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METRIC (SI)/ENGLISH

GUIDANCE ON STRENGTH, FATIGUE, AND FRACTURE CONTROL REQUIREMENTS FOR ADDITIVELY MANUFACTURED SPACEFLIGHT HARDWARE

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FOREWORD

This Handbook is published by the National Aeronautics and Space Administration (NASA) to provide engineering information; lessons learned; possible options to address technical issues; classification of similar items, materials, or processes; interpretative direction and techniques; and any other type of guidance information that may help the Government or its contractors in the design, construction, selection, management, support, or operation of systems, products, processes, or services.

This Handbook provides guidance for meeting the intent of NASA requirements related to structural qualification of additive manufactured spacecraft hardware in the areas of strength, fatigue, and fracture control. Guidance on background, details, and execution of these core activities is contained throughout this NASA Technical Handbook.

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Original Signed by:	2024-08-12
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GUIDANCE ON STRENGTH, FATIGUE, AND FRACTURE CONTROL REQUIREMENTS FOR ADDITIVELY MANUFACTURED SPACEFLIGHT HARDWARE

1. SCOPE

1.1 Purpose

This NASA Technical Handbook provides guidance on implementation of NASA requirements for Additively Manufactured (AM) spaceflight hardware in the areas of strength, fatigue, and fracture control. AM hardware owners should consider all requirements in these areas contained in NASA-STD-5001, Structural Design and Test Factors of Safety for Spaceflight Hardware, NASA-STD-5009, Nondestructive Evaluation Requirements for Fracture Critical (FC) Metallic Components, NASA-STD-5012, Strength and Life Assessment Requirements for Liquid-Fueled Space Propulsion System Engines, and NASA-STD-5019, Fracture Control Requirements for Spaceflight Hardware. This Handbook provides guidance for how to meet the intent of these technical standards when dealing with an AM part; the methodologies described may be developed and submitted to the delegated NASA Technical Authority (Technical Authority has been addressed in NPR 7120.5 document) for approval. The intent also is that this Handbook will serve as a primary location for guidance on strength, fatigue, and fracture control requirements for AM parts and, as such, includes some content that may be partially generic in nature and found in other NASA documentation. Should a conflict arise between this Handbook and a NASA Technical Standard, the technical standard always takes precedence. Elements of the Handbook may also be useful in developing alternative methodologies to meet the intent of NASA requirements. Alternative methodologies that are not described in this Handbook may be proposed to the delegated NASA Technical Authority.

The scope of this Handbook does not include process control requirements. A base assumption in this Handbook behind the discussion on strength, fatigue, and fracture control is that the part material quality complies with NASA Materials and Processes (M&P) requirements; and all relevant material properties such as strength or fatigue data are available. Typically, this procedure would involve compliance with NASA-STD-6016, Standard Materials and Processes Requirements for Spacecraft; in the case of AM parts, compliance with NASA-STD-6030, Additive Manufacturing Requirements for Spaceflight Systems, is also necessary. Mention of NASA-STD-6030 is contained to some degree for completeness in this Handbook, though details on its content are not included. An AM hardware cognizant engineering organization should utilize both this Handbook and NASA-STD-6030 for complete implementation of NASA structural integrity requirements.

Throughout this Handbook, use of the Responsible Fracture Control Board (RFCB) is invoked numerous times for purposes, including reviews, consultations, and approvals. In the context of this Handbook, the term RFCB includes the NASA delegated Technical Authority in all cases of its use.

1.2 Applicability

1.2.1 This Handbook applies to both human-rated and non-human-rated programs, though notably section 7 on fracture control will typically only be applicable to human-rated programs or when NASA-STD-5019 is required by the Program and/or Contract. Programs that are not working with NASA-STD-5019 or NASA-STD-6030 as a requirement may still choose to follow many of the practices, structures, and fracture control guidelines contained herein on a mission or hardware specific basis to bolster the program and/or to serve as a stepping-stone for a future human rating. The methods and discussions contained herein are intended to be universally applicable across all types of AM parts, with the notable exception of composite overwrapped pressure vessels (COPVs). This Handbook may be applied to parts with any AM process; the following list was envisioned at the time of writing: Laser Powder Bed Fusion (LPBF), Directed Energy Deposition (DED, any energy source), vat photopolymerization, material extrusion, etc.

1.2.2 This Handbook is approved for use by NASA Headquarters and NASA Centers and Facilities. This language applies to the Jet Propulsion Laboratory (a Federally Funded Research and Development Center), other contractors, recipients of grants, cooperative agreements, or other agreements only to the extent specified or referenced in the applicable contracts, grants, or agreements.

1.2.3 References to “this Handbook” refer to NASA-HDBK-5026; references to other documents state the specific document information.

1.2.4 In this Handbook, the terms “may” or “can” denote discretionary privilege or permission, “should” denotes a good practice and is recommended but not required, “will” denotes expected outcome, and “is/are” denotes descriptive material or a statement of fact.

2. REFERENCE DOCUMENTS

2.1 General

Documents listed in this section provide references supporting the guidance in this Handbook. Latest issuances of referenced documents apply unless specific versions are designated. Access reference documents at https://standards.nasa.gov or obtain documents directly from the Standards Developing Body, other document distributors, information provided or linked, or by contacting the office of primary responsibility designee for this Handbook.

2.2 Government Documents

American Institute of Aeronautics and Astronautics

AIAA S-080, Space Systems – Metallic Pressure Vessels, Pressurized Structures, and Pressure Components

American Society for Testing and Materials

ASTM E399, Standard Test Method for Linear-Elastic Plane-Strain Fracture Toughness of Metallic Materials

ASTM E1681, Standard Test Method for Determining Threshold Stress Intensity Factor for Environment-Assisted Cracking of Metallic Materials

ASTM E1820, Standard Test Method for Measurement of Fracture Toughness

ASTM E1823, Standard Terminology Relating to Fatigue and Fracture Testing
ASTM E3166, Standard Guide for Nondestructive Examination of Metal Additively Manufactured Aerospace Parts After Build

ASTM ISO/ASTM 52900, Additive manufacturing – General principles – Fundamentals and vocabulary

Department of Defense

MIL-HDBK-1823A, Nondestructive Evaluation System Reliability Assessment

Federal Aviation Administration (FAA)

FAA Advisory Circular 33.14-1, Damage Tolerance for High Energy Turbine Engine Rotors

NASA

NASA-STD-5001, Structural Design and Test Factors of Safety for Spaceflight Hardware

NASA-STD-5005, Standard for the Design and Fabrication of Ground Support Equipment

NASA-STD-5009, Nondestructive Evaluation Requirements for Fracture-Critical Metallic Components

NASA-STD-5012, Strength and Life Assessment Requirements for Liquid-Fueled Space Propulsion System Engines

NASA-STD-5019, Fracture Control Requirements for Spaceflight Hardware

NASA-STD-6016**,** Standard Materials and Processes Requirements for Spacecraft

NASA-STD-6030, Additive Manufacturing Requirements for Spaceflight Systems

NASA Centers

<u>Johnson Space Center (JSC)</u>

JSC-STD-65828, Structural Design Requirements and Factors of Safety for Spaceflight Hardware for Human Spaceflight

2.3 Non-Government Documents

Acevedo, R., et al., Residual stress analysis of additive manufacturing of metallic parts using ultrasonic waves: State of the art review. Journal of Materials Research and Technology, 2020. 9(4): p. 9457-9477

Altair Inspire 2021: Users Guide. Altair Engineering, Inc., Troy, MI

ANSYS 2021: Users Guide. Canonsburg, Pennsylvania

API-579, Recommended Practice for Fitness-For-Service, American Petroleum Institute, Washington, D. C., 2008.

BEGL, An Assessment Procedure for the High Temperature Response of Structures, British Energy Generation Ltd., R5 Issue 3, 2008.

Brust, F. W., Sallaberry, C. J., and Messner, M. C., “High Temperature Flaw Evaluation Code Case: Technical Basis and Examples,” Proceedings of ASME Pressure Vessels and Piping Conference, paper PVP2022-85957, July 17-22, Las Vegas, NV, 2022.

Christensen, R.M. (2013). The Theory of Materials Failure. United Kingdom: OUP Oxford. Chapters 4, 5, and 11

Dassault Systèmes 3Dexperience 2021: User Guide. Waltham, MA

Enright, M.P., Moody, J.P., Sobotka, J.C. Optimal automated fracture risk assessment of 3D gas turbine engine components. Proceedings of ASME Expo 2016: Turbomachinery Technical Conference and Exposition, June 13-17, 2016, Seoul, South Korea

Gorelik, M. “Additive manufacturing in the context of structural integrity,” International Journal of Fatigue 94 (2017) 168–177

Keller, N., Ploshikhin, V., New Method for fast predictions of residual stress and distortion of AM parts, 25th Annual International Solid Freeform Fabrication Symposium – An Additive Manufacturing Conference, Austin, TX: Austin, August 2014

Levine, L., et al. Outcomes and Conclusions from the 2018 AM-Bench Measurements, Challenge Problems, Modeling Submissions, and Conference, Integrating Materials and Manufacturing Innovation (2019) 9:1-15

2.4 Order of Precedence

2.4.1 The guidance established in this Handbook does not supersede or waive existing guidance found in other Agency documentation.

2.4.2 Conflicts between this Handbook and other documents will be resolved by the delegated NASA Technical Authority.

3. ACRONYMS, ABBREVIATIONS, SYMBOLS, AND DEFINITIONS

3.1 Acronyms, Abbreviations, and Symbols

μm	Micrometer
A/P	Actual-to-Predicted
AIAA	American Institute of Aeronautics and Astronautics
AM	Additive Manufacturing or Additively Manufactured
AMCP	Additive Manufacturing Control Plan
API	American Petroleum Institute
ASTM	American Society for Testing and Materials
BEGL	British Energy Generation LTD
BF	Burst Factor
CAD	Computer-Aided Design
CCS	Critical Crack Size
CIFS	Critical Initial Flaw Size
COPV	Composite Overwrapped Pressure Vessel
CR	Computed Radiography
CT	Computed Tomography
D	Dimensional
da/dN	fatigue crack growth rate
DED	Directed Energy Deposition
DTA	Damage Tolerance Analysis
DTM	Development Test Model
DR	Digital Radiography
EAC	Environmentally Assisted Cracking
EBM	Electron Beam Melting
ECF	Environmental Correction Factor
E-N	strain-life
ET	Eddy Current Test

FAF	Fatigue Analysis Factor
FC	Fracture Critical
FCGR	Fatigue Crack Growth Rate
FEA	Finite Element Analysis
FEM	Finite Element Model
FoS	Factor of Safety
g	Acceleration of Gravity
Gpa	Gigapascals
GSE	Ground Support Equipment
HCF	High-Cycle Fatigue
HDBK	Handbook
HEE	Hydrogen Environmental Embrittlement
HIP	Hot Isostatic Pressing
Hz	Hertz
in	inch
IRT	Infrared thermography testing
ISO	International Organization for Standardization
K_c	Plane stress fracture toughness
kg	kilogram
kHz	kilohertz
K_{Ic}	Plane strain fracture toughness
K_{Ie}	Effective fracture toughness for a surface or elliptically shaped crack
ksi	kilopound per square inch
LBB	Leak-Before-Burst
lbf	pound of force
LCF	Low-Cycle Fatigue
LEFM	Linear-Elastic Fracture Mechanics
LOF	Lack of Fusion
LPBF	Laser Powder Bed Fusion
LVDT	Linear Variable Differential Transducer
M&P	Materials and Processes
max	maximum
MDP	Maximum Design Pressure
MEOP	Maximum Expected Operation Pressure
mm	millimeter
MMPDS	Metallic Materials Properties Development and Standardization
MPS	Material Property Suite
MS	Margin of Safety
MSFC	Marshall Space Flight Center
MT	Magnetic Particle Testing
MUA	Material Usage Agreement
NASA	National Aeronautics and Space Administratoin
ND	Neutron Diffraction
NDE	Non-Destructive Evaluation
NDI	Non-Destructive Inspection

NDT	Non-Destructive Testing
NFC	Non-Fracture Critical
NHLBB	Non-Hazard Leak-Before-Burst
PBF	Powder Bed Fusion
PFM	Probabilistic Fracture Mechanics
PIFC	Polynomial Invariant Failure Criteria
POD	Probability of Detection
PPP	Part Production Plan
PSD	Power Spectral Density
psi	pounds per square inch
PT	Dye Penetrant Test
QMP	Qualified Material Process
RFCB	Responsible Fracture Control Board
RMS	Root Mean Square
RQI	Representative Quality Indicator
RTD	Residual Threat Determination
s	second
SCC	Stress Corrosion Cracking
SLC	Sustained Load Cracking
SI	Système Internationale or metric system of measurement
SLM	Selective Laser Melting
SME	Subject Matter Expert
S-N	stress-life
SSF	Strain Scale Factor
ST	Shearography Testing
STD	Standard
TDF	Test Demonstration Factor
TT	Infrared Thermography Testing
UT	Ultrasonic Test
UTS	Ultimate Tensile Strength
W	Watt
WRS	Weld Residual Stress

3.2 Definitions

90/95 Probability of Detection (90/95 POD): Refers to 90 percent probability of flaw detection with 95 percent lower confidence bound.

A-Basis: A statistically calculated number that at least 99 percent of the population of values is expected to equal or exceed with a confidence of 95 percent. (Source: NASA-STD-5019)

Acceptance Test: A test performed to demonstrate that the hardware is acceptable for its intended use. It also serves as a quality control screen to detect manufacturing, material, or workmanship defects in the flight build and to demonstrate compliance with specified requirements. *Note: Acceptance tests are performed on previously qualified hardware to limit loading conditions*. (Source: NASA-STD-5001)

Additive Manufacturing (AM): Process of joining materials to make parts from three-dimensional model data, usually layer upon layer, as opposed to subtractive manufacturing and formative manufacturing methodologies. Adj., additively manufactured. (Source: NASA-STD-6030)

B-Basis: A statistically calculated value that at least 90 percent of the population is expected to equal or exceed with a confidence of 95 percent.

Catastrophic Event: Loss of life, disabling injury, or loss of a major national asset. (Source: NASA-STD-5019)

Catastrophic Failure: A failure that directly results in a catastrophic event. (Source: NASA-STD-5019)

Catastrophic Hazard: Presence of a risk situation that could directly result in a catastrophic event. (Source: NASA-STD-5019)

Crack or Crack-like Defect: A discontinuity assumed to behave like a crack for fracture control purposes. (Source: NASA-STD-5019)

Critical Ground Support Equipment (GSE): GSE whose loss of function or improper performance could result in serious personnel injury, damage to flight hardware, loss of mission, or major damage to a significant ground asset.

Critical Initial Flaw Size (CIFS): Initial crack-like flaw size with the worst-case orientation that is predicted to precisely meet the defined service lifetimes the life/scatter factor and residual strength requirements (i.e., a margin of safety equal to zero on damage tolerance of the flaw).

Damage Tolerance: Fracture control design concept under which an undetected flaw or damage (consistent in size with the flaw screening method or residual threat determination [RTD]) is assumed to exist and is shown by fracture mechanics analysis or test not to grow to failure (leak or instability) during the period equal to the service life factor times the service life. (Source: NASA-STD-5019)

Design Values: Material properties that are established from test data on a statistical basis and represent the finished part properties. These values are typically based on material allowables and adjusted, using building block tests as necessary, to account for the range of part-specific features and actual conditions. Design values are used in analysis to compute structural design margin (e.g., margin of safety). (See also Material Allowable.)

Detrimental Yielding: Yielding that adversely affects the form, fit, function, or integrity of the structure.

Environmentally Assisted Cracking (EAC): A cracking process in which the environment promotes crack growth or higher crack growth rates than would occur without the presence of the environment (ASTM E1681, Standard Test Method for Determining Threshold Stress Intensity Factor for Environment-Assisted Cracking of Metallic Materials). An example is available in published literature (Source: NASA-STD-5019).

Environmental Correction Factor (ECF): An adjustment factor used to account for differences between the environment (thermal and chemical) in which a part is used and the environment in which it is tested. (Source: NASA-STD-5019)

Failure: Rupture, collapse, excessive deformation, or any other phenomenon resulting in the inability of a structure to sustain specified loads, pressures, and environments or to function as designed (NASA-STD-5001)..

Fatigue Limit: A cyclic stress or strain range below which fatigue initiation failures are unlikely at a defined number of cycles based on fatigue testing. The fatigue limit is commonly defined at a pragmatic cycle count appropriate for the hardware, often 10^7 or 10^8 cycles. For the context of this Handbook, a fatigue limit is defined to be $\geq 10^7$ cycles. At this time, AM materials are not considered to have an endurance limit (a cyclic stress level below which fatigue life is infinite). (Source: NASA-STD-6030)

High-Cycle Fatigue (HCF): A high-frequency, low-amplitude loading condition created by structural, acoustic, or aerodynamic vibrations that can propagate flaws to failure. An example of an HCF loading condition is the vibrational loading of a turbine blade because of structural resonance. (Source: NASA-STD-5019)

K_c: Plane stress fracture toughness. The value of stress intensity factor K at the tangency between a crack extension resistance curve (R-curve) and the configuration-dependent applied K curve (reference ASTM E1823, Standard Terminology Relating to Fatigue and Fracture Testing). This crack extension occurs under conditions that do not approach crack-tip plane strain. The R-curve and K_c vary with the material, specimen size, and thickness. K_c is used in NASGRO® to represent fracture toughness as a function of thickness for use in crack growth calculations. (Source: NASA-STD-5019)

K_{Ic}: Plane strain fracture toughness. The crack extension resistance under conditions of crack-tip plane strain in Mode I for slow rates of loading under predominantly linear-elastic conditions and negligible plastic-zone adjustment that is measured by satisfying a standardized procedure with validity requirements (reference ASTM E399, Standard Test Method for Linear-Elastic Plane-Strain Fracture Toughness K_{Ic} of Metallic Materials). Another quantity, K_{Jic}, defined for conditions with limited plasticity from Jic may also be useful (reference ASTM E1820, Standard Test Method for Measurement of Fracture Toughness). (Source: NASA-STD-5019)

Lack of Fusion (LOF)*: A type of process-induced porosity, in which the powder or wire feedstock is not fully melted or fused onto the previously deposited substrate. This type of flaw can be an empty cavity, or contain unmelted or partially fused powder, referred to as unconsolidated powder. LOF typically occurs in the bulk, making its detection difficult. Like voids, LOF can occur across single (horizontal LOF) or multiple layers (vertical LOF).

Limit Load: The maximum anticipated load, or combination of loads, that a structure may experience during its design service life under all expected conditions of operation. (Source: NASA-STD-5019)

Limit Stress: The limit load expressed as load per unit area.

Material Allowable: Material values that are determined from test data of the bulk material on a statistical basis. Allowable development approaches are established via industry standards (e.g., Metallic Materials Properties Development and Standardization (MMPDS) or company-specific methodology) and are based on testing conducted using accepted industry or company standards. Material allowables form the basis of design values. (See also Design Value.) (Source: NASA-STD-6030)

Maximum Design Pressure (MDP): The highest possible operating pressure considering maximum temperature, maximum relief pressure, maximum regulator pressure, and, where applicable, transient pressure excursions. MDP for human-rated hardware is a two-failure tolerant pressure, i.e., MDP will not be exceeded for any combination of two credible failures that will affect pressure. For all other hardware, MDP is equivalent to the maximum expected operating pressure (MEOP). (Source: NASA-STD-5001)

Maximum Expected Operating Pressure (MEOP): The maximum pressure which pressurized hardware is expected to experience during its service life, in association with its applicable operating environments. MEOP includes the effects of temperature, transient peaks, vehicle acceleration, and relief valve tolerance. (Source: NASA-STD-5001)

Non-Destructive Evaluation (NDE): Examination of parts for flaws using established and standardized inspection techniques that are harmless to hardware, such as radiography, penetrant, ultrasonic, magnetic particle, and eddy current. NDE is sometimes referred to as non-destructive testing (NDT) or non-destructive inspection (NDI). (Source: NASA-STD-5019)

Non-Hazardous-Leak-Before-Burst (NHLBB): A non-fracture critical classification for metallic pressurized hardware that contains a material that is not hazardous and that exhibits the leak-before-burst (LBB) failure mode in a non-hazardous manner. (Source: NASA-STD-5019)

Non-Structure: Any hardware element that is not intended to resist loads and does not contain any electronic or electromechanical components. Examples of non-structure items are cable harnesses, plumbing lines, radio-frequency waveguides, heaters, louvers, thermal blankets, purge equipment, small sunshades/shields, etc., or elements within black boxes.

Poor Dimensional Accuracy*: Physical measurements of geometrical features that do not meet engineering drawing, leading to an out-of-tolerance part. This type of flaw is caused by stair stepping, relief of residual stresses and associated warping, rapid contraction during cooling after fusion, or sagging under gravity of unsupported areas with vertical overhang or downward facing surfaces during build.

Porosity (Gas)*: Voids that are spherical or faceted in shape; with sufficient sources of gaseous species, may be intermittent within the deposit or elongated, interconnected, or chained due to the moving solidification front. Gas porosity is caused by absorption/desorption of gaseous species (nitrogen, oxygen) during solidification, or volatile contaminants (moisture or hydrocarbons) in the feedstock. Gas porosity on the surface can interfere with or preclude certain NDT methods, while porosity inside the part can reduce strength in its vicinity. Like voids, gas porosity causes a part to be less than fully dense.

Porosity (Keyhole)*: A type of porosity characterized by the appearance of spherical void formed due to instability of the vapor cavity during processing. Keyhole porosity is created when the energy density is sufficiently high due to high laser power and low scanning speed to cause a deep melt pool resulting in hydrodynamic instability of the surrounding liquid metal and subsequent collapse, leaving a void at the root of the keyhole. Like generic voids and gas porosity, keyhole porosity causes a part to be less than fully dense.

Primary Structure: The structure that is the principal load path for all subsystems, components, and other structures.

