be inspectable by use of borescope ports and diagnostic methods so that detrimental damage or other deterioration will be detected to facilitate economical repair and to prevent engine failure. A listing of the inspectable components and their methods of inspection will be specified.

REQUIREMENT RATIONALE (A.4.17)

Experience reveals that installed and uninstalled inspection capability for critical components in the assembled engine is tremendously vital to field-level maintenance and service operations. The capability to inspect for FOD/DOD and hot section airfoil distress without engine disassembly has been extremely important in past service operations.

REQUIREMENT GUIDANCE (A.4.17)

Emphasis should be placed in design on the attainment of the maximum possible degree of inspectability. Provisions for inspection of the installed engine by borescope (or equivalent devices) will be made for the fan, compressor, combustor, and turbine sections of the engine. The goal is that each rotor and stator stage be inspectable. Inspection of the combustor and the turbine blades and vanes in the installed engine is required. A positive means to rotate the rotor system(s) slowly will be provided to facilitate borescope inspection. Radiographic inspection capability should be provided for the completely-assembled engine. Location of the inspection provisions should ensure part access and radiographic access for the installed engine. The contractor should define in an appendix to this handbook or in the ENSIP Master Plan the design objectives for inspectability, the inspectable components, and the methods of inspection. Inspection provisions, including access envelope, will be shown on the engine configuration and envelope figure. The design objectives for inspectability should include special development of inspection methods if event development testing indicates a mode of deterioration or distress that is not inspectable in the installed engine.

Diagnostics in the form of blade metal temperature sensors (e.g., optical pyrometer), oil analysis methods, and bearing-mounted accelerometers will be utilized during the development program to reduce risk of engine failure. The diagnostic capability of these sensors will be developed with the engine and will be designed into the engine; i.e., externally-removable internal sensors. External vibration sensors location will be selected to maximize vibratory response from the engine.

Components and methods should be listed in table XXII, as follows:

TABLE XXII. Components and methods.

COMPONENT	INSPECTION METHOD	INSPECTION INTERVAL

REQUIREMENT LESSONS LEARNED (A.4.17)

Installed inspection capability for turbine stators (vanes) has historically not been provided since it requires some kind of travelling probe which is extremely difficult to develop. However, past experience has shown that when continued safe operation depends on installed inspection

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capability for a component (e.g., turbine vane), a strong design and development program often comes up with the inspection method. It is possible that, given more attention during design, inspection methods can be developed during initial design for static structures susceptible to deterioration (i.e., turbine nozzles and vanes).

Engine failures during development are very costly in terms of schedule and resources and every effort should be made to detect impending failure prior to loss of the test article. The use of diagnostics has been successful in the elimination of development engine failures.

A.5.17 Inspectability.

The ability to accomplish inspection requirements established by 4.17 will be verified during conduct of the engine tests detailed by this handbook.

VERIFICATION RATIONALE (A.5.17)

Adequacy of inspection methods requires repeated application to the assembled engine.

VERIFICATION GUIDANCE (A.5.17)

Each inspection method developed for the engine should be employed during routine inspections of development engines. Any deficiencies discovered should receive design attention as early as possible so improvement can be made prior to engine flight operation.

VERIFICATION LESSONS LEARNED (A.5.17)

None.

A.4.18 Engine/airframe structural compatibility.

The engine will meet the structural requirements of this document when installed in the airframe. The installed engine will operate satisfactorily in the thermal and aerodynamic environment produced by the engine/airframe configuration. The installed engine will possess flutter margin throughout the engine flight envelope.

REQUIREMENT RATIONALE (A.4.18)

The engine and airframe need be dynamically, functionally, structurally, and thermally compatible.

REQUIREMENT GUIDANCE (A.4.18)

An Interface Control Document between the engine and airframe contractors will be utilized to ensure functional and structural compatibility. Engine mounting will be such that critical engine clearances are not adversely affected under flight loadings. Aircraft flight and ground loads on the engine static and fatigue spectrum loading will be supplied in the ICD. These loads will include external airloading on the engine, if applicable. The engine contractor will supply to the airframe contractor inlet duct stall loads. The engine will be designed to withstand the distortion-induced vibrational loads associated with the air vehicle inlet for all operational conditions. The engine/nacelle cooling will be designed so installed engine temperature variations do not adversely affect engine critical clearances.

REQUIREMENT LESSONS LEARNED (A.4.18)

None.

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A.5.18 Engine/airframe compatibility.

Engine/airframe compatibility will be verified by an instrumented engine test installed in the aircraft. The scope of these tests will be contained in the Interface Control Document.

VERIFICATION RATIONALE (A.5.18)

Installed engine ground and flight testing is the only method to verify compatibility.