Proof Test: A test on the flight article that is performed to verify structural acceptability or to screen flaws. The proof test load and/or pressure level is the proof test factor times limit load and/or MDP. Proof tests may be conducted in the operational environment, or the test levels may be adjusted via an ECF. *Note that some sections within this NASA Technical Standard may specify when an ECF is optional versus when it is prescribed for the classification if the test is not conducted in the operational environment.*(Source: NASA-STD-5019)

Proof Test Factor: A multiplying factor applied to limit load or MDP required to qualify the flight hardware for workmanship, quality, and structural integrity. When proof tests are performed to establish structural acceptability, the proof test factor is specified. When screening for flaws with a proof test, the proof test factor is derived by fracture mechanics principles. (Source: NASA-STD-5019)

Protoflight Hardware: Hardware that is qualified using a protoflight verification approach (see Protoflight Test). (Source: NASA-STD-5001)

Protoflight Test: A test performed on flight or flight-like hardware (i.e., is built with same drawings, materials, and processes as the flight unit) to demonstrate that the design meets structural integrity requirements. The test is performed at loads or pressure in excess of limit load or MDP but below the yield strength of the structure. When performed on flight structure, the test also verifies the workmanship and material quality of the flight build. *Note: Protoflight tests combine elements of prototype and acceptance test programs.* (Source: NASA-STD-5001)

Prototype Hardware: Hardware of a new design that is produced from the same drawings and using the same materials, tooling, manufacturing processes, inspection methods, and personnel competency levels as will be used for the flight hardware. *Note: Prototype hardware is dedicated test hardware that is not intended to be used as a flight unit.* (Source: NASA-STD-5001)

Prototype Test: A test conducted using prototype hardware to demonstrate that all structural integrity requirements have been met. *Note: Prototype testing is performed at load levels sufficient to demonstrate that the test article will not fail at ultimate design loads.* (Source: NASA-STD-5001)

Qualification Test: A test performed to qualify the hardware design for flight. *Note: Qualification tests are conducted on a flight-quality structure at load levels sufficient to demonstrate that all structural design requirements have been met. Both protoflight and prototype tests are considered qualification tests.* (Source: NASA-STD-5001)

Qualification Test Factor: A multiplying factor to be applied to the limit load or MDP to define the qualification test load or pressure. (Source: NASA-STD-5001)

Residual Stress: The stress that remains in a material or component after processing, post-processing, fabrication, assembly, testing, or operation.

Responsible Fracture Control Board (RFCB): A designated multi-discipline group of experts that has the authority to develop, interpret, and approve fracture control requirements and the responsibility for overseeing and approving the technical adequacy of all fracture control activities.

Scatter Factor: A safety factor applied to life for fatigue and fracture mechanics analyses.

Secondary Structure: Any structural element whose failure would not result in the general failure of the structural support of the spacecraft or of a major spacecraft assembly. Normally, the vibro-acoustic environment predominates the loading of most of these structural elements that support relatively lighter items. The loads generation methodology employed for these environments and items is generally more conservative than that employed to generate loads for the transient regime and primary structure.

Service Life: Time interval for a part beginning with manufacture and extending throughout all phases of its specified mission usage. The period of time or number of cycles that includes all relevant loadings, conditions, and environments encountered during this period that will affect flaw growth, including all manufacturing, testing, storage, transportation, launch, on-orbit, descent, landing, and if applicable, post-landing events, refurbishments, retesting, and repeated flights until the hardware is retired from service.

Shatterable Material: Any material that is prone to brittle failures during operation that could release many small pieces into the surrounding environment.

Structure: All hardware elements that carry loads, sustain pressures, and/or provide physical support and/or containment.

Sustained Load Cracking (SLC): Growth of a pre-existing crack in susceptible metallic alloys7 under sustained stress without assistance from an external environment. A threshold stress intensity factor can be obtained by procedures such as those in ASTM E1681 for the case of an inert or vacuum environment. One publication determines the effects of hydrogen content and temperature on SLC in Ti-6Al-4V (Source: NASA-STD-5019).

Ultimate Design Load: The product of the ultimate FoS and the limit load.

Ultimate Factor of Safety (Ultimate Safety Factor): A specified factor to be applied to limit load. No ultimate structural failure is allowed for a load equal to the ultimate FoS multiplied times limit load. (Source: NASA-STD-5019)

Voids*: A general term encompassing both irregularly shaped or elongated cavities (process-induced porosity, LOF, skipped layers, large cracks, or delamination) and spherically shaped cavities (gas-induced and keyhole porosity). These cavities can be empty or filled with partially or wholly unfused powder. Voids are distinct from intentionally added open cells that reduce weight. Voids cause a part to be less than fully dense.

Yield Design Load: The product of the yield FoS and the limit load. (Source: NASA-STD-5001)

Yield Factor of Safety: A multiplying factor applied to the limit load to obtain the maximum load a structure is required to sustain without exceeding the yield stress. This does not apply to non-detrimental local yielding at stress concentrations or structures to be intentionally yielded during function such as energy absorbers.

**Definition was developed at NASA White Sands Test Facility and has been used both in this Handbook and in ASTM ISO/ASTM 52900, Additive manufacturing - General principles – Fundamentals and vocabulary.*

4. OVERVIEW OF ADDITIVE MANUFACTURING (AM) PART REQUIREMENTS

The focus of this Handbook is to provide guidance for additively manufactured (AM) part assessments in the areas of strength, fatigue, non-destructive evaluation (NDE), and fracture control. The flow chart shown in Figure 1, Requirement Compliance Flowchart for Strength, Fatigue, and Fracture Control, illustrates the compliance path through these requirements, including reference to the corresponding sections in this Handbook. Upon reaching one of the “Complete” points within the fracture control subsection of the flow chart, one would meet the intent of NASA technical standards containing strength, fatigue, and fracture control requirements (i.e., NASA-STD-5001 and NASA-STD-5019). If a viable path to one of the complete points in Figure 1 is not possible, and the “Non-compliant” point is reached in the flow chart, AM may not be an acceptable choice for NASA applications where fracture control is required. At the non-compliant point, if AM is still strongly desired, consultation with the Responsible Fracture Control Board (RFCB) is recommended and options to consider include enhanced NDE, part redesign, enhanced acceptance testing, and material change. Note: Independent of formal approval from the RFCB at the plan or part specific level, documentation of the applicable path in Figure 1 below is required in the Integrated Structural Integrity Rationale as part of the Part Production Plan (PPP) required by NASA-STD-6030 section 7.

As indicated in Figure 1, compliance with NASA process control requirements contained in NASA-STD-6030 is considered a prerequisite in the context of this Handbook, meaning AM parts subject to this Handbook are produced under the governance of an approved Additive Manufacturing Control Plan (AMCP) that implements the requirements of NASA-STD-6030 directly or through approved tailoring.

The flow chart is used to illustrate that all parts satisfy strength, fatigue, and fracture control requirements and offers additional details on options for compliance based on fracture classification, NDE capabilities, and proof test details. Use of probabilistic fracture mechanics (PFM) is a permitted approach for fracture control but necessitates specialized input data as well as collaboration with and ultimate approval from the RFCB. Guidance for each of the steps and on each of the paths to one of the five complete points illustrated in Figure 1 is provided in this Handbook.

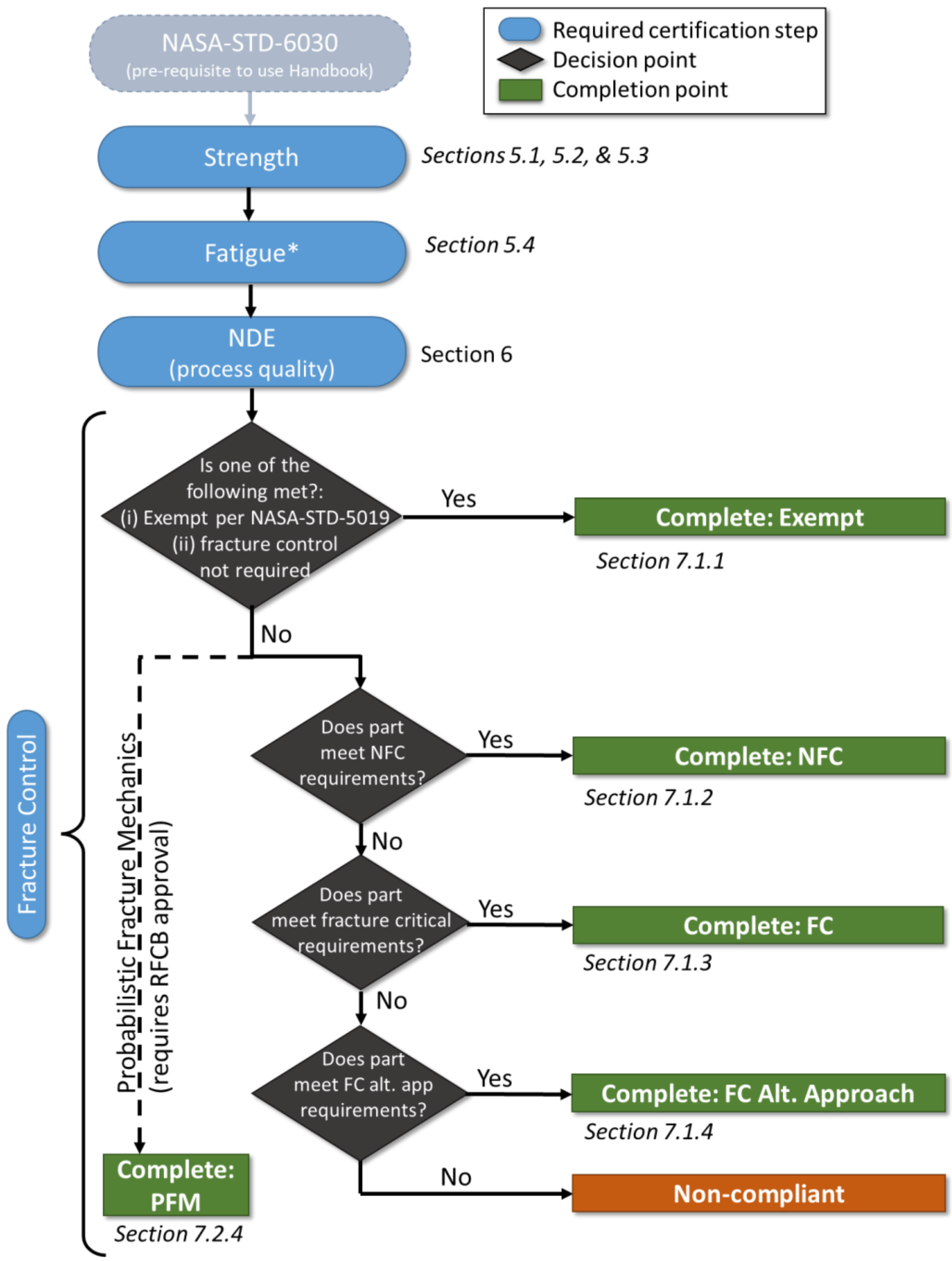

Figure 1—Requirement Compliance Flowchart for Strength, Fatigue, and Fracture Control (Corresponding Handbook sections shown below the "Complete" boxes). *Note fatigue requirements may be superseded in some cases by a damage tolerance analysis (DTA) for fracture control.

5. STRUCTURAL INTEGRITY GUIDELINES

The intent of the following guidelines is to ensure an AM part maintains its structural integrity in accordance with its risk classification and its requirements through its entire service life. The evaluation and qualification of AM part structural integrity consists of a combination of strength analysis, fatigue/fracture analysis, NDE, proof testing, and implementation of material and process controls as outlined in NASA-STD-6030. Note that the structural assessment methodology utilized according to this Handbook is required to be reported in the Integrated Structural Integrity Rationale per NASA-STD-6030.

Additively manufactured parts receive an “AM part classification” per NASA-STD-6030. It is a two-tier system of classifications: Primary Classification, based on the consequence of part failure, and a Secondary Classification, based on the structural demand and AM risk associated with the part. In this Handbook, only the Primary Classification tier, Classes A, B, or C, will be utilized[1]. Class A corresponds to parts with a high consequence of failure, Class C corresponds to parts with a negligible consequence of failure, and Class B corresponds to all parts that are neither A nor C. The AM part classifications should not be confused with the fracture classifications defined in NASA-STD-5019, as both the criteria and usage differ. Figure 2, AM Part Classification per NASA-STD-6030, shows the AM part classification as described in NASA-STD-6030.

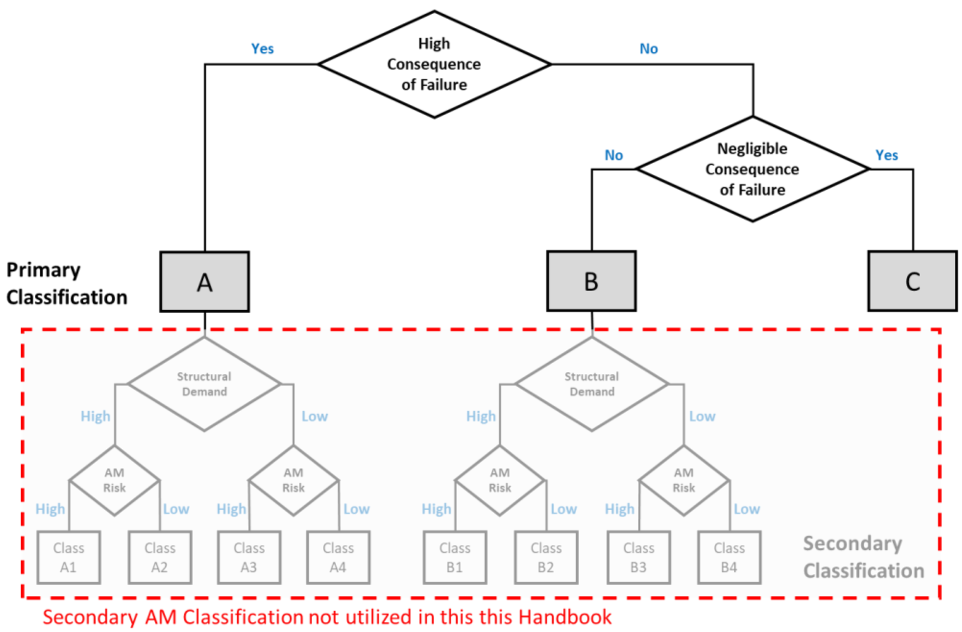

Figure 2—AM Part Classification per NASA-STD-6030

An initial step in implementation of structural integrity requirements is to determine the primary AM part classification. Primary structures, fracture critical (FC) structures, and critical ground support equipment (GSE) in spacecraft and major sub-systems should be designated as Class A, as failure of those items would result in a high consequence of failure. Structures that are secondary, non-fracture critical (NFC), exempt[2] (from fracture control standpoint), and non-critical ground support equipment may be designated as Class B or Class C depending on the nature of the design and consequences of failure (see section 7.1.2). Note that failure of Class C

[1] Note that throughout this document Class A, B and C are referring to AM part classifications according to NASA-STD-6030.
[2] Note that the definition of exempt in NASA-STD-6030 and NASA-STD-5019 are different and it should not be confused. Here “exempt” refers to the definition from fracture control perspective (NASA-STD-5019).

structures must be shown to have a near zero (negligible) consequence to the safety of flight and non-flight hardware. If the safety risk cannot be clearly determined in discussions with structures and fracture control, NASA Safety must be engaged to help determine the consequence of failure. Additional discussion on AM part classifications can be found in NASA-STD-6030, section 4.3.1.

5.1 Strength

Positive margins of safety (MS) are demonstrated for all AM flight and ground support structures. Structural analysis and MS calculations should be performed according to sections 5.1, 5.2, and 5.3.

5.1.1 Material Allowables and Design Values for AM Parts

NASA-STD-6030 provides specific requirements for the development and use of material properties for AM product forms. These properties, and the data that support them, are maintained in a Material Properties Suite (MPS), approved by the responsible NASA Materials and Processes engineer in accordance with NASA-STD-6016. NASA-STD-6030 adopts evolving standardized terminology for AM properties that defines “material allowables” as those properties derived from the bulk AM material with minimal influence of outside factors, such as thermal history variation during the build, AM surface finish, or test temperature effects. From these base material allowables, “design values” are developed that include the effects of any influence factors on material behavior present in the part’s application. The design values are used in structural assessment. In some benign cases where there are no influence factors present, the material allowable and design value are equal.

NASA-STD-6030 established procedures for the development of material allowables and design values, including specimen quantities and the assessment of sources of variability. All material allowables and design values produced per NASA-STD-6030 for application to parts that are Class A, Class B, FC, or primary structure are intended to be derived to bound 99% of the population with 95% confidence, which reflects the statistical reliability commonly associated with A-basis properties used in other product forms. For Class C parts, design values of typical basis (average values) are permitted. Because aerospace industry standards are still under development and review, there has yet to be a basis title applied to AM materials allowables. When established, it is not expected to use the same terminology as traditional materials. Even when materials allowables are published by the industry, it is not likely that NASA will accept these properties since the rigor of the process specifications and materials specifications are not yet high enough for aerospace applications.

Materials Allowables and other material, thermal, and physical properties derived using the methods in NASA-STD-6030 must be reviewed and approved through the Material Usage

Agreement (MUA) process per NASA-STD-6016. These cases might be considered somewhat analogous to the use of S-basis properties in traditional product forms; however, each case requires review and approval for specific applications.

NASA-STD-6016 allows for the use of B-basis properties (90% bound at 95% confidence) for certain types of redundant structure. If there is sufficient advantage to the use of values of reduced basis for Class B AM parts, an exception to NASA-STD-6030 may be coordinated through the AM PPP and/or the MUA system in accordance with NASA-STD-6016.

5.1.2 Design Factors of Safety (FoS) and Test Factors

Recommended Design Factors of Safety (FoS) for yield and ultimate strength, and test factors for use on AM parts are listed in Table 1, Recommended Minimum Design Factors of Safety and Test Factors for Additively Manufactured Parts. These factors are based on those found in NASA-STD-5001, NASA-STD-5009, NASA-STD-6030 and ANSI/AIAA S-080. The actual requirement for Factors of Safety is determined by the responsible NASA structures system manager for the specific hardware program. The values in Table 1 are the minimum to use with the flight limit load (or flight limit stress). The factors listed in Table 1 are applicable to all fracture classifications or otherwise to all parts in programs where fracture control is not required. The proof test factors may be higher in practice if needed for fracture control per section 7.2.3 in this Handbook. Proposals to flight projects for alternate factors should address specifically each item in the following list as justification:

- Conservatism of methods used for load analysis (e.g., use of a fitting factor).
- Category of structure (e.g., primary, secondary, fracture classification, AM class).
- Criticality of item relative to possibility of a catastrophic failure.
- Analytically tractable structure and amenable to analysis (i.e., analysis methods may be limited or unavailable for some types of structures, for example mechanisms or propulsion components).
- Fidelity of analysis (e.g., use of a model uncertainty factor).
- Completeness of the test verification program (subsystem-level testing, system-level testing, model correlation activities).

Table 1—Recommended Minimum Design Factors of Safety and Test Factors for Additively Manufactured Parts

	Design Factor			Recommended Additional Uncertainty Factor for Class A Parts (AM Factor)		Qualification Test Factor[IV]	Proof Test Factor[IV, V]
	Yield	Ultimate		Yield	Ultimate		
Metallic-Structure - protoflight	1.25	1.40		1.20	1.30	1.20	1.20 (for Class A)
Metallic-Structure - prototype[III]	1.00	1.40		1.20	1.30	1.40	1.05 (for Class A)
Metallic Pressure Vessel[II]	1.25	2.00 (BF)	1.40	1.20	1.30	1.20	See section 7.2.3.1 (Equation 5)
Metallic Lines, Fittings diameter < 38 mm (1.5 in)[II]	1.5	4.00 (BF)	1.40	1.20	1.30	1.20	1.50
Metallic Lines, Fittings diameter > 38 mm (1.5 in), and Other Pressurized Components[II]	1.5	2.50 (BF)	1.40	1.20	1.30	1.20	1.50
Metallic Structure - analysis-only	1.60	2.00		Not permitted for Class A[VI]	Not permitted for Class A[VI]	N/A	N/A
Non-metallic-structure - protoflight	N/A	2.00		Not permitted for Class A[V]	Not permitted for Class A[V]	1.20	1.20[I]
Non-metallic-structure - prototype	N/A	1.87[VI]		Not permitted for Class A[V]	Not permitted for Class A[V]	1.40	1.05[I]
Ground Support Structure qualified by testing	2.00	3.00		1.20	1.30	1.50	1.50[I]

[I]See section 7.2.3 for discussion on proof acceptance testing.

[II]ANSI/AIAA S-080 offers guidance to increase the probability of a successful pressure system. Determination of Burst Factors (BF), Proof Factors, Negative Pressure Factors, and Design Safety Factors that are defined in ANSI/AIAA S-080 may be proposed to the delegated NASA Technical Authority for use in lieu of those in Table 1 for metallic pressurized parts. For combined pressure and limit load loading, the BF may be reduced to 1.0.

[III]See NASA-STD-5001 for details on prototype verification approach.

[IV]The needed proof test factor may be higher than listed in Table 1 if required for fracture control flaw screening per section 7.2.3.

[V]Analysis-only metallic, and protoflight polymer AM structures are not permitted in Class A applications.

[VI] Derived based on the ratio between 1.4 and 1.5 for prototype and protoflight composite structures, i.e. 2.00x1.4/1.5=1.87

Rationale: The AM Factors (i.e., additional uncertainty factors) for Class A parts listed in Table 1 correspond to the "Low Structural Demand" criteria in NASA-STD-6030. If the AM factors are used, the design will meet with yield stress and ultimate stress margin criteria required for classification as "Low Structural Demand" (Class A sub-class 3 & 4) per NASA-STD-6030. The use of the additional factors is recommended by this Handbook to account for risk and uncertainty in AM parts related to as-built properties, material property uniformity, residual stresses, and a relatively limited foundation of experience and expertise in AM technology compared to other legacy material forms and conventional manufacturing processes. The AM Factors are subject to change as manufacturing processes, and material testing and characterization techniques improve and reduce uncertainty. This needs to be assessed by the delegated NASA Technical Authority.

All Class A AM flight parts should be proof tested using the proof test factor defined in Table 1 according to Equation 1:

$$\boldsymbol{Proof\ Test\ Load = Limit\ Load\ \times Proof\ Test\ Factor\ \times ECF} \qquad \text{(Equation 1)}$$

where ECF is the Environmental Correction Factor. The ECF is necessary because many proof tests will occur in a lab setting at room temperature, where the actual service environment is more severe. The ECF is calculated as the strength ratio between test and service environments (maximum yield or ultimate) or the Young's modulus ratio (for buckling considerations).
All proof-tested parts receive post-proof NDE (both surface and volumetric) regardless of the details and specific role of the test in the qualification methodology. Guidance on proof testing in both sections 7.2.3 and 7.2.3.1 is fully applicable to the proof testing called for in Table 1.

As specified in NASA-STD-5001, a prototype verification approach requires a separate, dedicated test structure, identical to the flight structure, be tested to ultimate loads to demonstrate that the design meets both yield and ultimate factor of safety (FoS) requirements. A protoflight verification approach tests the flight structure to levels above the limit load but below the yield strength to verify workmanship and demonstrate structural integrity of the flight hardware.

Both protoflight and prototype verification approach may be considered for Class A or FC AM parts, though caution should be taken. For propulsion structure or Solid Rocket Motor (SRM), where fatigue life and functional performance are critical to the flight hardware, a dedicated test article is strongly recommended to fully verify its design prior to integrating with the propulsion system. For a structure where its load path can be easily determined, the flight loads are well characterized, and its function is primarily strength driven, **the** protoflight testing approach may be permitted, subject to the approval of the responsible project. Analysis-only approaches should not be used for Class A parts.

For FC AM parts, the verification approach needs to be reviewed by the RFCB. For a protoflight test hardware assembly comprised of both prototype hardware and flight hardware, use the protoflight FoS, including the AM factors. The intent is to ensure that the protoflight assembly test not fail due to yielding of the prototype hardware as part of the assembly.

For an assembly comprised of both development test model (DTM) hardware and flight test hardware, use the protoflight FoS as defined in Table 1.

5.1.3 Margins of Safety (MS)

The yield and ultimate MS should be positive. Class A parts should utilize the AM **f**actors, as defined in Table 1, together with the FoS yield and FoS ultimate factors to calculate the respective margin of safety. For minimum-weight design, the MS should be as small as practicable, with zero as acceptable.

The ultimate margin of safety (MS_{ult}) is calculated as in Equation 2:

$$\boldsymbol{MS_{ult}} = \frac{\textit{Allowable Ultimate Stress}}{\textit{Limit Stress} \times \boldsymbol{FoS_{ult}} \times \boldsymbol{AM\ Factor_{ult}}} - \mathbf{1} \qquad \text{(Equation 2)}$$

where FoS_{ult} is the ultimate factor of safety.

The yield margin of safety (MS_{yld}) is calculated as in Equation 3:

$$\boldsymbol{MS_{yld}} = \frac{\textit{Allowable Yield Stress}}{\textit{Limit Stress} \times \boldsymbol{FoS_{yld}} \times \boldsymbol{AM\ Factor_{yld}}} - \mathbf{1} \qquad \text{(Equation 3)}$$

where FoS_{yld} is the yield factor of safety.

5.2 Verification and Validation

5.2.1 General

Structural verification and validation are accomplished by a combination of analyses and tests. Depending on the design complexity and criticality, some structures are verified by test only, while some secondary structures are verified by analysis only with test augmentation. Primary and/or critical structures are verified by test and augmented with analyses. This section describes the full range of verification and validation for both structural integrity and functional integrity.

5.2.2 Structural Analyses

One must perform detailed structural analyses on all primary and secondary flight structures; as well as test-correlate finite element models (FEMs) when test data is available (i.e., when test data becomes available from modal, qualification, or proof test). Detailed analyses should be carried out at the piece-part level (including mechanical interfaces such as bolted and bonded joints, to facilitate MS assessments (using applicable FoS and AM factors of Table 1) and drawing sign-off by the structural analysts. The von-Mises failure criterion is recommended for use in stress analysis of ductile metallic AM parts that are approximately isotropic (less than 5% difference in properties by orientation). A maximum principal stress criterion is recommended for materials that are orthotropic or brittle. As defined in NASA-STD-6030, materials with less than 3% tensile elongation (plastic strain), should be assessed using a "brittle material" design criterion. Other failure criteria such as polynomial invariants failure criterion (PIFC) or Tsai–Wu failure criterion can be considered for orthotropic AM parts with complex stress states. Specific failure criteria and rationale should be approved by the delegated NASA Technical Authority.

Some AM parts tend to be multi-functional, such as a pressurant tank that is not only internally pressurized but also supports other non-structural components. This type of structure should have sufficient strength (i.e., show positive MS) to simultaneously withstand all loading conditions, including internal loads throughout its expected service life and external flight limit loads induced by both launch and operational environments.

Structural analyses using classical hand analyses and/or finite element analysis (FEA) methods should be performed following acceptable aerospace industry methodologies and modeling guidelines. For AM parts that have complex geometry or that are structurally optimized, a detailed FEM should be used to assess stress, strain, displacement, stiffness, and stability.

In creating the FEM, modeling assumptions and idealizations should be realistic, accurately reflect the nominal stiffness, and be properly documented. For cases where model parameter approximations must be made, they should be conservative with respect to the particular analytic purpose of the model. For complex geometry, which tends to be the case for topology-optimized AM parts, the finite element mesh should be detailed enough to capture the stress concentrations and instability behaviors in all locations of interest. Mesh convergence checks should be performed to ensure mesh density and quality to be adequate. This is particularly important since high stresses relative to its material allowable tend to be more evenly distributed for a structurally optimized part. The assembled FEM should be thoroughly assessed against a set of established FEM checkout procedures for the finite element code being used.

Approaches and results of structural analyses should be adequately documented for reviews and/or audits. For AM parts, at a minimum, documentation should include derivation of loads, modeling assumptions, boundary conditions, physical dimension of parts, material properties (including influence factors such as thin section effects), surface roughness, residual stress if present, heat treatment, build directions, multiaxial stress states, and MS, including the accompanying failure mode.

5.2.3 Residual Stress

Residual stress in AM parts resulting from improper manufacturing processing and post-processing, thus leading to poor dimensional accuracy (warping, cracking, etc.), should be accounted for in the analysis. A best effort should be made to quantify residual stress for use in the analysis or, alternatively, evidence should be provided showing that residual stresses are enveloped by the analysis method via factors, conservative assumptions, or other mitigations. A test-validated process simulation model of printing could be used as another data point for residual stresses. Quantified residual stresses used in an analysis need not be accurate if they can be demonstrated to be conservative. The stress analyst must provide the delegated NASA Technical Authority with (a) a description of how residual stresses were either quantified or accounted for otherwise in the analysis, including methodology, and (b) a description of manufacturing processing and post-processing steps taken to minimize or alleviate residual stresses.

Schajer, G.S. [2013] offers guidance on residual stress measurement techniques, including hole drilling (shallow and deep), slitting, contour, X-ray, neutron diffraction (ND), synchrotron diffraction, and more.

5.3 Structural Integrity Qualification

Qualification refers to activities that demonstrate a hardware design meets requirements. Typically, once the governing requirements are verified, structural integrity qualification is achieved through a combination of analysis and testing. Although analysis with higher FoS is presented in Table 1 for use in lieu of testing to qualify the strength of a metallic structure’s design, the design of Class A and FC structures should be qualified by both analysis and testing. Note that NASA-STD-6030 requires reporting these details in the Integrated Structural Integrity Rationale. Much of the work described in this Handbook may be summarized to help satisfy this requirement in NASA-STD-6030.

Structural tests may be devised and performed to verify design concepts, to validate analysis results, and/or to proof test flight hardware, ground handling equipment, and test fixtures. Tests may be conducted at the system, subsystem, assembly, component, and/or coupon levels as appropriate.

Structural tests on flight hardware should be supported by analysis. Test plans should be written clearly identifying test objective; success criteria; hardware pedigree; load cases, including loading spectrum if applicable; other factors that may alter the stiffness or strength of the material or component under test (for example, the effect of thermal and chemical environments evaluated during test); and key personnel. At the conclusion of testing, the results and observations should be summarized in a test report.

For metallic pressurized parts subjected to rupture testing, non-hazardous-leak-before-burst (NHLBB) testing, or proof testing a test-analysis correlation should be performed. For example, burst prediction results, including percent error on strain distribution compared to test, should be reported to the delegated NASA Technical Authority. A prediction error of less than 10% is desired; an error greater than 10% but less than 20% may be acceptable also after review from the delegated NASA Technical Authority.

Qualification of both primary and FC AM structure and AM ground support equipment, such as hoist structures, should be accomplished through testing to a proof test factor times design limit load per Table 1. In a multi-phase mission, peak limit load for specific structural elements may occur during different phases. Qualification of AM structural elements is achieved when analysis demonstrates a positive MS (with applicable FoS and AM factors), and the test load level of proof test factor times limit load is met without any detected failures. Note that the direction of the applied test load should be considered if the critical structural margin is buckling. For a critical AM structure subject to complex loading conditions, multiple proof load cases may be required.

The minimum proof test factor for all non-metallic AM structures is 1.2 per Table 1.

5.4 Fatigue

All AM parts, both metallic and non-metallic, should receive a fatigue assessment and are shown to be capable of surviving a minimum of four (4) service lives in cases of low-cycle fatigue (LCF) or surviving a minimum of 10 service lives in cases of high-cycle fatigue (HCF) (see NASA-STD-5012 and JSC-65828 for reference to scatter factor 10 for high-cycle fatigue). Note that both LCF and HCF are defined in section 3.2 and in NASA-STD-5019. Fatigue Analysis Factors (FAF) should be applied in all fatigue analyses to the magnitude of cyclic loading. See Table 2, Fatigue Analysis Factor. If a damage tolerance assessment is performed per section 7.2.1, this may substitute the need for a fatigue analysis as it is more conservative.

Table 2—Fatigue Analysis Factor

	Fatigue Analysis Factor
Class A	1.5
Class B and C	1.15

Fatigue performance is critical to consider in the design of metallic AM parts as it often can be the limiting factor in a design. This observation, along with a higher propensity for defects in AM parts compared to other manufacturing methods, is the rationale behind the FAF applied to cyclic loading or alternating stress. In addition to the manufacturing process parameters, there is a wide variety of factors that significantly influence the fatigue performance of AM parts, such as effects of build direction, surface finish, residual stresses, heat treatment and multiaxial stress states. Although not based on experimental evaluations of AM materials, the selected FAF are derived from heritage factors and engineering judgment based on previous experience with fatigue-sensitive materials. Similar to the NFC Low-Risk parts fracture control classification criteria in NASA-STD-5019, the FAF for Class A parts is defined to be 1.5. Use of this factor would also effectively drive Class A AM parts into a condition unlikely for crack growth just as if a conventionally manufactured metallic part were classified as NFC low risk. An alternative approach for certain FC AM parts is proposed in section 7.1.4 of this Handbook. Special attention should be given to (a) parts subjected to a large number of cycles at high stress levels, and (b) the effect of the selected AM build process as it influences fatigue characteristics. The

FAFs are subject to change as AM techniques and material testing and characterization improve and decrease uncertainty. This should be assessed by the delegated NASA Technical Authority.

5.4.1 Additive Manufacturing (AM) Fatigue Considerations

Fatigue life is highly dependent on the manufacturing processes associated with a metallic AM part. While metallic AM processes can result in apparent macroscopic uniformity, the associated microstructure and finished part properties may vary greatly depending on the process methods used and subsequent post-processing and finishing techniques. Two commonly used processes for metallic AM parts are Powder Bed Fusion (PBF) and Blown Powder Directed Energy Deposition (DED). Both processes involve metal powder (metal wire is also used in DED) being melted by an energy source such as a laser or electron beam. Microstructure variations and porosity generations are strongly influenced by the process parameters such as energy input, scanning speed, hatch spacing, atmospheric conditions, heat conduction, and part thickness.

Fatigue life can be improved with various manufacturing and post-processing techniques, such as having closely monitored process control, improving surface finish via machining, imparting compressive residual stress such as peening, or implementing heat treatment such as Hot Isostatic Pressing (HIP) to reduce residual stresses and close internal defects. The variability in AM processes should be considered when assessing fatigue life for an AM part. If a fatigue life scatter factor (life reduction factor) of 4, including the FAF specified in Table 2, is known or observed to be insufficient (e.g., fatigue failure occurs), a higher factor should be used.

5.4.2 Additive Manufacturing (AM) Fatigue Data

Fatigue life assessment relies heavily on test data, such as stress-life (S-N) curves. All data used for fatigue analysis should be from NASA-approved sources and comply with the NASA-STD-6030 requirement. Data should ideally be obtained from samples made from the same AM machine using the same powder/wire feedstock lots, Qualified Material Processes (QMP), and post-processing used in the flight part builds. Multiple machines may be used for fatigue data collection and flight part builds if data exists to suggest that the machines are in-family with one another (i.e., same model, manufacturer, feedstock, software, etc.) and data exists demonstrating that the machines produce parts with the same material properties in accordance with the QMP registration process in NASA-STD-6030.

If HCF and LCF fatigue data for the material and process are not readily available, a test campaign should be performed according to NASA-STD-6030, section 5.5.3.5, Tables 13 and 14. Generation of material allowables fatigue data follows NASA-STD-6030, section 6.11.4.4. Representative test coupons should be designed to accurately capture the loading of the AM part during its service life at all locations of interest. Regions of a part that have locally varying surface conditions are enveloped by the fatigue test coupon surface conditions. Surface finish characteristics may include roughness, build orientation patterns, build orientation relative to the horizontal (i.e., vertical, flat, overhanging), and embedded features. A multidisciplinary assessment of fatigue coupon design is recommended. See section 7.3 of this Handbook for additional guidance on fracture and fatigue crack growth testing.

Witness coupon fatigue testing for independent builds should occur for each Class A or FC flight build as specified in NASA-STD-6030, section 4.11.1. S-N curves should be representative or enveloping of the flight build orientation and anisotropic stress state, including the effects of residual stress. Alternatively, residual stress can be considered as elevated mean stress in the fatigue analysis.

With all approaches to assess fatigue performance, the delegated NASA Technical Authority should be consulted.

5.4.3 Additive Manufacturing (AM) Fatigue Analysis Approach

Fatigue analysis should be performed for metallic AM parts with a minimum fatigue life scatter factor of four. Note that a minimum service life factor of 10 is required (as specified in NASA-STD-5012) for liquid-fueled space propulsion system engines under HCF. Per Table 2, a FAF should be applied to the cyclic limit load in the fatigue analysis load spectrum. For fatigue assessments generally associated with random vibration excitations, Miner's rule cumulative fatigue damage approach can be used to assess the fatigue life margin for both S-N fatigue and strain-life (E-N) fatigue. While S-N fatigue is only applicable to elastic stress and tends to be limited to low stress/high-cycle fatigue, E-N fatigue can be applied to both LCF and HCF with high stress applications. The E-N fatigue approach uses local elastic-plastic strains to assess the fatigue damage. Maximum principal stress or strain should be used in the fatigue analysis due to brittle behavior of the AM materials. Residual stresses associated with the manufacturing process should be included in the fatigue analysis. This is especially true of tensile residual stresses because it can significantly reduce fatigue life. If these tensile residual stresses are possible and have not been captured within the fatigue data, then characterization of these residual stresses for inclusion in analysis should be performed. If this characterization is not possible or is impractical, conservatively assumed levels should be included in the analysis. Any fatigue assessment approach for non-metallic AM parts should be reviewed and approved by the delegated NASA Technical Authority.

Due to variations in material microstructures, porosity, surface finish, residual stress, and the multiaxial stress state during the service life of AM parts, fatigue life assessment presents a unique challenge. This is particularly true for AM parts used in space applications that undergo cyclic loadings throughout their service life. Since AM parts may be prone to fatigue-induced failure (i.e., resulting from rough surface finish or manufacturing defects), it is essential that a fatigue life assessment be performed for space applications.

6. NON-DESTRUCTIVE EVALUATION (NDE)

6.1 Non-Destructive Evaluation (NDE) for Additively Manufactured (AM) Parts

This section provides guidance and discussion on NDE of AM parts and how it supports NASA technical standard compliance. Refer to ASTM E3166, Standard Guide for Nondestructive Examination of Metal Additively Manufactured Aerospace Parts After Build, for extensive discussion on NDE procedures used to inspect metal parts made by AM. Existing requirements developed for parts made of conventionally manufactured materials may be tailored for AM parts by modifying the stated discontinuity types of interest to include discontinuities specific to AM (see Table 3, Application of NDE Methods for Detection of AM Discontinuities, for a list). NDE for NASA FC metal AM spaceflight hardware should be developed and qualified as Special NDE using the NDE techniques and other requirements covered in NASA-STD-5009. For NFC AM hardware, NDE serving as a process control NDE only (i.e., not used to support a damage tolerance fracture analysis) need not be quantitative or adhere to special NDE requirements of NASA-STD-5009. However, it is highly recommended to consult with NASA NDE subject matter experts (SMEs) on any NDE technique that is used independent of the requirement. Content in this section is included for both FC and NFC hardware. For NDE or AM flaw terminology, refer to ASTM ISO/ASTM 52900.

The following discussion is primarily for metallic AM parts after manufacturing and post-processing (machining, heat treatment, etc.). NASA-STD-6030 requires a quantitative NDE that meet the special NDE requirements of NASA-STD-5009 for all Class A parts. Process control NDE is required for all Class B parts. HIP is often applied during AM post-processing. A dedicated volumetric NDE occurs prior to the HIP to attempt to discover flaws that may be otherwise undetectable after the HIP for all FC, Class A, and primary structure parts. Pre-HIP NDE results should be provided to the delegated NASA Technical Authority and to the RFCB. If empirical data specific to the material and process under consideration exist to indicate that the HIP process not only closes flaws but also fuses them together (thereby eliminating the possibility of closed and hard-to-detect cracks), this may be provided as justification for not doing a pre-HIP volumetric NDE. If NDE results are used as the basis for a fracture analysis, the pre-HIP detected flaw size should be defined as the initial flaw size. Figure 3, Typical Class A Fracture Critical Hardware Acceptance Steps Highlighting Placement of NDE Steps, illustrates a typical flow for acceptance of Class A FC flight parts that receive both HIP and proof testing.

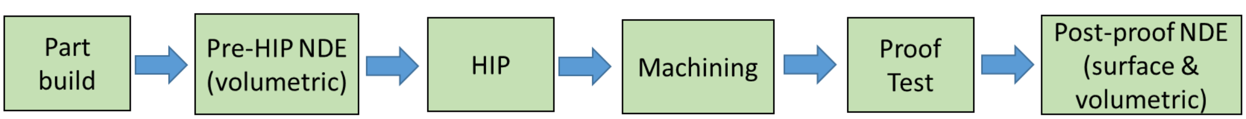

Figure 3—Typical Class A Fracture Critical Hardware Acceptance Steps Highlighting Placement of NDE Steps

Note that a process control NDE (non-90/95 probability of detection [POD] NDE) is not intended to be the basis for a damage tolerance fracture analysis or simulated service life test. Generally, a process control NDE always should include a visual and surface inspection to the extent possible on a part to screen for cracks and other surface-breaking flaws, as well as a volumetric inspection to screen for voids, inclusions, and porosity. The extent possible for the visual and surface inspection may typically be limited to areas that are accessible with minimal effort such as external surfaces.

FC parts that use a fracture analysis or test-based approach require NDE with a POD such that the initial flaw size used in a damage tolerance analysis (DTA) can be found a minimum of 90% probability and a 95% confidence level (90/95 POD). There are no differences in the procedural steps and requirements for developing an NDE technique meeting the 90/95 POD for an AM part compared to doing so for a conventional part made by casting or forging. Note that the entire part still receives a process control NDE if the 90/95 POD NDE is only needed in zoned, or targeted regions (see section 7.1.5 of this Handbook). If the 90/95 POD NDE covers the entire part, this may take the place of a process quality NDE. Assuming that a detected void or pore is instead a crack of equivalent size at the worst-case orientation is an acceptable conservative assumption for a damage tolerance assessment.

All crack-like detected flaws in Class A and FC AM parts should be reported to the delegated NASA Technical Authority to review, coordinate with other boards if necessary, and approve of the forward actions associated with the found flaws. Unless specifically developed and approved by the delegated NASA Technical Authority on a case-by-case basis, there is no preapproved or predefined accept/reject criteria such as flaw size or multiple flaw spacing for Class A and/or FC AM parts. AM parts that are neither Class A nor FC may utilize an NDE indication accept/reject criteria based on the expected manufacturing flaw size and intended use of the part. Any predefined NDE accept/reject criteria should be reviewed and approved by the RFCB. In cases where only a process quality NDE is applied, accept/reject criteria should be defined to verify that the process controls are functioning as intended. In-service reinspection may be used at the discretion of a project to extend part usage. In this case, for a part life extension, a DTA is run starting from the time of reinspection that assumes an initial crack size that corresponds to the reinspection of NDE capability.

One of the advantages of AM is the ability to make design-to-constraint parts with complex internal geometries and features, leading to the fabrication of shapes that cannot be produced using conventional processing methods. While these parts can be lighter and made with fewer parts, they can also present line-of-sight issues making them difficult to characterize by NDE. Two-dimensional (2D) X-ray images, for example, will show superimposed internal features reducing NDE flaw detectability. Depending upon part complexity, 2D X-ray[3] may be applicable. X-ray Computed Tomography (CT) is useful when feasible because it enables three-dimensional (3D) reconstruction of the part without superimposing features. It can be used to detect volumetric flaws within the material, as long as the minor dimension of the volumetric flaw is greater than X-ray CT resolution. However, the fundamental limitations of X-ray CT are the cross-sectional thickness and density of the part, which dictate the X-ray energy that must be used for penetration, in turn limiting the achievable resolution. Another limitation is the data acquisition time. The larger the part and the higher the requested resolution, the longer the scan will take. An additional limitation may arise from complex component geometry that can generate CT image artifacts that may obscure otherwise detectable defects.