VERIFICATION GUIDANCE (A.5.18)

It is recommended the scope for these tests include fan stress survey; nacelle temperature survey; vibration survey of the aircraft mounting structure, controls, and accessories during rotor imbalance; and evaluation of clearances and deterioration. These tests should be integrated with the aircraft flight loads survey program to minimize aircraft/flight time test requirements. The flight survey will be conducted at all specified operating conditions, both engine and aircraft, within the aircraft flight operating envelope—including takeoff, transition, climb, descent, landing, altitude restarts, maximum yaw, and mission flight maneuvers. The investigation, where applicable, will explore engine stress conditions during maneuvers consistent with the mission of the aircraft, ordinance firing, thrust reverser operation, augmentation, and during any other unusual maneuver or mode of operation peculiar to a particular aircraft system which could have an effect on engine dynamic vibratory characteristics. Details of the instrumentation, such as strain gauge locations, instrumentation ranges, responses, recorders, etc., will be set forth in the approved test plan.

VERIFICATION LESSONS LEARNED (A.5.18)

None.

A.4.19 Component life management.

Required maintenance actions (component inspection, repair, or replacement requirements) will be defined to ensure adequate structural integrity and operational readiness of each engine for the design service life. Required maintenance actions will be based on duty cycles defined by operational usage of the airframe/engine. Individual component maintenance times will be based on the parameter that causes life degradation.

REQUIREMENT RATIONALE (A.4.19)

Required structural maintenance actions need be defined so the U.S. Air Force can maintain the structural integrity and operational readiness of the engine systems.

REQUIREMENT GUIDANCE (A.4.19)

Identification of the required structural maintenance actions will be based on the results of the analyses and tests detailed by this handbook. Required actions will include component inspection, repair, or replacement needs. Detailed inspection requirements will be included relative to the component to be inspected, location(s) on component, inspection method, and inspection interval. Required actions will be initially defined based on verification tasks that utilize the design service life and design usage. The actions will be updated to reflect the results of verification tasks that utilize operational data as detailed by this handbook. Finally, individual maintenance times will be based on the most significant parameter that influences life degradation (e.g.; 0-max-0 throttle cycles, hot time, engine operating time).

It is recognized that only the initial structural maintenance plan per A.5.19.a will be completed as part of the full-scale engineering development (FSED) contract and that the requirement for an updated plan based on operational data and subsequent endurance test will be contracted tasks under follow-on Component Improvement Program. However, it will be an FSED

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requirement to develop programs to gather operational usage data and to establish an individual engine tracking program per A.5.19.b and A.5.19.c.

REQUIREMENT LESSONS LEARNED (A.4.19)

None.

A.5.19 Component life management.

Component life management will be defined and implemented by analysis, test, and recording of the operational usage of the engine as follows:

A.5.19.a Plan.

A structural maintenance plan will be prepared.

VERIFICATION RATIONALE (A.5.19.a)

Required structural maintenance actions are generated from several analyses and tests conducted during development of the engine. It is necessary that the various actions be consolidated into a single plan for input to the overall engine maintenance plan. The information contained in this plan will also be used to support evaluation of the engine for production.

VERIFICATION GUIDANCE (A.5.19.a)

The structural maintenance plan will be initially prepared to reflect the status of analyses and tests completed as part of full-scale development. Later, the structural maintenance plan will be updated to reflect results of analysis and accelerated mission testing of the production configuration to a duty cycle which utilizes operational data of A.5.9.1.4. Additionally, the plan will be kept current to identify required structural maintenance actions for design changes incorporated in production.

VERIFICATION LESSONS LEARNED (A.5.19.a)

None.

A.5.19.b Data recording.

Engine signals will be provided to the airframe data recording system to record parameters required to establish operational usage duty cycles for the engine. The data recording system will record the following parameters: _____.

VERIFICATION RATIONALE (A.5.19.b)

Usage and engine parameters critical to structural limits need be monitored during aircraft/engine operations so the design duty cycle required by A.5.3 can be updated. The updated duty cycle is then used in analyses and tests to define engine structural characteristics and maintenance actions. The intent is to record continuous time histories of multiple parameters on a percentage of operational engines so that a statistically-based definition of mission profiles and usage can be established.

VERIFICATION GUIDANCE (A.5.19.b)

The recording system will be able to monitor mission profile parameters such as rotor RPM, power level angle (PLA), engine inlet temperature, and turbine temperature. Small interval sampling of the measured parameters is needed so that frequency distributions can be established (e.g.; PLA level, intermediate and idle dwell time, ground operation time, acceleration/deceleration rates, etc.). The number and selection of airframes and engines which require recorders will be established so that sufficient data is available within three (3) years after initial operational capability to validate operational usage. Significant

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factors to be considered in the analysis include planned flying rate, airframe/engine production rate, and number of bases.