CT is recommended as process control inspection here and can be validated to detect volumetric flaws. Special CT demonstration is needed to qualify detection of cracklike flaws reliably. Without special CT demonstration, CT generally is invalid for use as the basis of an initial flaw size in a DTA Guidelines and requirements for Special NDE demonstration are given in NASA-

[3] 2D X-ray may consist of film, Digital Radiography (DR), or Computed Radiography (CR) techniques.

STD-5009. Additionally, both NASA-STD-5009 and MIL-HDBK-1823A, Nondestructive Evaluation System Reliability Assessment, can be used for guidance on demonstrating a 90/95 POD capability. It is recommended to consult with NASA NDE SMEs when choosing between 2D X-ray and X-ray CT.

Many AM parts will require the use of multiple NDE techniques to achieve full coverage. A combination of radiography, penetrant testing (PT), eddy current testing (ET), or ultrasonic testing (UT) techniques may be common and should be considered. Surface inspection techniques may require the as-built surface to be improved to render a successful inspection (depending upon the flaw sizes of interest and the signal-to-noise ratio). Surfaces improved by methods such as machining or abrasion require etching prior to penetrant inspection to remove smeared metal. Note that removal of the as-built AM surface merely to a level of visually smooth may be insufficient to reduce the NDE noise floor due to the propensity for AM near-surface porosity and boundary artifacts.

In the application of NDE, the types of flaws (see ASTM E3166, Table 1) that are relevant to the AM process are considered. ASTM E3166 further defines manufacturing discontinuity classes and subclasses (see ASTM E3166, Table 2). The physics of the layered AM process tends to prohibit volumetric flaws with significant height in the build direction. The concern instead is for planar flaws such as aligned or chained porosity or even laminar cracks to form along the build plane, where the build plane is defined as the plane normal to the build direction. These types of flaws have a number of implications:

- Planar flaws are particularly well suited for growth.
- The primary flaw orientation of concern is defined. This may be meaningful in analysis or with detection methods dependent upon alignment with flaws.
- AM planar flaws will generally exhibit very low contained volume.
- The limited height of planar flaws can be demanding on radiography and CT.

There are longstanding NDE standard flaw classes for welds and castings. The flaws characteristic to these processes, although illustrative in the case of large wire-DED systems, will generally not be applicable to the AM process. *It is not recommended that welding or casting flaw quality standards be applied to AM hardware.* This implies that until an accepted AM flaw catalog and associated NDE detection limits for AM flaws are established, the NDE techniques and acceptance criteria may remain part specific.

Table 3 in this Handbook contains a summary of typical AM flaw types and corresponding applicability of various NDE techniques based on ASTM E3166. The labels shown in Table 3 are defined in the notes following the table. Selection of an NDE technique for use and quantification of its capability should involve further consultation with NASA NDE SMEs and necessary steps to certify as Special NDE per NASA-STD-5009.

Table 3—Application of NDE Methods for Detection of AM Discontinuities

Flaw Type	General Remarks	NDE Method Applicability ✓ = NDE method is applicable for flaw type * = Limited applicability. Recommend consulting NASA NDE SME.								
		X-ray			ET[6]	PT[5]	UT[7]	MT[8]	ST[9]	IRT[10]
		CT[1]	DR/CR[2]	Film[3]						
Lack of fusion (LOF) Laminar (along build interface), Cross layer, Cold lap, Oxide lap	A, E, G	✓	✓	✓	✓	✓	✓	✓	✓	✓
Build layer LOF voids: Fully/partially unfused build layer with trapped powder, laminar voids in build layer	B, E, G	✓	✓	N/A	✓	✓	✓	✓	✓	✓
Cross layer LOF voids	B, E, G	✓	✓	✓	✓	✓	✓	✓	✓	✓
Surface anomalies: Roughness, undercuts, underfill and/or overfill, crater, stair stepping, worm track, contour separation, surface breaking pores	F	✓*	✓*	✓*	✓	✓	✓	✓	N/A	N/A
Powder in voids/passages: Unfused/partially fused, trapped powder in voids and designed passages	C, F, G	✓	✓	✓	N/A	N/A	N/A	N/A	N/A	N/A
Porosity: Spherical and/or keyhole, micro-porosity, voids, surface-breaking porosity	F, G	✓	✓	✓	✓	N/A	✓	N/A	N/A	N/A
Inclusions: Low/high density inclusions, segregation, banding, planar inclusions	F, G	✓	✓	✓	✓	N/A	✓	N/A	N/A	N/A
Cracking:	D, E, G									

Flaw Type	General Remarks	NDE Method Applicability ✓ = NDE method is applicable for flaw type * = Limited applicability. Recommend consulting NASA NDE SME.								
		X-ray			ET[6]	PT[5]	UT[7]	MT[8]	ST[9]	IRT[10]
		CT[1]	DR/CR[2]	Film[3]						
Hot cracking, cold cracking, crater cracking, heat-affected zone cracking, tearing		✓*	✓*	✓*	✓	✓	✓	✓	✓	✓*
Low density (localized or gross)	F	✓	✓*,4	✓*,4	N/A	N/A	N/A	N/A	N/A	N/A
Out of tolerance dimensions	F	✓	✓*,4	✓*,4	N/A	N/A	N/A	N/A	N/A	N/A

Notes:

1	Discontinuity gap > resolution, Representative Quality Indicator (RQI) validation is recommended, limited by ability of X-ray to penetrate through the part
2	Discontinuity gap > total unsharpness, sensitive to flaw orientation, RQI validation is recommended, limited by ability of X-ray to penetrate through the part and superimposition of part features in the image
3	Discontinuity gap > geometric unsharpness, sensitive to flaw orientation, RQI validation is recommended, limited by ability of X-ray to penetrate through the part and superimposition of part features in the image
4	May work for simple geometry parts
5	Surface only
6	Surface and sub-surface
7	Coverage limited to outer layer (air on both sides) or limited by geometry. Surface textures can inhibit the passage of sound into the part.
8	Surface and sub-surface limited for ferromagnetic parts only
9	Subsurface planar flaws only
10	Primarily for detecting shallow near-surface planar flaws

General remarks:

A	Tight gap interface, need to know LOF opening morphology for Class A and FC
B	Discontinuity opening ≥ build layer thickness
C	Discontinuity thickness > build layer thickness
D	Need to know typical crack opening morphology and orientations for Class A and FC
E	None accepted under NDE acceptance criteria
F	Specified in acceptance criteria
G	Surface texture can obscure visibility of small flaws (relocate)

6.2 Non-Destructive Evaluation (NDE) Plan

NASA-STD-5009 requires that an NDE plan be created for FC parts. All required content for the plan that is described in NASA-STD-5009 is applicable to AM parts. An NDE plan for AM hardware should also include descriptions of how flaws in any orientation will be addressed (flaw orientation may render a flaw undetectable with certain NDE techniques). Specifically, flaws oriented with the build plane should be addressed in this regard. Generally, it may be challenging to detect internal flaws at all orientations. The NDE plan should describe either how this challenge is addressed or why it is not relevant for a given case.

7. FRACTURE CONTROL

Figure 4, Detail for Fracture Control Requirement Compliance, consists of an excerpt from Figure 1 highlighting the fracture control section. Recall that in Figure 1 and Figure 4, the diamond-shaped items are decision points and the boxes are required steps. Recall from section 4 also that by reaching one of the “Complete” points within the fracture control subsections of the overall flow chart, one would meet the intent of NASA technical standards containing strength, fatigue, and fracture control requirements.

Implementation of fracture control involves first determining the fracture classification of a part and then following test, analysis, or other verification steps specific to that classification. This section is structured with the assumption that the reader is first interested in determining the part fracture classification and then in what is required for that classification. This perspective is in line with NASA-STD-5019 and if followed according to guidance here, the intent of NASA-STD-5019 can be met. Note that the requirements in NASA-STD-5019 are all fully applicable for AM parts. If needed, implementation of fracture control according to this Handbook may be proposed as an alternative approach for the RFCB to review and approve in a given project. Note that the “AM part classification” as defined in NASA-STD-6030 is different from the fracture classification, and a given AM part will have both an AM classification and fracture classification.

Much of the discussion in section 7 is geared toward metallic applications. Non-metallic AM parts that are FC are not recommended. Methods for implementation of fracture control on non-metallic AM parts can follow guidance herein but should also include both consultation with and approval from the RFCB. Note that NASA-STD-6030 specifically disallows non-metallic Class A AM parts and an approved tailoring of that standard would be required for non-metallic Class A AM parts.

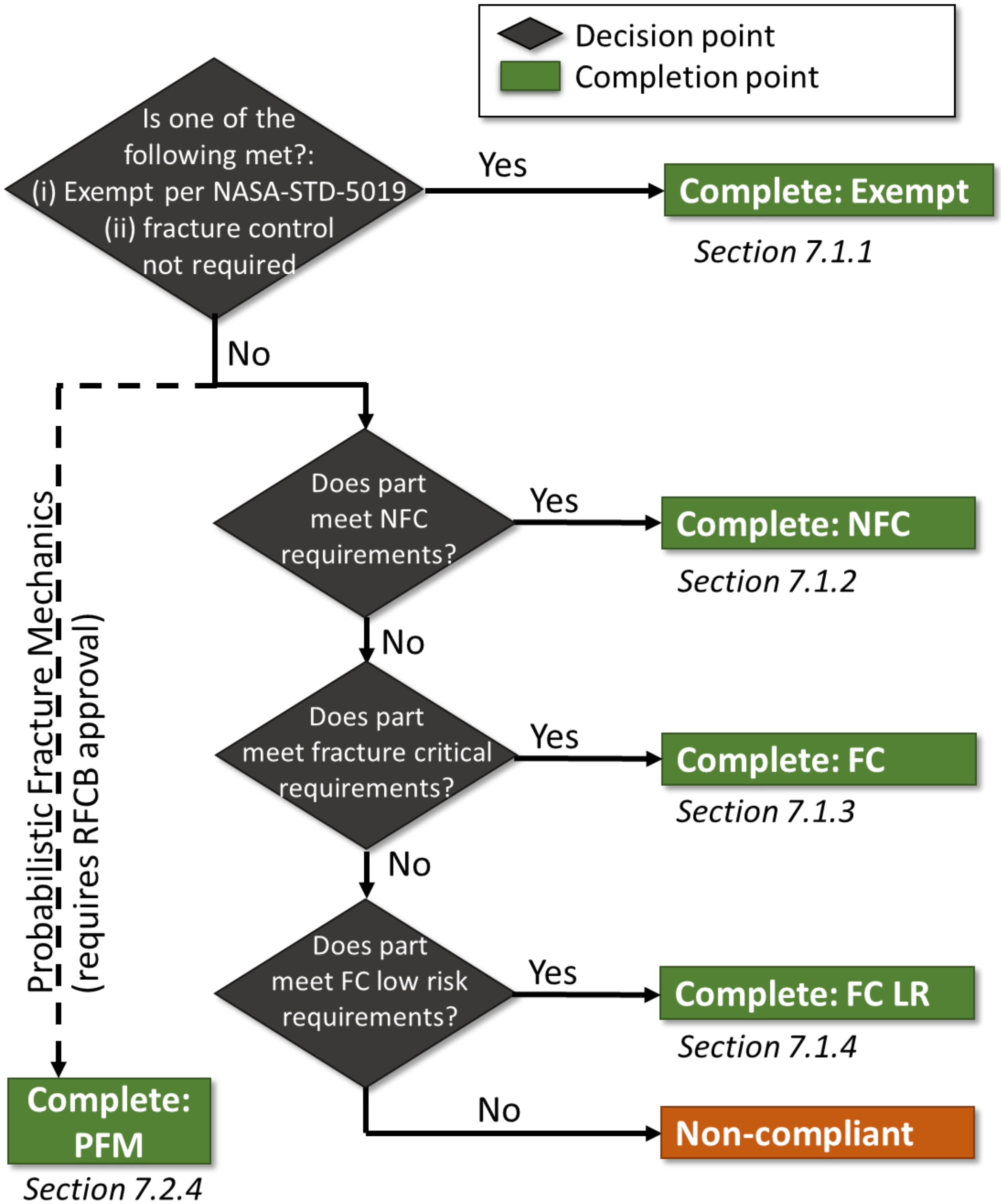

Figure 4—Detail for Fracture Control Requirement Compliance
(Corresponding Handbook sections shown below "Complete" blocks)

7.1 Fracture Classifications

7.1.1 Exempt Parts

Each part classified as exempt should fit into one of the following categories according to NASA-STD-5019:

- Non-structural parts with no credible failure mode caused by a flaw.
- Non-structural parts with no credible potential for causing a catastrophic event.
- Other non-structural parts approved by the RFCB for exempt status.

There is no difference in the exempt criteria for an AM part compared to other materials and manufacturing techniques.

Note that the NASA-STD-6030 definition of exempt is different from NASA-STD-5019. Here in this section, the NASA-STD-5019 definition has been utilized.

7.1.2 Non-fracture Critical (NFC) Parts

This section provides instruction on reaching the NFC "Complete" point in Figure 4. Generally, for each NFC classification defined in NASA-STD-5019, this Handbook contains either guidance on an AM-specific implementation or confirmation that there are no AM-specific considerations, and the requirement can be implemented as written in NASA-STD-5019. In cases where Class C is not permitted for a given NFC classification, the rationale generally is that the process controls for Class C are below a minimally acceptable level for an NFC structural part. A quick reference for all the NFC classifications is listed in Table 4.

Table 4—Quick reference for all NFC classifications

NFC Category	**Applicability to AM Classes**			**Section**
	Class A	**Class B**	**Class C**	
Low Released Mass	X	✓	✓	7.1.2.1
Contained	X	✓	✓	7.1.2.2
Fail Safe	X	✓	X	7.1.2.3
Non-Hazardous LBB	X	✓	X	7.1.2.4
Rotating Machinery	X	✓	X	7.1.2.5
Low Risk Fasteners and Shear Pins	X	X	X	7.1.2.6
Shatterable Components and Structures	X	✓	X	7.1.2.7
Sealed Containers	X	✓	X	7.1.2.8
Tools/Mechanisms	X	✓	X	7.1.2.9
Batteries	X	✓	X	7.1.2.10
Low Risk	X	X	X	7.1.2.11

7.1.2.1 Non-Fracture Critical (NFC) Low Released Mass

AM parts can be classified NFC low released mass if they meet the associated requirements stated in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC contained are as follows:

Class A: Not acceptable for NFC low released mass.

Class B: May be acceptable for NFC low released mass.
Class C: May be acceptable for NFC low released mass.

7.1.2.2 Non-Fracture Critical (NFC) Contained

AM parts can be classified NFC contained if they meet the associated requirements in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC contained are as follows:

Class A: Not acceptable for NFC contained.
Class B: May be acceptable for NFC contained.
Class C: May be acceptable for NFC contained.

7.1.2.3 Non-Fracture Critical (NFC) Failsafe

AM parts can be classified NFC failsafe if they meet the associated requirements in NASA-STD-5019. Fasteners classified as NFC failsafe should not have printed threads. Candidate AM classes per NASA-STD-6030 for NFC failsafe are as follows:

Class A: Not acceptable for NFC failsafe.
Class B: May be acceptable for NFC failsafe.
Class C: Not acceptable for NFC failsafe. (*Rationale: NFC failsafe parts may be structural despite failsafe nature; Class C material quality is not sufficient for a structural part.*)

7.1.2.4 Non-Fracture Critical (NFC) Non-Hazardous Leak-Before-Burst (NHLBB) Pressurized Components

AM parts can be classified as NFC NHLBB pressurized components if they meet the associated requirements in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC NHLBB pressurized components are as follows:

Class A: Not acceptable for NFC NHLBB pressurized components.
Class B: May be acceptable for NFC NHLBB pressurized components.
Class C: Not acceptable for NFC NHLBB pressurized components.

7.1.2.5 Non-Fracture Critical (NFC) Rotating Machinery

AM parts can be classified NFC rotating machinery if they meet the associated requirements in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC rotating machinery are as follows:

Class A: Not acceptable for NFC rotating machinery.
Class B: May be acceptable for NFC rotating machinery.
Class C: Not acceptable for NFC rotating machinery.

7.1.2.6 Non-Fracture Critical (NFC) Low Risk Fasteners and Shear Pins

AM parts should not be classified as low risk fasteners and shear pins per NASA-STD-5019.

7.1.2.7 Non-Fracture Critical (NFC) Shatterable Components and Structures

AM parts can be classified as NFC shatterable components and structures if they meet the associated requirements in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC shatterable components and structures are as follows:

Class A: Not acceptable for NFC shatterable components and structures.
Class B: May be acceptable for NFC shatterable components and structures.
Class C: Not acceptable for NFC shatterable components and structures.

7.1.2.8 Non-Fracture Critical (NFC) Sealed Containers

AM parts can be classified as NFC sealed containers if they meet the associated requirements in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC sealed containers are as follows:

Class A: Not acceptable for NFC sealed containers.
Class B: May be acceptable for NFC sealed containers.
Class C: Not acceptable for NFC sealed containers.

7.1.2.9 Non-Fracture Critical (NFC) Tools/Mechanisms

AM parts can be classified NFC tools/mechanisms if they meet the associated requirements in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC tools/mechanisms are as follows:

Class A: Not acceptable for NFC tools/mechanisms.
Class B: May be acceptable for NFC tools/mechanisms.
Class C: Not acceptable for NFC tools/mechanisms.

7.1.2.10 Non-Fracture Critical (NFC) Batteries

AM parts can be classified as NFC batteries if they meet the associated requirements stated in NASA-STD-5019. Candidate AM classes per NASA-STD-6030 for NFC batteries are as follows:

Class A: Not acceptable for NFC batteries.
Class B: May be acceptable for NFC batteries.
Class C: Not acceptable for NFC batteries.

7.1.2.11 Non-Fracture Critical (NFC) Low Risk

The NFC low risk fracture classification is not permitted per NASA-STD-6030 for AM parts. If there is a strong design motivation for use of AM parts in this family, an alternative approach that meets the intent of NASA-STD-5019 can be adopted based on the guidelines in section 7.1.4 of this Handbook.

7.1.3 Fracture Critical (FC) Parts

Depending on the circumstances or design approach, a variety of methods for implementation of FC requirements may be considered. This section contains guidance for specific hardware types and reaching the FC "complete" point in Figure 4.

7.1.3.1 General Approach for Fracture Critical (FC) Parts

Unless a part falls into one of the hardware categories described by sections 7.1.3.2 through 7.1.3.7 where a specific approach is called for, FC parts are required to be shown damage tolerant to undetected flaws either by analysis or by test. This necessitates application of a 90/95 POD NDE (with the exception alternative approaches such as the one outlined in section 7.1.4). Methodologies for performing this assessment on AM parts are described in section 7.2. Alternatively, with RFCB approval, the PFM methodology may be used instead of a damage tolerance assessment as described in section 7.2.4 of this Handbook.

7.1.3.2 Fracture Critical (FC) Pressure-Containing Parts

All pressure-containing parts, including pressure vessels, lines, fittings, other pressurized components, habitable modules and volumes, other pressurized fluid containers, and pressurized structures are required to have a proof test performed with proof test factors on maximum design pressure (MDP) applied as defined in section 7.2.3 of this Handbook. NASA-STD-5019 contains requirements for these types of hardware that largely also include the need for a proof test. All FC or Class A AM pressurized flight parts must receive a proof test no matter the circumstances (see section 5.1.2 of this Handbook). A damage tolerance assessment is recommended (see section 7.2.1) for all pressurized AM parts even in cases where a proof test is applied according to section 7.2.3. If a damage tolerance assessment is performed, the initial flaw should be based on 90/95 POD NDE (volumetric and surface). If a HIP is performed, NDE should occur pre-HIP (volumetric) and post-proof test (volumetric and surface). If no HIP is performed, the only NDE required for fracture control is post-proof test (volumetric and surface).

Note that NASA-STD-6030 does not permit an AM part to be classified using the category of "fracture critical lines, fittings, and other pressurized components" per NASA-STD-5019. If there is a strong design motivation for use of AM parts in this family, an alternate approach that meets the intent of NASA-STD-5019 may be proposed for RFCB review and approval that is based on a combination of the FC low risk criteria described in section 7.1.4[4] and the proof testing guidelines described in section 7.2.3 of this Handbook. Alternatively, simply following the FC classification and doing both an NDE and damage tolerance assessment is another

[4] Note that the FC: alternative approach criteria in section 7.1.4 prohibits pressurized components when using this classification. Nevertheless, an alternate approach as described here may be proposed to the RFCB for review and approval.

acceptable approach. In this case, if NDE and damage tolerance are performed, a proof test still should occur.

7.1.3.3 Fracture Critical (FC) Rotating Hardware

AM parts can be classified FC rotating hardware if they meet the associated requirements in NASA-STD-5019.

7.1.3.4 Fracture Critical (FC) Fasteners

Threaded FC fasteners made from AM materials are not permitted. FC AM fasteners are not recommended as a general design practice.

7.1.3.5 Fracture Critical (FC) Shatterable Components and Structures

AM parts can be classified FC shatterable components and structures if they meet the associated requirements stated in NASA-STD-5019.

7.1.3.6 Fracture Critical (FC) Tools, Mechanisms, and Tethers

AM parts can be classified FC tools, mechanisms, and tethers if they meet the associated requirements stated in NASA-STD-5019.

7.1.3.7 Fracture Critical (FC) Batteries

AM parts can be classified FC batteries if they meet the associated requirements stated in NASA-STD-5019.

7.1.4 Fracture Critical (FC): Alternative approach

This category describes a potential fracture control alternative approach intended for hardware that is precluded from the NFC Low Risk category by NASA-STD-6030 [AMR-15], but additional activities, as delineated below, may provide adequate mitigations for catastrophic failure. However, there is highly critical hardware that is excluded from this category. This approach does not apply to the following hardware: pressure vessels, habitable modules, pressurized component in a pressurized system containing hazardous fluids, high-energy or high-momentum rotating components, hazardous fluid containers, solid rocket motor cases and nozzles, or primary thrust structures. These types of hardware would need to follow standard fracture control methodologies as specified in NASA-STD-5019. This alternative approach may include parts whose failure results in a catastrophic hazard (although parts where this is not the case may use this approach as well). This approach as described in this Handbook may be submitted to the RFCB for approval as an alternate approach that meets the intent of FC guidelines described in this Handbook and the requirements defined in NASA-STD-5019. It is strongly recommended that the decision to adopt the FC Alternate Approach be agreed on early in AM part development. If a part fails to meet the alternate approach requirements, changes in the process controls and verifications, including quantitative NDE, may affect schedule and resources, or even viability of the AM part. See section A.1 in Appendix A of this Handbook for an example.

The FC: alternative approach may be used if all criteria listed below are met:

a. The part is built using a reliable manufacturing process that is fully qualified and in compliance with NASA-STD-6030.

b. The part is a non-pressurized metallic component.

c. The part receives a proof test in cases where failure of the part would result in a catastrophic hazard.

d. The part has an "AM risk score" per section 4.3.2.2 of NASA-STD-6030 Table 4 of less than 5.

e. The maximum concentrated limit stress is lower than 50% of the yield strength.

f. The maximum concentrated limit stress is lower than 30% of the ultimate strength.

g. Metallic parts have a material property ratio of $K_{IC}/F_{ty} > 1.66\ mm^{1/2}$ ($0.33\ in^{1/2}$) and do not have sensitivity to Environmentally Assisted Cracking (EAC), Hydrogen Environmental Embrittlement (HEE), Sustained Load Cracking (SLC), or Stress Corrosion Cracking (SCC) as defined in NASA-STD-6016

h. One of the following is met:

(1) Maximum concentrated stress is less than the fatigue limit where fatigue limit (see section 3.2 for definition) corresponds to the orientation with a minimum value and is defined according to material testing.
(2) A DTA from a 0.127 mm (0.005 in) initial crack that conservatively accounts for the effects of notches and mean stress and shows a minimum of four complete service lives with a factor of 1.5 on alternating stress.

i. NDE performed includes visual, volumetric, and surface inspection. Surface inspection methods such as ET or PT may be used. If a part's rough surface finish is such that a meaningful surface inspection is not possible, rationale for this approach may not exist. If useful, consider a zoned classification (see section 7.1.5) where this alternative approach is limited to areas where a meaningful surface NDE is possible. All crack-like NDE indications are reported to the RFCB.

If a part satisfies the FC: alternative approach criteria:

a. A quality screen level of NDE fidelity is sufficient (i.e., no need for 90/95 POD).

Note: a full-coverage quantitative NDE inspection is required by NASA-STD-6030 for Class A AM components. When this approach is selected for a component and approved by the RFCB, the NASA-STD-6030 requirement for quantitative NDE may be tailored by documenting the change in the Integrated Structural Integrity Rationale section of the PPP and submitting the PPP for approval by the appropriate NASA Technical Authority.

b. A damage tolerance assessment based on 90/95 POD NDE or by other means as is typically required for FC parts is not necessary.

7.1.5 Zone-based Approaches

Throughout this Handbook, the reader should consider the use of zone-based approaches to Fracture Control when necessary. Zone-based approaches in this case are defined as applying different fracture control mitigations (i.e., inspections, analysis, and testing) to different regions of a single part. Note that a zoned part defaults to the highest fracture classification (e.g., fracture critical) to invoke part-level configuration management, traceability, non-conformance disposition, process controls, drawing markings, etc. This strategy can be useful in demonstrating part compliance where certain local attributes may rule out use of a particular approach for the entire part. The best way to approach this strategy is to zone areas of parts such that more stringent mitigations are applied only where necessary. For example, some regions of a single part may be redundant or very lightly stressed. For these regions are classified NFC, the need for 90/95 POD NDE, fracture analysis, or proof testing may be relaxed or eliminated. By restricting the more stringent mitigations to only a local region where needed, a part may even be designed to accommodate the necessary NDE or proof testing in this area. Zone-based approaches are frequently used but must be fully disclosed and approved within the fracture control documentation (i.e., Fracture Control Plan, Fracture Control Summary Report).

To help illustrate the application of zone-based, consider two examples.

1. An AM bracket exists whose failure would result in a catastrophic hazard and is classified as a fracture critical part. The bracket has a local region of high stress, and low stress elsewhere. The high stress region receives focused post-production machining and etching to allow a 90/95 POD surface NDE to be performed. The low stress regions receive minimal mitigation with the logic captured in the fracture control documentation for RFCB approval. A DTA of the FC region only is performed based on the surface and volumetric NDE. The part is labeled FC on the drawing and in program documentation.
2. A non-rotating AM engine component exists that contains intricate internal features that are inaccessible for a 90/95 POD NDE. The part is designed such that the features inaccessible to 90/95 POD NDE are redundant and lightly stressed and can be logically subject to minimal mitigations that are documented for approval by the RFCB. Elsewhere in the part without redundancy, such as the exterior surface, the stringent

mitigations are applied, 90/95 POD NDE is possible, and a DTA is performed. The part is labeled FC on the drawing and in program documentation.

7.2 Test and Analysis Techniques

Section 7.1.3.1 indicates that FC parts should be shown to be damage tolerant by (1) damage tolerance demonstration (test or analysis) or (2) use of a PFM approach. All damage tolerance assessments show that undetected crack-like defects will not cause a catastrophic hazard within four service lives. This section contains guidance that is applicable beyond AM parts specifically and may be found in other standards and documentation to some extent. Guidance on test and analysis techniques for fracture control such as fracture analysis methods and proof test methods is included in this Handbook to consolidate and tailor the details specifically for AM applications.

7.2.1 Damage Tolerance Demonstration

Damage tolerance demonstration should be performed according to NASA-STD-5019 as summarized in the selected excerpts from both NASA-STD-5019 and NASA-STD-5009 listed below (note: “shalls” and the italic texts are included here as direct quotes from these standards):

a. *The damage tolerant demonstration shall be based on an initial flaw size that could be present in the part. This flaw size shall be established by NDE or proof test* screening as described in section 7.2.3.2. Critical Initial Flaw Size (CIFS) analysis, explained in section 7.2.2, to inform the necessary NDE capability is another approach that can be utilized in lieu of determining the initial flaw size.

b. If NDE is used for screening, the maximum crack size that the NDE technique would miss (i.e., the size corresponding to the POD) shall be used as the initial flaw size in the analysis. The NDE technique shall detect the initial crack sizes with a 90% probability of detection at a 95% confidence level.

c. *Analysis or test shall consider all significant loadings, both cyclic and sustained, that the part can experience during* the service life.

d. Loads from these phases shall be considered for each mission the hardware may undertake.

e. Damage tolerant parts shall be shown to have a service life factor of at least four and subsequently have a positive margin on toughness.

f. Account for the effects of environments and flight hardware structural conditions to simulate performances throughout the specified service lifetime. If tests are not performed in the operational environment, test levels are adjusted via an ECF.

Any damage tolerance demonstration should define the initial flaw at the worst-case location in the worst-case orientation and consider the possibility of multi-site crack growth in all cases. In practice, this may entail performing crack growth analyses using the 90/95 POD NDE flaw size at all stress concentration locations even though only one qualifies as the most critical location for the part. Analysis parameters such as geometry and NDE capability should be defined as location-specific inputs for each location considered. Rationale should be provided (1) if there were multi-site crack growth, as crack growth in one location would not affect part stresses to the degree that fracture analysis assumptions in another location are invalidated, or (2) if residual static strength would be exceeded in the presence of multiple sub-critical growing cracks. Another approach may be to assume multi-site initiation has already linked together across a region thought to be susceptible and use a larger initial flaw size. This approach is valid if the far field stress used for the fracture analysis remains unchanged after assumptions about multi-site crack linking.

Consideration of multi-site crack growth in the context of damage tolerance as described here (i.e., assume an initial flaw) would envelop concerns of multi-site crack initiation in the context of fatigue. Note that a direct consideration of multi-site crack growth in this way may go beyond what is typically expected for compliance to NASA-STD-5019 but is used in this case because of life being a more likely design driver for AM parts. The AM part manufacturer should make quality sheets available to the fracture analyst so that insight on the nature and type of possible defects may be considered when defining the worst-case initial flaw.

Consideration of multi-site crack growth aside, definition of the worst-case flaw location may not be straightforward. It can be founded on stress concentrations in a FEA, but this alone may not be a sufficient criterion. Material thickness, temperature, environmental effects, peak static, and dynamic stresses, as well as NDE flaw size also have a direct influence on location-specific flaw criticality. Analysts may also want to assess near-surface embedded cracks that would break through quickly during service to become surface cracks. Following the guidelines discussed above to address concerns on multi-site crack growth also provide a useful outline in determining critical flaw location in general.

The analyst should clearly define in their damage tolerance assessment what specifically constitutes a catastrophic hazard. For example, formation of a through crack may constitute a catastrophic hazard in some cases even though residual strength is not exceeded and unstable crack propagation has not yet occurred. For single-load events or those determined to have negligible crack growths, no-growth failure criterion may be applied in cases where material toughness data are available but crack growth data are not available. For multi-cycle load events, no-growth failure would be verified if fatigue crack growth threshold is available.

In instances where in-service inspection is performed, provided that the NDE is 90/95 POD quality, a damage tolerance assessment may be reset and start with the flaw size despite any previous service life prior to the reinspection. In-service reinspection combined with DTA may be used at the discretion of a project to extend part usage.

Use of the NASGRO® fracture analysis software is acceptable for performing analyses on metallic AM parts. Any use of NASGRO® for AM will involve defining a new material or demonstrating thoroughly that an existing material is enveloping for the AM application (see section 7.3). If an AM part has anisotropic fracture properties, this information along with a description of how it is addressed in an analysis should be provided to the RFCB.

7.2.1.1 Damage Tolerance Demonstration by Analysis

DTA should be performed such that the initial crack is placed at the most critical location in the most critical orientation. Determination of the most critical location may occur as described in section 7.2.1. DTA for AM parts is conducted similar to other metallic materials. Material properties used in the analysis such as toughness and crack growth behavior are approved by the RFCB. (Note that at the time this Handbook was written, there are no material or fracture properties in the NASGRO® database for AM material.) Material properties are representative of the constraints, service environment, product form, material orientation, and processes used in the manufacturing of the part. DTA methods for non-metallic materials should be reviewed and approved by the RFCB. If non-metallic DTA methods are deemed to involve too much uncertainty, the test-based approach described in section 7.2.1.2 may be needed. Note that parts made of polymeric materials cannot be Class A per NASA-STD-6030; while not recommended, there is no restriction on classifying other non-metallic parts as FC.

The following analysis features are recommended for use in a DTA of AM parts[5]. If positive margins on life cannot be shown with these conservative assumptions in place, provide test data to quantify and justify a lower level of uncertainty in the fatigue crack growth rate (da/dN) and toughness properties used in the analysis:

- To account for the scatter in the fracture properties, a knockdown (at least 20% is recommended) should be applied to (a) the typical fracture toughness and (b) the typical crack growth threshold. If data are obtained from specimens representative of the actual hardware to quantify the scatter, lower bound properties may be used and no knockdown is needed.
- A 1.25 multiplier factor applied to da/dN data (equivalent to 20% shift in curve upwards) to account for uncertainty in the data unless evidence is provided to show that the lower bound material properties (upper bound in growth rates) are used.
- Plane strain fracture toughness, K_{IC}, for through cracks and K_{Ie} (effective fracture toughness for a surface or elliptically shaped crack) for part-through cracks is used as the fracture toughness in the analyses; and the effects of thickness correction are ignored unless sufficient data is provided to the RFCB to show otherwise. If NASGRO® is used, set the B_k parameter equal to zero (B_k is a parameter in NASGRO® that accounts for the thickness effect on fracture toughness).
- Residual strength, fracture stability, and through crack formation are assessed in parallel throughout a crack growth analysis.

[5] Not needed necessarily in a PFM approach as described in section 7.3.

Ideally, residual stress should be understood and defined in the fracture analysis. Use of the "suppress closure" option in NASGRO® is not recommended as the only means to account for tensile residual stress. See section 5.2.3 for guidance and expectations related to residual stress.

7.2.1.2 Damage Tolerance Demonstration-by-Test

A simulated service life damage tolerance test may be useful in cases where a credible analysis method is not available due to any of the following: (a) material property characterization is not feasible, (b) damage mechanisms are not well understood (such as effects of surface finish or behavior of intricate geometric features), (c) linear-elastic fracture mechanics (LEFM) analysis tools are not valid, or (d) residual stresses are not well understood. Non-metallic AM parts, where fracture analysis methods are not mature, may necessitate a test-based approach. Additionally, the test-based approach described in this section may be used simply to confirm analysis results where some uncertainty exists or to demonstrate that an existing analysis predicting failure is overly conservative.

The success criteria of a damage tolerance simulated service life test is to demonstrate that a defect at a critical location does not grow to cause a catastrophic hazard within four service lives. A damage tolerance demonstration-by-test approach uses dedicated test articles (i.e., not flight parts) and must meet both items listed below:

1. Has a capability to insert realistic crack-like defects (i.e., representing the shape and size of possible undetected cracks in the hardware) at the orientation and location of interest. This includes the ability to "pre-crack" the inserted flaw to achieve a sharp natural crack tip prior to the simulated service life test.
2. The simulated service life test should utilize a flight-like test article that is either a full-scale build of the part or representative of the full-scale build and features.

Regarding item one (1), inserted defects should consist of realistic crack-like flaws at the worst-case location and orientation (see section 7.2.1 for discussion on worst-case definition). Multiple defects can be present in one test specimen if there are multiple locations of concern and there is sufficient spacing between defects so that they do not affect one another. Defects may be introduced deliberately in the AM process if this method is shown by test to produce a defect that satisfies item one (1) above. If pre-cracking is not possible, this may rule out use of a damage tolerance demonstration-by-test approach. The necessity for pre-cracking is that an artificial flaw may have a blunt crack tip and give an indication that it is more damage tolerant than a naturally occurring sharp-tipped flaw would be. Dedicated simplified test coupons may be useful in developing a pre-cracking strategy for the test article.

Item two **(**2**)** listed above may include testing part families enveloped by a single coupon configuration. It is necessary that any simulated service life test coupon be flight-like and representative of the full-scale part as stated. It may be that one test coupon configuration is representative of numerous other "part family designs" that share at a minimum the same material, process controls, service environment, loading, residual stress distribution, and general design. Use of "part family" test strategies should be developed and approved by the RFCB.

Technical evidence is necessary to indicate that, from a fracture mechanics perspective, the tested part envelops others that are in-family.

In cases where both proof test logic and 90/95 POD NDE are not available, conservative alternative methods may be considered to define initial flaw size in a simulated service life test. If there is empirical data and experience specific to the part process control development on what types and sizes of flaws are likely/possible, the initial flaw may be defined assuming an upper bound or basis size based on this information. The defined size in this case should be based on quantified and observed data from part process control and qualification development. Use of this approach should be limited because collected data on flaw sizes may not contain a critical rogue flaw (see section 7.3). Similarly, the initial flaw size for a simulated service life test may instead be defined using a conservative bound for an NDE method that is available. For example, if a through crack is not catastrophic, assuming a through flaw would bound uncertainties on NDE capability in the depth direction. Note that NDE capability uncertainties in the length direction (on the surface) are not addressed by a through crack assumption. If the critical initial length determined by analysis of a through crack is not thought to be feasible considering process controls in place, this may be a valid initial flaw size assumption to use in a simulated service life test. Any approach using this method should be reviewed and approved by the RFCB.

Growth of inserted defects should be quantified at the end of the test by destructive or non-destructive means such that both the initial and final defect sizes can be obtained. The NDE technique used for post-test inspection should be the same that is used to define the initial flaw size. If destructive means are required to examine flaw growth, it is recommended to attempt to collect and report residual strength data in the process of coupon destruction. If it is discovered that inserted flaws were not present or inserted as intended, the test should be repeated with the correct flaw configuration. If a fatigue crack growth analysis is used as part of the part qualification methodology, flaw growth observed in a full-scale simulated service life test should be compared to toughness or fatigue crack growth coupon test data to demonstrate similitude between flight unit behavior and expected material fracture behavior. DTA may be used to make this comparison, where if successful, the analysis methodology could reliably be applied throughout the part as needed. On the other hand, if this comparison indicates a poor understanding of expected defect behavior compared to actual behavior, the source of the discrepancy should be discussed with the RFCB. Regarding analysis of the test, consideration should be made to ensure specific analysis of the actual loads and configuration in the test in case there are differences (known or unknown) in the test versus flight conditions.

Test run-out should be considered when planning the test. Test run-out may be advantageous to determine the actual life capability beyond the required scatter factor of four. Alternatively, a residual strength check after four lives may be desired. Test-based evidence for life or strength margin above and beyond the requirement can be useful in supporting efforts where some uncertainty exists. It is recommended to perform test runout either in terms of life or residual strength if possible. Rationale should be provided for why test runout will not interfere with obtaining the important results of the test. For example, if a through crack is considered the event that would cause a catastrophic hazard, this may not be detectable in the test if unstable crack growth does not also occur. If the test is runout until unstable fracture occurs and then a post-test inspection is performed, it may not be known if the through crack formed before or after four service lives. Another example of a case where test runout may interfere with results is one where a no-growth criterion is used (i.e., no growth is desirable within a certain number of service lives e.g., 4). In this case, crack growth results are needed at exactly four service lives.

An actual full-scale test coupon may not be needed on a case-by-case basis. Simulated service life damage tolerance test coupons are representative of the flight part at a minimum in terms of material, process controls, service environment, loading, residual stress distributions, and general design. This list of attributes should serve as the criterion for whether or not a full-scale test coupon is warranted. If a full-scale test coupon is needed, a minimum of three test coupons and data points should be obtained. To reduce the scope and cost of testing in this scenario, one alternative would be to obtain toughness, strength, and crack growth data from smaller scale material coupons (see section 7.3) and use only a single, full-scale test to validate fracture analysis predictions of the test. Full-scale testing is recommended for pressurized components as it offers a means to recreate realistic service loading in a laboratory test setup.

7.2.2 Critical Initial Flaw Size (CIFS)

The notion of CIFS can be useful either as an alternate perspective in approaching a damage tolerance assessment or as a tool to aid in AM part optimization. The CIFS can be interpreted to be the initial crack-like flaw size with the worst-case orientation that is predicted to precisely meet both the defined service life times (×) the life/scatter factor and the residual strength requirements (i.e., a MS equal to zero on damage tolerance of the flaw). CIFS is similar to DTA, and all the considerations required for DTA must be considered for CIFS analysis as well. The main difference between CIFS and DTA is that, in the DTA, the initial flaw size is given (by NDE or proof test screen); and the analysis is performed to determine the number of lives and critical flaw size that the hardware survives. In a CIFS analysis, the (critical) initial flaw size is unknown but the number of required lives is given. Just as in a DTA, the CIFS may be assessed for a specific location, such as one finite element in a model; or it may be assessed across an entire part at many locations to find the worst-case.

One way to make use of the CIFS in an AM part application is to compare the CIFS with the flaw size determined by NDE flaw size to check if the NDE method used for inspection is sensitive enough to detect the flaws sizes at all locations in a part that could cause failure. This can be useful if NDE capability or geometric complexity varies across a part. If NDE can detect a flaw size smaller than the CIFS, this suggests that the NDE method can detect a crack that could lead to failure of the part.

The CIFS is determined with an iterative fracture analysis. The following procedure may be automated (for example, by using a DTA tool such as NASGRO®) or performed manually if practical. First considering a single location, a guess for the initial flaw size should be input to a crack growth analysis model. The model is then run using the part’s service life load spectra. If the part fails prior to service life times (×) the scatter factor, the initial flaw size should be reduced and the analysis run again. If the part fails after service life times (×) the scatter factor, the CIFS was underestimated; and the initial flaw size should be increased before running the model again. The initial flaw size iteration sequence is usually stopped when it falls within a predefined allowable flaw size tolerance band. In this case, the initial flaw size is considered as CIFS. It should be noted for the sake of simplicity the above procedure was explained for a through crack. In case of surface or embedded cracks, the effects of depth to length ratio should be considered as well. If worst-case orientation is not obvious, this parameter should be varied as well (one option is to define the worst-case orientation according to the local maximum principal stress angle).

It may be cumbersome to perform this iterative analysis procedure across an entire part, especially if it is performed manually. If mapping the CIFS at all locations in a part (i.e., in all finite elements within a model) is desired, an automated computational tool is likely required. Even in cases where such a tool is available, it may be reasonable for the user to identify numerous “locations of interest” to limit CIFS calculations. Locations of interest may include those that have stress/strain concentrations, exposed to temperature or aggressive environments, geometric features that may limit crack growth, local consequences of failure (i.e., a fluid leak may be an issue at one location and not others), different materials, different NDE methods, or any additional source of uncertainty driving greater scrutiny at a specific location such as process control inconsistencies, loads uncertainty, or known higher likelihood for defects.

7.2.3 Proof Test

A proof test is a ground-based test on a flight part. This test causes stresses in the part exceeding flight limit stresses by a predesignated “proof test factor” multiplied by an ECF if the service environment is more severe than the test environment. In pressure proof tests, the proof pressure is the product of the proof test factor, the ECF, and the MDP. In mechanical proof tests, the proof load is the product of the proof test factor, the ECF, and the limit load. This might even require multiple load cases to cover all the locations of interest. The ECF is calculated as the strength ratio between test and service environments (maximum of yield and ultimate ratio) or Young's modulus ratio (for buckling considerations) between test and service environments. The following sections describe two ways in which a proof test can be used: proof test as part of acceptance testing and proof test for flaw screening. Note that per Table 1, all AM parts, with the exception of “metallic structure analysis-only” (Class C), require a proof test as part of acceptance testing independent of fracture control. Proof testing to meet fracture control requirements may or may not be required depending on the exact fracture classification that is used (see section 7.1.3). If proof testing is used for fracture control, it may involve higher proof test factors than are listed in Table 1.