VERIFICATION LESSONS LEARNED (A.5.19.b)

None.

A.5.19.c Counter.

Each engine will contain a counter which will record parameter events that control the structural limits of engine components. The counter will record the following events: _____.

VERIFICATION RATIONALE (A.5.19.c)

The intent is to place a simple, highly-reliable counter on each engine that will record occurrences of parameter events that dominate the rate at which structural life is consumed (e.g.; 0-intermediate/max-0 cycles, idle-intermediate/max-idle cycles, hot time, etc.). Experience has shown that these events can vary significantly between individual engines and that accrual of damaging events need be tracked at the component level to establish when maintenance actions are required.

VERIFICATION GUIDANCE (A.5.19.c)

As a minimum, the engine counter will record: (1) engine operating time, (2) time at or above intermediate power, (3) number of 0-max/intermediate-0 throttle events, and (4) number of idle-max/intermediate-idle throttle events. The capability to record other types of throttle events will be provided in the event the usage sensitivity study shows these events significantly affect life consumption. Consideration will be given to the location of the recording device to minimize exposure to thermal and vibration environment.

VERIFICATION LESSONS LEARNED (A.5.19.c)

None.

A.5.19.d Tracking program.

A critical component tracking program plan will be established. This system will define the analysis procedures, serialization, data collection, and computer programs necessary to establish maintenance times of individual components based on accrual of parameter events.

VERIFICATION RATIONALE (A.5.19.d)

A tracking system is needed to establish individual engine and component maintenance times. The rates at which equivalent 0-max-0 throttle cycles and hot time are accrued will vary significantly depending on such variables as base of operation and mission type. Accrual of events that dominate the rate of life consumption need be tracked at the component level for efficient management of life-limited components.

VERIFICATION GUIDANCE (A.5.19d)

The tracking system needs to have the following features: (1) a simple, reliable device that records damaging events, (2) a data retrieval system that provides for transcription of the recorded data from the engine and recorder, and transmittal of the data to a central computational facility, (3) computer software programs that provide a summation of the number of damaging events for each component based on damage analysis equations and recorded data, (4) a serialization procedure to enter components into the tracking system, (5) a procedure to recall engines, modules, or components when individual structural limits have been attained, and (6) a procedure to identify components that have received required maintenance actions.

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VERIFICATION LESSONS LEARNED (A.5.19d)

None.

A.6 NOTES

A.6.1 Data requirements.

When this handbook is used in an acquisition which incorporates a DD Form 1423, Contract Data Requirements List (CDRL), data requirements substantially like those identified below will be specified by a newly-created or an existing Data Item Description (DD Form 1664) and delivered in accordance with the approved CDRL incorporated into the contract. When the provisions of Design Analysis Report 7104.9(n)(2) are invoked and the DD Form 1423 is not used, data such as that specified below will be delivered by the contractor in accordance with the contract or purchase order requirements. Dependent upon what requirements are actually specified in the acquisition documents, deliverable data may be required from the following paragraphs:

PARAGRAPH NO.	APPLICABLE DATA REQUIREMENT TITLE	DID NUMBER	OPTION
4.2 and 5.2	ENSIP Master Plan		
5.3	Design Usage (Design Duty Cycle)		
5.3.1	Engine Hot Parts Analysis and Test Plan		
5.6	Material Characterization Plan		
5.8	Damage Tolerance Control Plan		
5.8	Damage Tolerance Analysis		
5.9	Durability		
5.9.1	Low Cycle Fatigue (LCF) Life Test and Analysis		
5.9.1.1	Accelerated Mission Test (AMT)		
5.9.2	High Cycle Fatigue (HCF) Life		
5.9.3	Corrosion Prevention and Control Plan		
5.10	Strength (Structural analysis and part, component, and engine tests)		
5.10.3	Containment Analysis		
5.12	Creep Analysis and Design Development Test Plan		
5.13.1	Dynamic Analysis of Engine Summary		
5.12.b	Mechanical Impedance Test Plan		
5.15	FOD/DOD Analysis and Test Plan		
5.16	Maintainability Assessment Plan		
5.19.a	Structural Maintenance Plan		
5.19.d	Component Tracking Program Plan		

Custodians:
Army – AV
Navy – AS
Air Force – 11

Preparing activity:
Air Force – 11
(Project 15GP-0068)

NOTE: The activities listed above were interested in this document as of the date of this document. Since organizations and responsibilities can change, you should verify the currency of the information above using the ASSIST Online database at www.dodssp.daps.mil.