All proof-tested parts receive post-proof test NDE (both surface and volumetric) regardless of the details and specific role of the test. Pre-proof test NDE may also be performed if desired and is used to indicate that the part is sound, produced as expected, and ready to be proof tested. The post-proof NDE is to observe any defect growth (if it is compared with the pre-proof NDE) or observe new defects that were opened up and made detectable by the proof test. It is the post-proof NDE that provides the actual justification to fly the part or substantiate a damage tolerance assessment.

In any application of a proof test, the test should be accompanied by an analysis that shows spatially the proof test coverage. Proof test coverage for a given location is defined as the test demonstration factor (TDF) and is defined in Equation 4:

$$\boldsymbol{TDF = \frac{Proof\ Test\ Stress}{Limit\ Load\ Stress\ \times\ Proof\ Test\ Factor\ \times ECF}} \qquad \text{(Equation 4)}$$

where *Proof Test Stress* may be obtained by analysis and refers to the actual stress achieved in the proof test. All locations of interest (high stresses, critical geometry, at risk for defects, NDE limitations, etc.) should see a TDF > 1.0 even if this level of coverage is not achieved across every location in a part. The TDF should be provided to the delegated NASA Technical Authority and the RFCB as part of the fracture control methodology rationale. Any locations in the part that do not have a TDF > 1.0 should either have alternate rationale in these locations (i.e., such as NDE + damage tolerance) or have a local zoned NFC classification. If there are locations where a critical initial flaw could go undetected by both a proof test and NDE, a hardware redesign or rethinking of the fracture control strategy may be needed.

Equation 4, and the above paragraph are essentially pointing out that the whole structure will, in reality, see different proof factors, but it is important that the required proof factor be reached, hence the statement TDF > 1.0. In other words, calculating the proof pressure or proof load is the input to the proof test, while the TDF calculation is like the output of the proof test.

A proof test may introduce concentrated stresses above yield strength, but it should not cause detrimental deformation. If there are concentrated proof stresses present that are greater than yield strength, this should be reported to the delegated NASA Technical Authority and/or the RFCB.

7.2.3.1 Proof Test as Part of Acceptance Testing

See Table 1 for acceptance proof test factors. The purpose of the acceptance test is to demonstrate that the part can survive a quasi-static load greater than it will ever see again in service, thereby providing confidence that a static strength failure later is unlikely to occur (but still possible if continued fatigue crack growth occurs after the proof test, further degrading the residual strength).

For FC pressurized hardware where the stresses due to pressure are dominant (all other stresses are not greater than 20 percent of the stresses due to pressure), perform a proof test that achieves minimum stress factors described below at all locations of concern. Any pressure-containing region or feature supporting a pressure-containing region should not experience stresses exceeding yield strength during the proof test.

The proof test factor and proof test pressure may be defined as shown in Equations 5 and 6:

$$\boldsymbol{Proof\ Test\ Factor = \frac{1 + burst\ factor}{2.0}} \qquad \text{(Equation 5)}$$

$$\textbf{\textit{Proof Test Pressure}} = \textbf{\textit{MDP}} \times \textbf{\textit{ECF}} \times \textbf{\textit{Proof Test Factor}} \qquad \text{(Equation 6)}$$

where the *burst factor* refers to the ultimate strength factor in Table 1. Following the proof test on pressurized parts, a leak test should be performed at a pressure equal to 1.0 × MDP. Leak test methodologies and success criteria used for non-AM parts are equally applicable for AM parts. Note the following:

1. Proof Test Pressure defined by Equations 5 and 6 does *not* include use of the AM Factor from Table 1; in general proof test procedure should be compliant with NASA-STD-5019.
2. Leak test methodologies and success criteria used for non-AM parts are equally applicable for AM parts; in general leak test procedure should be compliant with NASA-STD-7012.

If the pressure is not the dominant load case, then the proof test factor should be adjusted in such a way that all the stress components caused by proof testing are greater than the stresses caused by all combined service loads by a factor of 1.5.

7.2.3.2 Proof Test for Flaw Screening

A 90/95 POD NDE is always the preferred flaw screening method. In cases where this is not feasible, it may be possible to use a proof test for flaw screening to define an initial flaw size for a DTA or simulated service life test with approval of the RFCB. It is especially discouraged to be used for ductile metallic parts, thin parts, or parts that have an R-curve effect, as the proof loads may cause stable crack growth in the material. Another situation where the proof test for flaw screening is not encouraged is in cases where strain or displacement is the driving force (e.g., liner of COPV). In such cases, the proof test screening method needs specific scrutiny and is less likely to work. Moreover, an elastoplastic analysis might be needed to accurately calculate the screened flaw size. The 90/95 POD NDE is a more appropriate method for these situations to define an initial flaw size. However, if a proof test method is strongly needed to be used to screen for flaws (commonly referred to as “proof test logic”), the planned approach and anticipated effectiveness must be approved by the RFCB. In general, parts that are thick (i.e., high constraint) or brittle are the best candidates for proof test flaw screening. This section describes points that need to be considered when the proof test is used for screening flaws. The proof test procedure should be similar to that described in section 7.2.3.

The critical crack size (CCS) in the proof test determined using LEFM analysis, CCS_{proof}, may be used as the initial flaw size in a DTA or simulated service life test if it also can be shown to survive 4× the remaining service life after the proof test, and it can be verified that LEFM is applicable. A useful way to make this judgement is to apply the concept of the CIFS described in section 7.2.2. First, use LEFM analysis to determine the CIFS for 4× the service life, excluding proof test loads. This quantity will be defined as $CIFS_{post\text{-}proof}$ as it represents the CIFS for service loads that occur post-proof only. CCS_{proof} must be less than $CIFS_{post\text{-}proof}$ to be able to claim that the proof test screens for critical flaws. If CCS_{proof} is less than $CIFS_{post\text{-}proof}$, CCS_{proof} may be used as the initial flaw size definition in a damage tolerance assessment in place of 90/95 POD NDE. Proof test logic as described here is summarized in Table 4, Proof Test Logic Summary.

Table —Proof Test Logic Summary

CIFS	Critical Initial Flaw Size: the initial flaw size present in a part at the beginning of service life that would grow to the critical crack size (CCS) after exactly four service lives, calculate using LEFM crack growth analysis
CCS	Critical Crack Size: the crack size that corresponds to the onset of a catastrophic failure of the part via either crack growth instability or net section strength failure, calculate using LEFM
$CIFS_{post-proof}$	The CIFS based only on loading that occurs after proof testing
CCS_{proof}	The CCS based only on proof test loading; this is the smallest crack size that would fail during the proof test
if	$CCS_{proof} < CIFS_{post-proof}$ and LEFM is shown to be valid
then	Proof testing is capable of screening for critical flaws, and CCS_{proof} can be defined as the initial flaw size for DTA

No matter how the results of the proof test flaw screening are used, the method requires a special state-of-the-art fracture analysis, usually a special CCS analysis, to calculate critical flaw size due to application of proof load case only. It is called “special” because, in contrast to the common fracture mechanics type analyses where the lower bound assumptions are applied, the upper bound assumptions on toughness and crack growth behavior need to be used for this type of analysis.

Note that some conditions/assumptions used in a typical DTA, such as tensile residual stresses, knocked down material toughness values, and elevated applied loads/stresses are non-conservative when applied to proof test-based demonstrations of structural reliability. When using a proof test for screening, it is recommended that the K_C (plane stress fracture toughness) values corresponding to the part thickness or lower are used instead of K_{IC} (plain strain fracture toughness) and that the tensile residual stresses are ignored unless they are reliably calculated. The same applies to the applied stresses, where the lower bound values are recommended to be used unless the exact values of applied stresses/loads are known. Additionally, in the fracture analysis, it is recommended that the net section yielding failure criterion is ignored and only fracture mechanics-based criteria are used for flaw size calculations. Note that a CIFS analysis involves use of inconsistent material property assumptions (i.e., lower vs. upper bound). This is necessary to get a conservative prediction for CIFS.

Another point that could affect the credibility of the proof screening method is the inaccuracy in the K solutions. A K solution might be conservative for normal life analysis (where the objective is to demonstrate something is unlikely to fail); this is not necessarily the case if it is used to predict that something is likely to fail. Therefore, the conservatism may not be consistent over the full range of parameters. In many cases, this may tend to cancel out between crack growth and proof test analysis but may not always be the case.

In contrast to the cases where the proof test is used only as part of the acceptance testing and the requirements call for specific proof test factors (see Table 1), no recommendation can be made for the proof factor when the proof test is used for flaw screening. The proof test factors in this case should be determined by analysis and should be high enough to be able to reliably screen the flaws that might be present in the part, though it also should be low enough so that no detrimental yielding is introduced.

A non-pressurized hardware proof test that fully screens part geometry for critical flaws is difficult compared to pressure proof testing. For pressurized hardware, the design operational load may be easily replicated in a lab setting and tends to fully stress the entire part as it is stressed in operation. Unpressurized parts, especially if the geometry is complex, may have strength margins driven by random vibration loading, shock, or other non-uniform loading. While simplified loading may be possible in a test fixture to envelop operational mechanical loading, this is more challenging and can drive an inefficient design if certain areas are “over-proofed,” i.e., designed simply to survive the proof test so that other areas can reach proof stresses that can screen flaws.

Certain hardware types that are mechanically loaded in ways that are relatively simple such as truss members, brackets, rod ends, or inserts may be well suited for mechanical proof testing that screens flaws. Section 7.5 includes discussion on design approaches to assist a proof test such as printing temporary test fixturing onto a part and performing design optimization to ensure flaw screening at all locations. Additionally, consideration may be placed on use of zoned methods such as proofing certain locations of a part and using NDE or NFC classification in other regions.

In a flaw-screening proof test, there is no need to apply an ECF. The reason for this is that the test load is determined purely as needed to reveal critical flaws in the test laboratory, not to envelop a flight environment.

7.2.4 Probabilistic Fracture Mechanics (PFM)

PFM may be used to meet the intent of fracture control requirements for FC parts as indicated in Figure 4. Use of this approach requires consultation with and ultimate approval from the RFCB. If implemented appropriately, the PFM approach offers a fracture control strategy that is potentially significantly less conservative than a deterministic approach but is still technically rigorous and meets the intent of NASA-STD-5019.

The PFM approach is based on performing DTA where several input variables are defined according to a statistical distribution. The outcome of a PFM assessment is the total risk of a fracture failure for a part in its given service lifetime. A project may define required minimum probability levels to maintain on total risk of fracture. For example, in cases where failure of a part leads to loss of life or vehicle, a project could require “six 9s” reliability where the probability that a part will not fail due to an undetected defect has to be equal to or greater than 0.999999 (i.e., less than a 1-in-1,000,000 chance of failure). This section describes the PFM approach, how it can be implemented, and discusses common related challenges, including part failure map discretization and how to deal with unavailable or limited statistical input data. If challenges related to limited or unavailable statistical input data cannot be overcome, the PFM approach may not be appropriate.

As noted, use of a PFM approach may be significantly less conservative than using a deterministic approach. Both methods account for the worst-case flaw, but the PFM approach considers that a defect may not be present in a given location and therefore can be seen to "reward" part and process quality. Deterministic methods effectively allow the worst-case flaw to describe the failure assessment of the entire part; when in reality if all part locations are considered, there may be elevated risk of fracture failure only in one location. This section also identifies numerous opportunities for partial or hybrid implementation of PFM to still gain some benefit in cases where required statistical input data is limited or unavailable.

A recommended procedure for a PFM methodology is listed below:

1. Gather probabilistic input data.
2. Perform stress analysis of part.
3. Calculate a part failure probability map.
4. Calculate total risk of part fracture and compare to project-defined allowable.

The first step listed involves gathering probabilistic input data for a location-specific life assessment. "Location-specific" in this discussion refers to gathering data such that variations are distinguished according to local part features such as geometry, NDE capability, or material. Probabilistic input data should be data directly applicable to the part, process, material, and machine under consideration. Typically, distribution of (a) defect size/shape, (b) defect spatial frequency of occurrence, and (c) NDE probability-of-detection are the input variables defined according to a statistical distribution as PFM results are highly sensitive to these items. On the other hand, PFM results may have minimal sensitivity to other types of input data, including, but not limited to, defect type (pore, closed crack, LOF, surface vs. embedded, etc.), material properties, residual stress field, crack growth behavior, crack aspect ratio, and geometric features' associated tolerances (hole size, thickness, radius, etc.). Service loads, such as temperature or stress, may also have minimal sensitivity on PFM results; if loads are unknown, highly uncertain, or highly variable, it may be prudent to consider loads using a statistical distribution. Rationale based on experimental data, analysis, or heritage information should be provided for why or why not an input variable is defined as deterministic.

One complication that could arise in quantifying defect distribution variables is that AM parts may have many defects that are inconsequential for the intended use. Some effort may be required to define of what an inconsequential defect consists of. For example, can the formation time be measured or predicted for a cyclic load to transform a given defect type or size into a sharp crack? Can a conservative analysis be used to indicate that cracks below a certain size are below the growth threshold? Insight into these types of questions using simple fracture analysis or existing test data could help categorize some commonly occurring defect types or sizes as negligible.

In the context of fracture control, it is the rogue flaw that is of primary concern. The rogue flaw is one that occurs rarely and is not representative of the characterized nominal operation of a qualified AM process. Unfortunately, it is for this exact reason that the rogue flaw is the hardest to produce in a lab setting and difficult to characterize in a statistical definition since it cannot be observed easily. Its size may be considered in probabilistic terms or assumed equal to the NDE capability as a worst-case assumption. Generally, lack of probabilistic input data may be seen as the biggest impediment to using the PFM approach. Some options are listed below describing strategies for how to still utilize a PFM approach when there are limitations on availability of defect distribution input data and/or it is not feasible to perform a dedicated production program to generate representative samples to build a new database:

a. Define defect distribution as the inverse of the NDE probability-of-detection curve. Note that the NDE POD curve may vary by location. Additionally, this approach may be overly conservative if the NDE capability is much larger than any likely defect size.

b. Consider if in-situ manufacturing sensor data is available and if it would be useful in determining defect distributions.

c. Define an enveloping defect distribution based on an available but limited data set according to one of the following options:

(1) Take a mean and standard deviation of a limited dataset and fit a distribution curve to these parameters. The confidence level of the selected mean and standard deviation values may or may not drive a need for a penalty factor to scale the mean.
(2) Assume a confidence level and define a distribution curve of the random variable based on this assumption. From this perspective, the onus is reversed where the upper bound assumption is confirmed by test which may be a much easier task. For example, if an assumption is made that 95% of parts do not contain a rogue flaw or that 95% of defects are smaller than a certain size, testing should be done to confirm this assumption.

d. Use a hybrid deterministic PFM approach. Generally, a deterministic fracture analysis has three major assumptions: (a) a rogue flaw is present in the part, (b) the flaw is at the worst-case location, and (c) the flaw size corresponds to a known value or fixed NDE capability. If statistical data is not available to better describe all three of these assumptions in a PFM, consider keeping some as deterministic. For example, assume one rogue flaw is present in every part in the worst location, but statistically characterize defect size distribution. Other characteristics of the rogue flaw such as aspect ratio may be defined as a deterministic value. This hybrid method will likely be in between PFM and deterministic techniques in terms of overall conservatism.

e. If statistical data from “part families” are available, consider if defining distribution curves from this data set is appropriate (see section 7.2.1.2 for part family definition). For example, there may be statistical defect data available from numerous parts that are not identical but still sufficiently similar in terms of machine, process controls, applications, design, material, etc., such that it is reasonable to assume defect distribution data are applicable across the part family.

f. If testing is planned to generate statistical data for an input variable, consider truncation of the test program to minimize data collection in ranges likely to be of little concern or with a known outcome. For example, instead of populating a full distribution curve for defect size equally across a range of all sizes, focus testing on the range thought to be critical and define the distribution curve elsewhere according to insight from limited data, analysis, or other conservative assumptions.

A final note on input variables is that PFM results may be sensitive also to scatter in fatigue crack growth lifetime due to scatter in material properties. This can be accounted for by determining a probabilistic scatter factor to include in crack growth analyses. One approach is to perform fatigue crack growth tests and, for each test, record the ratio of actual-to-predicted (A/P) lifetime. A statistical distribution can be fit to the A/P data from all tests and act as the definition for lifetime scatter to include along with other variable input parameters. Take care to ensure direct applicability of the A/P ratio. If testing is dominated by the Paris Region and the application is dominated by near threshold fatigue crack growth behavior, the test A/P ratio will be far too small.

Alternatively, a global life scatter factor may be defined as a deterministic value. In this approach, the scatter factor (i.e., covering material scatter) may be defined as a number lower than the usual required four lifetimes if testing is performed to determine a value that bounds observed lifetime test data either completely or to a predefined required reliability. Note that a global scatter factor is conservative and treating it this way partially undermines the advantage of using PFM.

The second step listed in the PFM procedure is to perform a stress analysis of the part. The analysis model should include both location-specific and orientation-specific data, if needed, including varying material properties and residual stresses. The stress results are used to feed into fracture analyses at each discretization of the failure probability map as described below.

The third step listed in the PFM process is to use stress analysis results to calculate a failure probability map across a part. Note that the sum of failure probability at all locations approximately equals the failure probability for the whole part when the individual failure probabilities are small. At the most refined level, the failure map may be discretized at the element level within a FEM; as will be discussed later, zoning strategies can be used to reduce the number of locations defining the failure map.

At each discretization point in the part failure map, numerous fracture analyses are performed where in each individual analysis a different sampling combination of the random input variables is considered. This step can be performed using a Monte Carlo approach or other similar technique where random samples from each of the input variable distributions are used in a series of trial analysis runs. The result of each individual analysis is either that the part failed or it d**id** not fail. After numerous trial runs, each with different combinations of sampled input variables, it will become apparent that a certain percentage of the overall results are predicting failure. This percentage will converge before all possible permutations of input variables are considered. If enough trials are not run, sampling errors may result. The analyst achieves and documents evidence of convergence. Note that all **of** these analysis runs inherently assume that a defect is present. The final step in calculating risk of fracture for a given map discretization is calculating the product of (a) the probability of having a defect of a certain size in the zone and (b) the probability of fracture, given that the defect is present in the zone as determined from the converged results from the trial analysis runs.

Typically, discretizing a failure probability map at the element level is not necessary in PFM for accurate results. In fact, this approach may rule out the method entirely as it could necessitate excessive computational resources. Improved computational efficiency, while maintaining accuracy, may be achieved using a zone-based approach where each zone consists of one discretization in the failure probability map and may encompass many elements. There is one fracture analysis location per zone. Just as with an element level failure map discretization, a zone-based discretization maintains the notion that there could be a critical defect anywhere in the part; and the total probability of failure of the part is approximately equal to the sum of failure probability at all locations. Typically, all locations within each zone share similar stresses, defect distributions, inspection quality, etc. The fracture analysis calculations are performed by placing the defect at the worst-case location and orientation in each zone. As the number of zones increases (and the size of the zones decreases), the total risk of failure for the part will converge. Reference FAA Advisory Circular 33.14-1, Damage Tolerance for High Energy Turbine Engine Rotors; Enright, et al. (2016); Moody, et al. (2013); and Gorelik (2017) for examples of zoning strategies[6].

Following the zoning discussion, one may begin to recognize more clearly how using a PFM approach compared to a deterministic method may be helpful if part optimization is desired. The deterministic method essentially penalizes the entire part by assuming that the worst-case location for a defect is distributed everywhere. The PFM approach still captures the worst-case location but weighs its effect on the overall failure prediction by considering the probability that the rogue flaw, should it occur, happens to fall in that location.

The part failure probability map is generated by plotting the probability of failure at each map discretization point (i.e., at each zone) for the required service life and scatter factor. In truth, an actual graphical map is not necessary; it is a useful way to visualize, track, and report a PFM assessment. The final step in the PFM approach is simply to show that the probability of failure for the entire part is less than the project-required value. Total risk of part failure is approximated by summing failure probabilities across all zones.

One publicly available example of applying PFM to AM hardware is provided by Mardaras, et al. (2020).

7.3 Fracture Mechanics Material Properties

The following guidelines should be considered applicable when obtaining fracture mechanics material properties for use in fracture control implementation on an AM part. Additional detail

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[6] Note the difference in concept between zoning in a PFM analysis and zoning of zone-based approach as described in section 7.1.5. The former is related to a numerical solution strategy and the latter is related to a description of properties, stresses, and hazards.

and requirements related to obtaining fracture mechanics properties can be found in NASA-STD-6030:

- Fracture properties are representative of the material process condition found in the hardware.
- All data corresponds to the expected in-service temperature and chemical environments.
- Fatigue crack growth rate (FCGR) data and fracture toughness is obtained based on ASTM E647, Standard Test Method for Measurement of Fatigue Crack Growth Rates, and ASTM E399 or ASTM E1820 for various material orientations to capture the fracture anisotropy of the material. At a minimum, for each property type, three characteristic orientations of the manufacturing process, as well as three 45° angles with respect to the characteristic directions are needed. If there is rationale that properties from a single 45° angle orientation would be representative of the other two 45° orientations, the RFCB may approve a reduced test scope where only that single orientation is included. A minimum of three repeats are needed for each orientation.
- FCGR data envelops the service life R-ratios.
- The FCGR and fracture toughness values for predicting crack growth and instability are upper bound and lower bound values of the tested specimens, respectively. The rationale for using upper bound FCGR data is that AM parts are often fatigue-sensitive (due to rough as-built surface finish) and additional conservatism in this area is justified. If there is test data available indicating that data scatter is limited and there are no additional risks or uncertainties driving a need for a more conservative fatigue crack growth analysis, average values may be used. Rationale for use of average values in the fatigue crack growth analysis should be presented to the RFCB.
- Rationale is necessary to justify similitude in residual stresses between test coupons and the flight part at all locations fracture analysis will be performed. One example of rationale in this case may be similar post-processing thermal treatments.
- NASGRO® material parameters are developed using test results as described in the NASGRO® manual. It should be noted that NASGRO® parameters are developed for defining typical properties, and extra caution is required for developing minimum capability material properties.
- If crack growth behavior is anisotropic, it may be acceptable to determine the enveloping orientation and then assume the corresponding toughness and crack growth properties in all directions in a fracture analysis as if the material were isotropic. Enveloping in this case refers to the orientation with minimum toughness and fatigue crack growth resistance.

An approach may be developed where an existing “non-AM” material in NASGRO**®** is desired for use in an AM part fracture analysis based on the rationale that the existing material is similar and the properties envelop those of the AM part. In this type of approach, material and fracture testing is still required to verify the envelop of all properties; the total number of test coupons may be reduced compared to a case where no preexisting material definition exists. Use of this approach or similar should be reviewed and approved by the RFCB.

7.4 High Temperature Considerations

Some AM space flight components may operate at temperatures that are high enough such that creep and/or creep-fatigue crack growth and instability may occur. This fracture control document does not explicitly address creep damage. The following steps should be followed when creep crack growth is a possible damage mechanism:

1. Determine if creep damage can occur during the component lifetime. If creep is deemed a possibility, then follow steps 2-7.
2. Determine the creep rupture life of the remaining ligament of the initial cracked component (t_R). If t_R is less than the component design lifetime, then the component fails the test.
3. Determine the instability loads of the initial cracked component. If the instability load of the component (with NASA safety factors applied) is greater than design loads, the component fails the test.
4. Perform a creep and creep-fatigue crack growth analysis of the cracked component for the time period of interest. This can be done using accepted creep and creep-fatigue crack growth procedures based on C*-integral assessment procedures as defined in the current high temperature codes such as ASME[7]; API-579[8], Fitness-for-Service Assessments; or R5[9]. Most current engineering methods for high temperature crack growth are based on simple and conservative reference stress approaches, although finite element methods are permitted. Note that creep and fatigue interact so this effect must be included. Determine the amount of additional crack growth along the entire crack or at the deepest and surface points of the crack (Da_{CF}, Dc_{CF}).
5. Add crack growth from (4) to the initial crack sizes to find the final crack size for the time period of interest. The current crack size is then a_f and c_f.
6. Recheck the creep rupture life of (2) to ensure that creep rupture does not occur for the end of time period crack sizes in (5).
7. Ensure that crack instability (criterion 3) does not occur for the final crack size, including safety factors as applicable. The material crack instability resistance curve should include reductions caused by creep damage if applicable.

The effects of residual stress in the AM part should be included in this assessment. The procedures to be used for the creep-fatigue crack growth assessment can be chosen by the designer or spacecraft owner. The procedures and material data used for the crack growth and failure assessment are proven to the NASA Fracture Control Board. This includes justification of the crack growth modeling procedure of (4) and all material parameters necessary for (2) through (6), including the creep crack growth laws used.

7.5 Design for Additive Manufacturing (AM) Fracture Control

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[7] Code described in American Society of Mechanical Engineers technical basis paper (reference Brust, et al. [2022]).
[8] Reference API-579 [2008]
[9] Reference British Energy Generation LTD (BEGL) [2008]

An active area in AM technology development is known as “Design for AM.” This methodology takes the position that AM should be utilized beyond simply producing existing parts cheaper but actually enabling new product forms that are more efficient and/or functional that otherwise could not exist using legacy manufacturing methods. A subset of “Design for AM” may be “Design for AM fracture control.” The following discussion on “Design for AM fracture control” is high level and serves only to introduce the concept.

One goal in “Design for AM fracture control” may be designing parts to be NFC. Failsafe is one NFC designation used in fracture control to identify parts that, if failed, do not result in a catastrophic hazard. While a traditionally manufactured part may have a single load-carrying member, an AM part may be created to have multiple smaller members with intricate geometry where if any one failed, the overall part would still sustain the design load.

The FC: alternate approach described in section 7.1.4 is intended for use in lightly loaded parts that may have a catastrophic consequence of failure but are highly unlikely to fail due to meeting strict criteria. This includes, among other things, stresses less than 30% ultimate strength and fatigue analysis demonstrating four service lives with a factor of 1.5 on cyclic stress. One of the strengths of AM hardware is that part geometry can be highly optimized for meeting strength allowable-based design criteria. Strength-based optimization strategies could be easily modified to target the FC: alternative approach strength and fatigue thresholds specifically.

“Design for AM fracture control” may also include efforts to design parts able to be proof tested. As discussed, proof testing is most effective on hardware types that can be subjected to a simple load in a lab test apparatus that envelops flight loading at all locations in the part. For non-pressurized parts subjected to complex mechanical loading and with complex geometry, a fully enveloping proof test can be difficult. AM techniques may offer opportunities to alleviate this challenge. At a conceptual level, some options could include building in load application points, “lever arms,” that could facilitate applying loads or test fixture attachment points. The features could then be severed after the proof test is complete. Another option may be simulation of a proof test, mapping proof test coverage (i.e., TDF), and then optimizing part design such that flight loads can be sustained, proof testing achieves full coverage on a part, and the proof test screens for the CIFS.

Finally, another measure that may be taken is intentionally driving the CIFS to be larger at locations of concern by increasing structural robustness (i.e., increasing strength MS). Within “Design for AM fracture control,” this approach is based on the concept of treating NDE capability as part of the design constraints. The CIFS is a function of material properties, local geometry, and stress. Generally, the more robust a structure is, the larger the CIFS is making it easier to find using NDE. This approach may be optimized by only locally driving up the CIFS at critical stress locations or at locations where NDE is not effective. A mapping strategy may be useful to identify initially non-compliant locations to target geometry changes to drive up the CIFS size such that it falls within NDE capabilities.

APPENDIX A: EXAMPLE PROBLEMS

A.1 Example 1: Fracture critical (FC): Alternative approach bracket (Class A)

This example outlines the application of this Handbook to the qualification of a Class A Al 6061 RAM2 harness bracket produced using laser powder bed fusion (LPBF).

A.1.1 Part Overview

The bracket, as depicted in Figure 5, Example of a Secondary Structure used to Restrain a Cable Bundle on the Outside of a Human-rated Spacecraft, is a secondary structure that is used to restrain a cable bundle on the outside of a human-rated spacecraft. There are 3x bolt clearance holes on the bottom of the bracket to mount it to the spacecraft structure and 4x tie down slots on the top of the bracket to secure the cables. Spacecraft launch is the primary load case that drives the design.

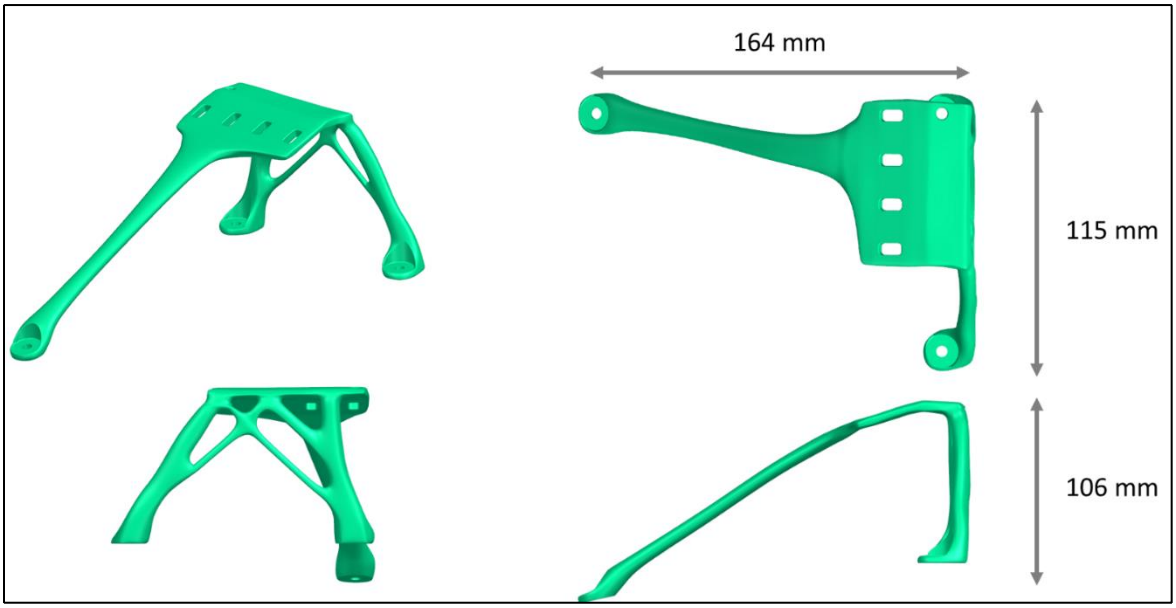

Figure 5—Example of a Secondary Structure used to Restrain a Cable Bundle on the Outside of a Human-rated Spacecraft

A.1.2 Additive Manufacturing (AM) Classification

Failure of the bracket would result in a catastrophic hazard; therefore, the harness bracket is a *Class A* part as defined per NASA-STD-6030. Due to the low structural demand and AM risk, the secondary classification is A4.

A.1.3 Manufacturing and Processing

Figure 6 shows Co-printed Flight, Flight Spare, and First Article, along with Witness Coupons Required by NASA-STD-6030. To help prevent part warpage during post-processing (primarily heat treatment), crossbeams between the mounting feet are incorporated into the build. Once printed, the parts are hot isostatically pressed (HIPed) and heat treated. To improve surface roughness, they are then chemically etched. The final step in the manufacturing process is machining of the bolt holes and spacecraft mating surfaces. The full post-processing flow is shown in Figure 7, Printed, Post-processed, Inspected, and Proof Loaded Part Prior to Delivery for Service.

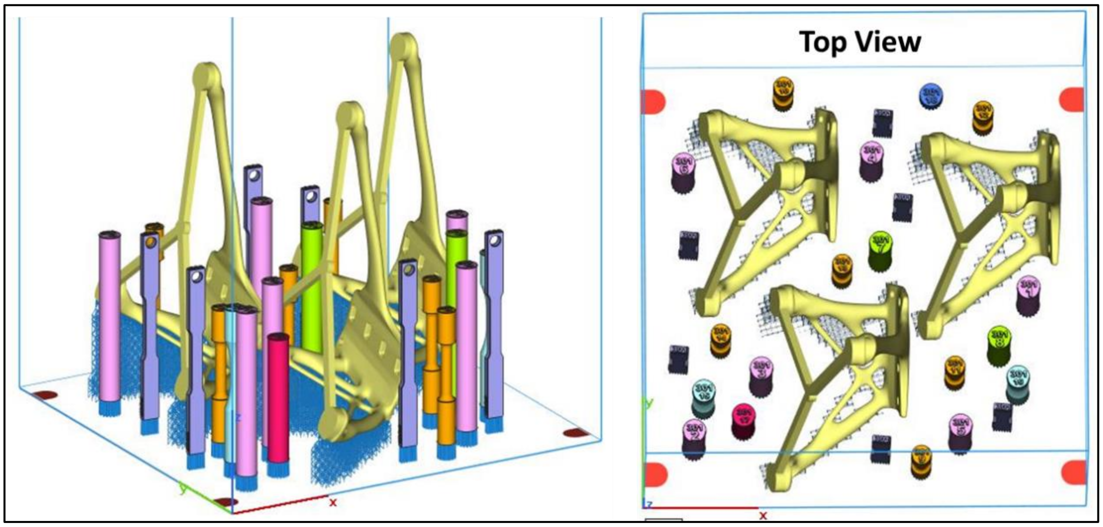

Figure 6—Co-printed Flight, Flight Spare, and First Article, along with Witness Coupons Required by NASA-STD-6030

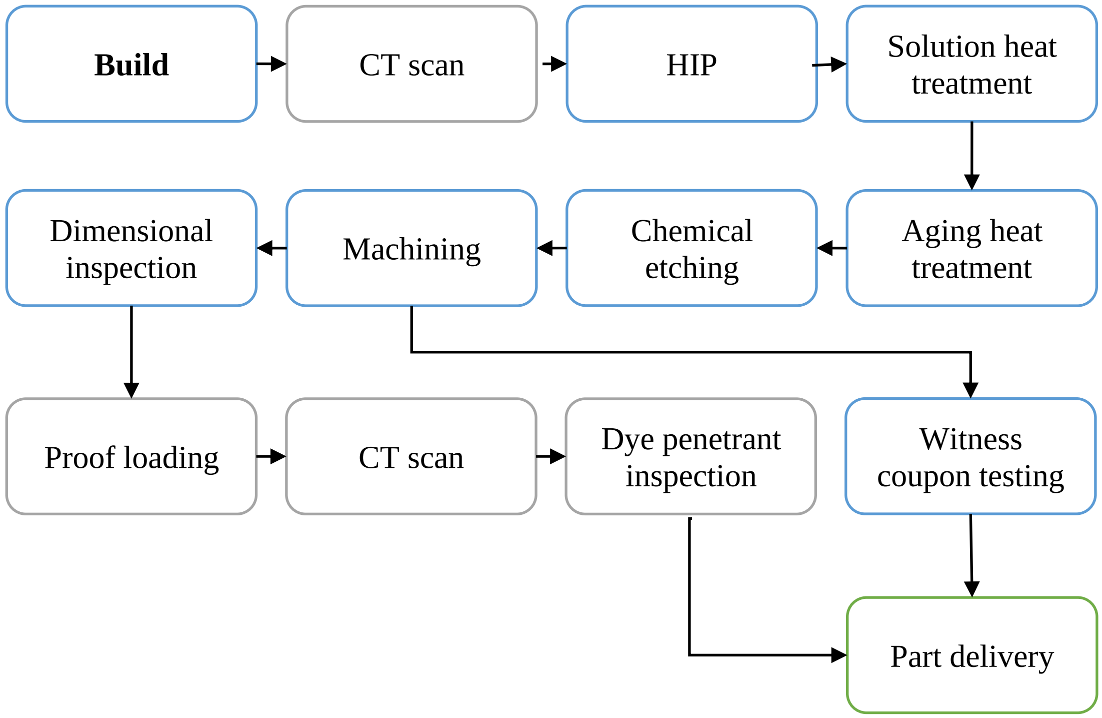

Figure 7—Printed, Post-processed, Inspected, and Proof Loaded Part Prior to Delivery for Service

All witness coupons rode along during the past post-processing steps and were tested to verify workmanship and process control. Per NASA-STD-6030, Table 5, the following coupons were manufactured:

- 6x Tensile Test Machined (ASTM E8/E8M, Standard Test Methods for Tension Testing of Metallic Materials)
- 2x HCF Fatigue Machined (ASTM E466, Standard Practice for Conducting Force Controlled Constant Amplitude Axial Fatigue Tests of Metallic Materials)
- 6x Dogbone Tensile REM (ASTM E8/E8M)
- 2x HCF Fatigue REM (ASTM E466)
- 1x Full height contingency
- 1x Metallography
- 1x Chemistry
–

All coupons were tested and found to be consistent with the Material Property Suite (MPS) for the Al 6061 RAM2 materials.

A.1.4 Structural Qualification

The governing load case for the harness bracket is launch. Per the environmental requirements document, the maximum quasi-static launch acceleration for this bracket is 100 g and defined to be in the direction that results in the lowest strength MS. Since all Class A parts are deemed FC, the structural analysis approach according to Figure 1 of this Handbook evaluates the part strength and fracture behavior (noting that the facture analysis additionally meets the fatigue requirements).

The Class A QMP (QMP-A) and MPS for the Al 6061 RAM2-based LPBF AM processes is developed per NASA-STD-6030. The material is assumed to be isotropic with an elastic modulus of 68 GPa and a Poisson's ratio of 0.33. However, since it is well documented that AM parts have anisotropic failure characteristics, the lowest yield strength (F_{ty}) and ultimate strength (F_{tu}) where conservatively selected and applied in all directions. For this material system, the A-basis design values are F_{ty} = 239 MPa (34.6 ksi) and F_{tu} = 281 MPa (40.7 ksi).

Section 5.2.3 of this Handbook calls for mitigation of residual stresses. The part is considered to be free of appreciable residual stresses due to the post-processing methodology. Residual stresses in the part are initially reduced through HIP. Further reduction in the residual stresses is achieved by the solutionization and aging process, where the alloy elements are placed into a solid solution at an elevated temperature and aged to form the dispersed precipitates that give the alloy its strength. Additionally, the removal of surface material via chemical etching removes surface residual stresses.

Due to the complexity of the geometry, the structure was modeled with 3D solid elements as show in Figure 8, Structure Modeled with 3D Solid Elements and Analyzed against Five Possible Loading Configurations. To include loading contributions from the harness, the total supported mass was lumped together and located at the center of gravity for the harness. This concentrated mass was then connected to the harness bracket in five possible mounting arrangements using RBE3 elements (notes Case 1 through Case 5), with each configuration being evaluated separately.

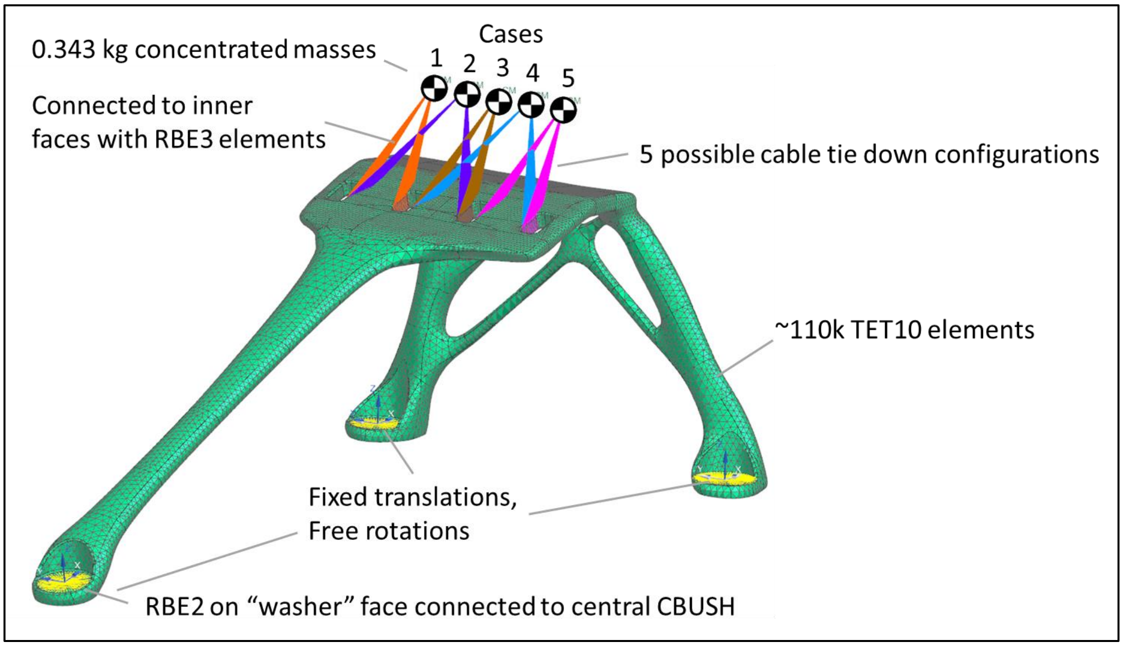

Figure 8—Structure Modeled with 3D Solid Elements and Analyzed against Five Possible Loading Configurations

To evaluate the worst-case stresses and corresponding strength margins, the 100 g's load is applied to the structure in three orthogonal directions, and the results are post-processed using an eigenvalue analysis to identify both the worst-case stress (eigenvalue) and corresponding load direction (eigenvector). Due to the RBE connections, there are artificially high stresses in the regions where cables will be attached to the bracket with zip ties. These artificial stresses are excluded from the MS calculations. In calculating MS, a Yield FoS of 1.5 and Ultimate FoS of 1.82 are used, which include the recommended additional AM factors per Table 11 in this Handbook. The MS for the evaluated cases is $MS_{ult} = 0.89$ and corresponds to ultimate failure (see Figure 9, Stress-based Margins of Safety for the 100 g's Load Case Evaluated against the Enveloping Worst-case von Mises Stress Field for the 5x Loading Configurations, the Minimum of which is $MS_{ult} = 0.89$).

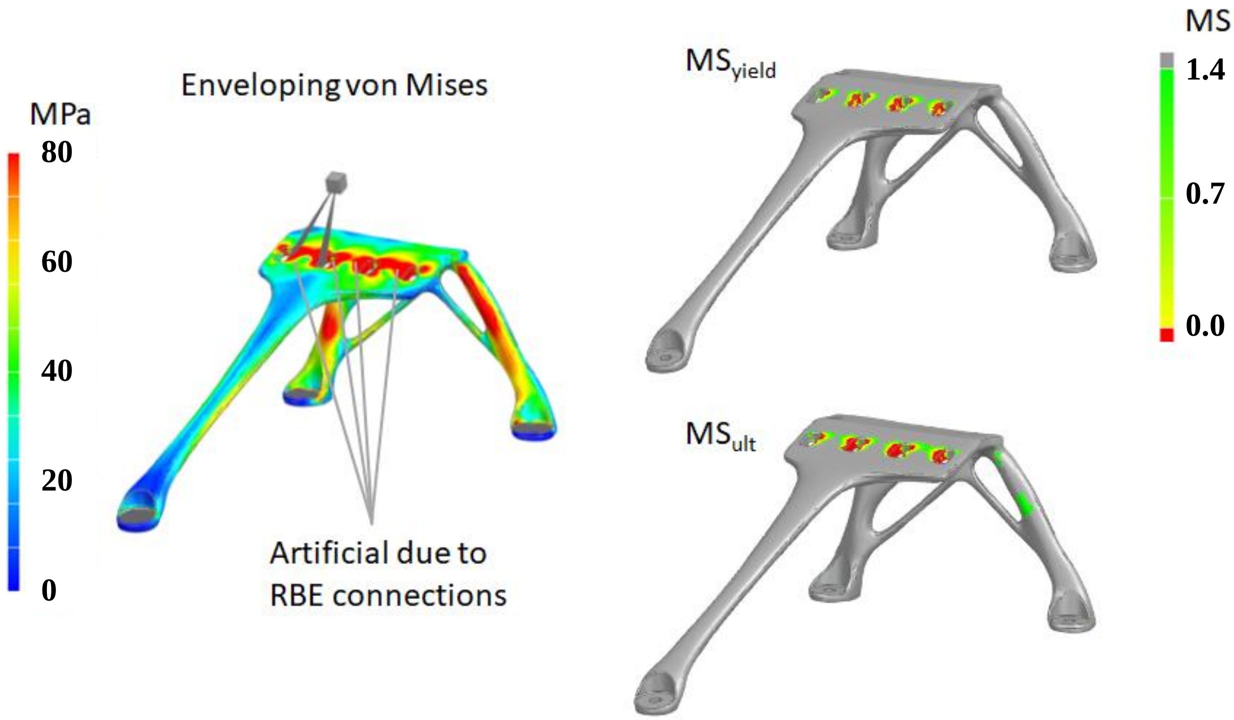

Figure 9—Stress-based Margins of Safety for the 100 g's Load Case Evaluated against the Enveloping Worst-case von Mises Stress Field for the 5x Loading Configurations, the Minimum of which is MS_{ult} = 0.89

A.1.5 Fracture Control

Because failure of the part results in a catastrophic hazard, the part is designated FC by default and requires additional qualification. This Handbook describes in section 7.1.4 an alternate approach option for FC parts that will be utilized. Assessment of the FC: alternative approach criteria for the bracket is shown below:

1. The manufacturing process has a full QMP-A and MPS, indicating a well-characterized and reliable process.
2. This is not a pressurized component.
3. The part is proof tested in the worst-case loading direction.
4. The AM risk score is 3 (which relates to removal of the as-build surface), which is less than 5 as required.
5. $\sigma_{max}/F_{ty} = 0.34 < 0.5$ and $\sigma_{max}/F_{tu} = 0.29 < 0.3$.
6. $K_{IC}/F_{ty} = 0.7584\ in^{1/2} > 0.33\ in^{1/2}$.
7. Durability analysis is completed as defined below.
8. Due to the size of the part, NDE of the full part is available for both volumetric (CT) and surface (dye penetrant) inspection and are completed as a process quality screening.

The modified part damage tolerance is assessed by analysis with NASGRO**®** per the requirements defined for a "FC: alternative approach" part: initial crack of 0.127 mm with a minimum of four complete service lives with a factor of 1.5 on alternating stress. The 100 g quasi-static load is assumed to correspond to the 3σ stress case during launch-based random vibrations and is scaled according for the varying load levels applied during ground testing and the natural frequency of the part, along with the loading environment durations, are used to estimate cycle. Due to the bending dominated loading, in which the max principal stresses are on the surface of the part, a surface crack on a cylindrical surface (SC07) is used. The fracture toughness (K_{1c}) and da/dN are measured (ASTM E1820, E399, ASTM E647 coupons) for the Al 6061 RAM2 system, fit per the NASGRO**®** manual (section 6), and imported into NASGRO**®** as a new material. Additionally, the knockdown factors as specified in this Handbook, section 7.2.1.1, are applied to the fracture properties to add additional conservatism to the analysis: da/dN increased by 25%, K_{1e}, K_{1c}, and ΔK_1 reduced by 20%, and B_k set to zero. For both crack types, no crack growth is predicted for the worst-case loading conditions (see Table 6, Damage Tolerance Analysis for Both Surface and Internal Cracks), meeting the damage tolerance and fatigue analysis requirements.

Table 6—Damage Tolerance Analysis for Both Surface and Internal Cracks (Indicates no crack growth for the worst-case loading environment, meeting the damage tolerance requirements)

Crack geometry	**Applied service lives**	**Initial crack size, a (in)**	**Final crack size, a (in)**	**< Minor Diameter d=0.3386 in?**	**Max SIF (ksi-in$^{1/2}$)**	**K_{Ic} (ksi-in$^{1/2}$)**	**SIF < K_{Ic}?**
SC07	4	.005	.005	yes	2.133	27	yes

A.1.6 Proof Testing

As both a Class A and "FC: alternative approach" part, proof testing is required for qualification. Based on the strength analysis, a 1.2x proof factor is applied to the 100 g load case in the worst-case direction. During proof loading, a linear variable differential transducer (LVDT) is used to measure displacement and verify the linearity of the mechanical response (see Figure 10, Part

Proof Loaded to 1.2x the Flight Limit Load Required for Class A and “Fracture critical: alternative approach” Parts). Additionally, surface and volumetric NDE are conducted after testing to verify that no damage occurred.

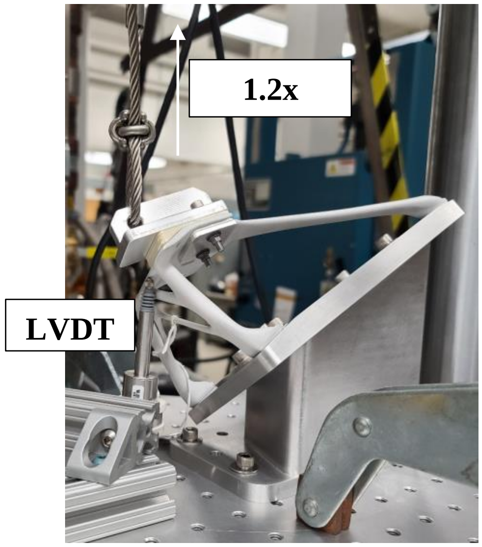

Figure 10—Part Proof Loaded to 1.2x the Flight Limit Load Required for Class A and “Fracture critical: alternative approach” Parts

A.2 Example 2: Nonfracture critical (NFC) bracket (Class C)

This example outlines the application of this Handbook to the qualification of a Class C Al 6061 RAM2 harness bracket produced using LPBF.

A.2.1 Part Overview

The bracket, as depicted in Figure 11, Bracket with Three Bolt Holes to Mount it to the Spacecraft Structure and an Array of Tie-down Holes Located on the Top Surface of the Part to Secure Cables, is a secondary structure that is used to restrain a cable bundle on the outside of a spacecraft. There are 3x bolt clearance holes on the bottom of the bracket to mount it to the spacecraft structure and 4x tie down slots on the top of the bracket to secure the cables. Spacecraft launch is the primary load case that drives the design.

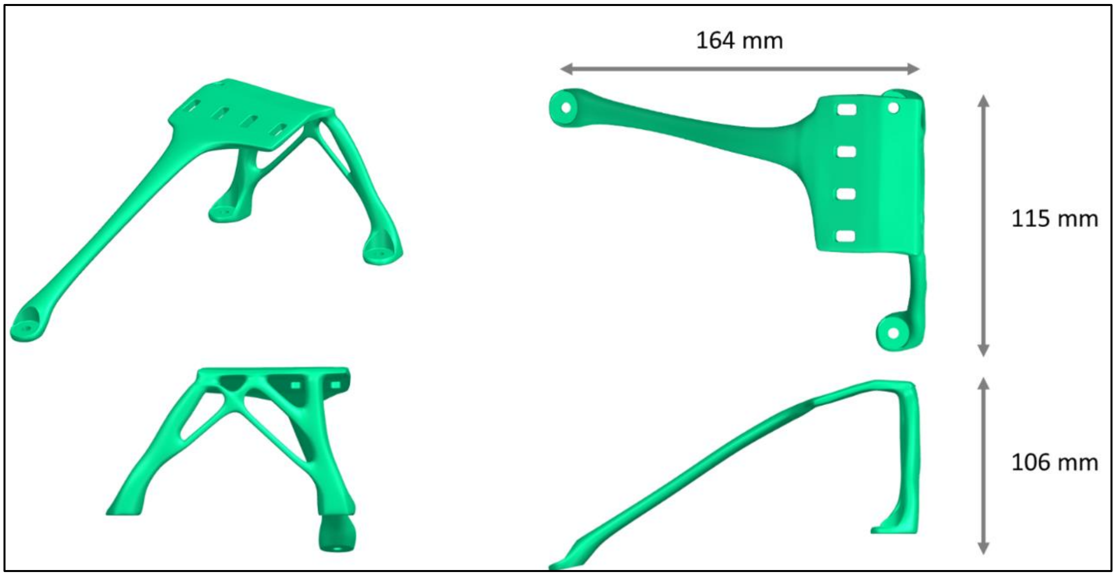

Figure 11—Bracket with Three Bolt Holes to Mount it to the Spacecraft Structure and an Array of Tie-down Holes Located on the Top Surface of the Part to Secure Cables

A.2.2 Additive Manufacturing (AM) Classification

As a harness bracket with low structural demand, low AM risk, and on a mission with negligible consequence of failure, this is designated as a Class C part as defined per NASA-STD-6030. Fracture control is not required in this case. Compliance and rationale for each requirement relevant to Class C parts in NASA-STD-6030, Table 22 (Appendix B), is documented in Table 7, Class C Part Designation Justified per the Compliance Table Defined in NASA-STD-6030, Appendix B, Table 22. Here, "failure" is defined as plastic deformation.

Table 7—Class C Part Designation Justified per the Compliance Table Defined in NASA-STD-6030, Appendix B, Table 22

NASA-STD-6030 Section 4.3.1.3 Requirements	Compliance	Rationale
Failure of part does not lead to any form of hazardous or unsafe condition.	Comply	Part is not a critical or highly loaded structure. Failure will not lead to hazardous or unsafe condition.
Failure of part does not adversely affect mission objectives.	Comply	Failure of part will have no impact on mission objectives.
Failure of part does not adversely affect other systems or operations.	Comply	Failure of part will not adversely affect other systems or operations.
Failure of part does not alter structural margins or related evaluations on other hardware.	Comply	Failure of part will not alter margins or evaluations on other hardware.

NASA-STD-6030 Section 4.3.1.3 Requirements	Compliance	Rationale
Failure of part causes only minor inconvenience operations.	Comply	Failure of part prior to launch would cause minor inconvenience of replacement. After launch, no operations will be impacted.
Failure of part does not cause debris or contamination concerns.	Comply	Part is fastened to metal structure and attached to harnessing. Release of material/debris is not a concern. There is no contamination-sensitive hardware in the vicinity and the part materials (metal) are not known contaminants.
Failed part would not require repair or replacement.	Comply	Failed part would not require repair or replacement if part fails during operation.
Part is not a protoflight article.	Comply	Part is not a protoflight test article.
Load environments are defined.	Comply	Load environments are well defined in the Environmental Requirements Document for the project.
Part is not exposed to environment with potential for material degradation over the expected service life.	Comply	The environment for this part does not pose a risk for material degradation. Part is robust to the anticipated thermal and radiation environment. Furthermore, part is not in contact with any incompatible materials.
Part is not primary structure.	Comply	Part is not primary structure.
Part does not serve as redundant structure for fail-safe criteria per NASA-STD-5019.	Comply	Part is not used as a redundant structure for fail-safe criteria. NASA-STD-5019 is not applicable to this part.
Part is not designated “Non-Hazardous Leak-Before-Burst” per NASA-STD-5019.	Comply	Part is not a pressure vessel and cannot be designated NHLBB. NASA-STD-5019 is not applicable to this part.
Part does not serve as primary or secondary containment.	Comply	Part does not provide containment.

NASA-STD-6030 Section 4.3.1.3 Requirements	Compliance	Rationale
Part is not subjected to impact loads.	Comply	Part is not subjected to impact loads.
Part has no printed threads.	Comply	Part has no printed threads.
Part is not a fastener nor does it serve the purpose of a fastener.	Comply	Part is not a fastener nor does it serve the purpose of a fastener.
If a structural analysis is required, MS ultimate tensile strength (UTS) > 6 on local maximum principal tensile stress.	N/A	Not required for a Class C part.

A.2.3 Manufacturing and Processing

The flight part, a flight spare, and first article are co-printed, along with the witness coupons required by NASA-STD-6030, as shown in Figure 12, Co-printed Flight, Flight Space, and First Article, along with Witness Coupons Required by NASA-STD-6030. To help prevent part warpage during post-processing (primarily heat treatment), crossbeams between the mounting feet are incorporated into the build. Once printed, the parts are HIPed and heat treated. To improve surface roughness, they are then chemically etched. The final step in the manufacturing process is machining of the bolt holes and spacecraft mating surfaces. The full post-processing flow is shown in Figure 13, Printed, Post-processed, and Inspected Part, Prior to Delivery for Service.

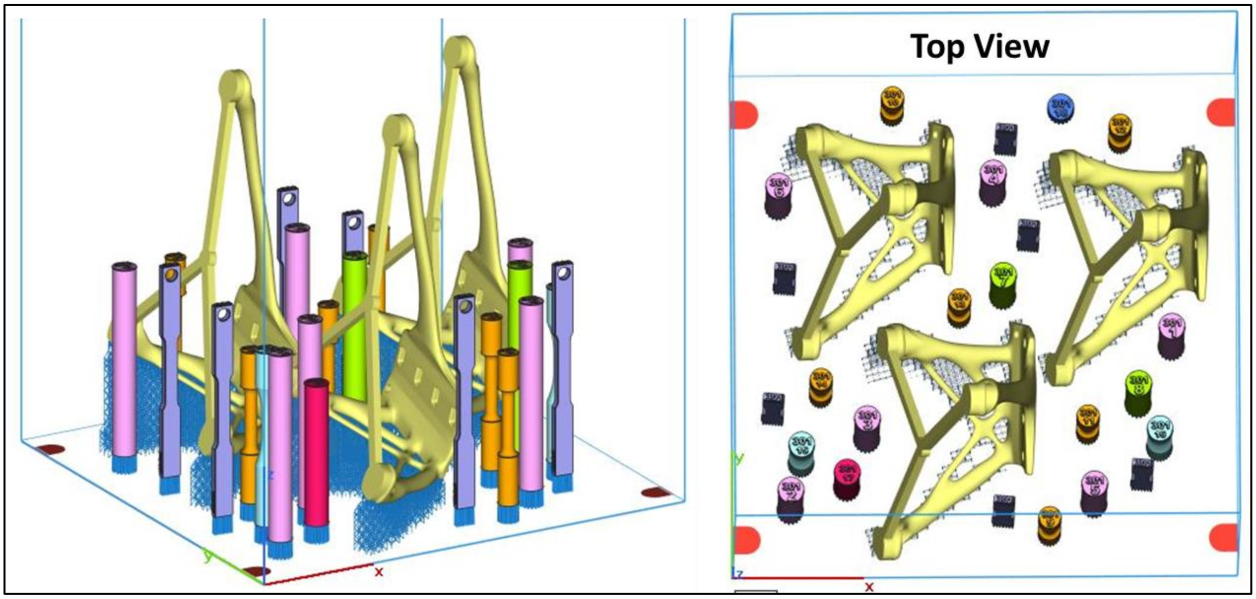

Figure 12—Co-printed Flight, Flight Space, and First Article, along with Witness Coupons Required by NASA-STD-6030

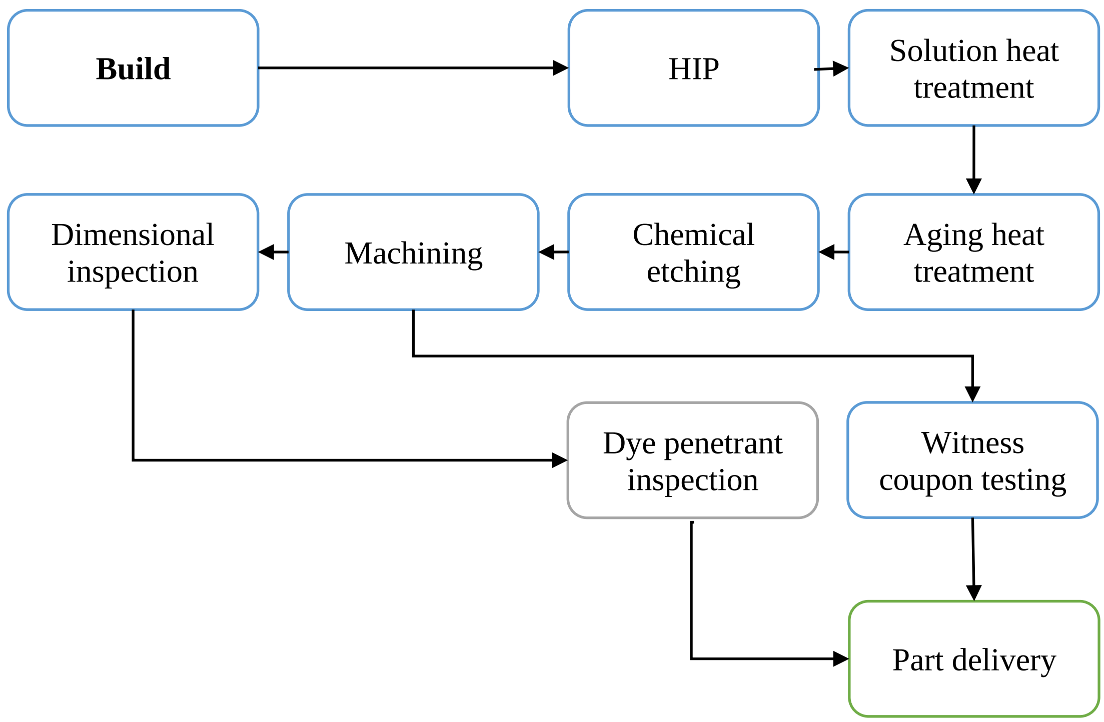

Figure 13—Printed, Post-processed, and Inspected Part, Prior to Delivery for Service

All witness coupons rode along during the past post-processing steps and tested to verify workmanship and process control. Per NASA-STD-6030, Table 5, 2x tensile test coupons (ASTM E8/E8M) were required. All coupons were tested and found to be consistent with the MPS for the Al 6061 RAM2 materials.

A.2.4 Structural Qualification

The governing load case for the harness bracket is launch. Per the environmental requirements document, the maximum quasi-static launch acceleration for this bracket is 100 g and defined to be in the direction that results in the lowest strength MS. Since this is a Class C part on a non-human-rated mission, the part is exempt from fracture control requirements. The structural analysis approach according to Figure 1 of this Handbook evaluates the part strength and fatigue behavior.

The Class C QMP (QMP-C) and MPS for the Al 6061 RAM2-based LPBF AM processes are developed per NASA-STD-6030. The material is assumed to be isotropic with an elastic modulus of 68 GPa and a Poisson's ratio of 0.33. Since it is well documented that AM parts have anisotropic failure characteristics, the lowest yield strength (F_{ty}) and ultimate strength (F_{tu}) were conservatively selected and applied in all directions. For this material system, the S-basis failure properties are F_{ty} = 239 MPa (34.6 ksi) and F_{tu} = 281 MPa (40.7 ksi). In addition to strength allowables, a series of machined and chemically etched coupons are fatigue tested at varying stress levels (under R = -1 loading conditions) and an S-N curve fit to envelop the data as shown in Figure 14, Machined and Chemically Etched Fatigue Coupons Manufactured and Tested at Varying Load Levels for R = -1 and a Conservative Enveloping S-N Curve Generated for Fatigue Analysis.

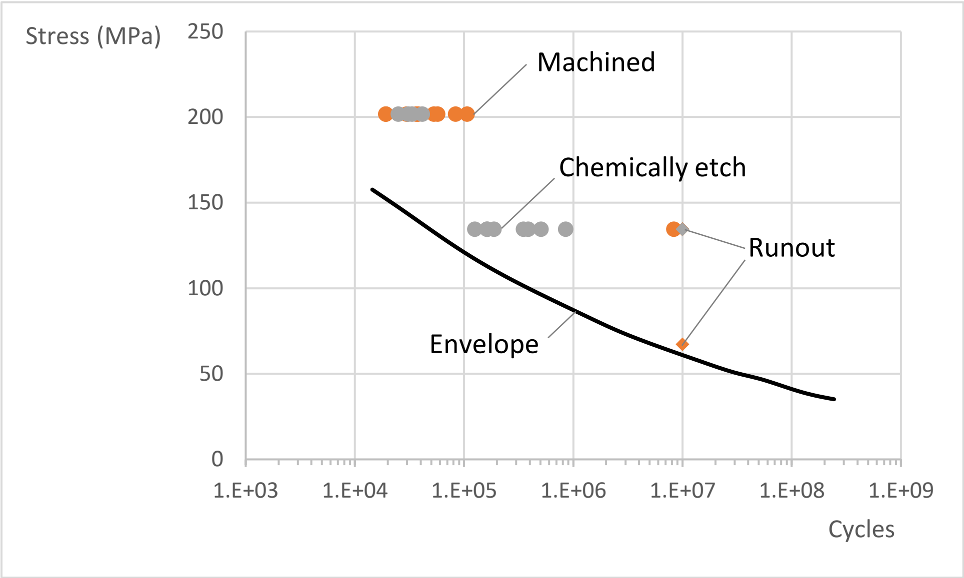

Figure 14—Machined and Chemically Etched Fatigue Coupons Manufactured and Tested at Varying Load Levels for R = -1 and a Conservative Enveloping S-N Curve Generated for Fatigue Analysis

Section 5.2.3 of this Handbook calls for mitigation of residual stresses. The part is considered to be free of appreciable residual stresses due to the post-processing methodology. Residual stresses in the part are initially reduced through HIP. Further reduction in the residual stresses is achieved by the solutionization and aging process, where the alloy elements are placed into a solid solution at an elevated temperature and aged to form the dispersed precipitates that give the alloy its strength. Additionally, the removal of surface material via chemical etching removes surface residual stresses.

Due to the complexity of the geometry, the structure was modeled with 3D solid elements as show in Figure 15, Structure Modeled with 3D Solid Elements and Analyzed against Five Possible Loading Configurations. To include loading contributions from the harness, the total supported mass was lumped together and located at the center of gravity for the harness. This concentrated mass was then connected to the harness bracket in five possible mounting arrangements using RBE3 elements (notes Case 1 through Case 5), with each configuration being evaluated separately.

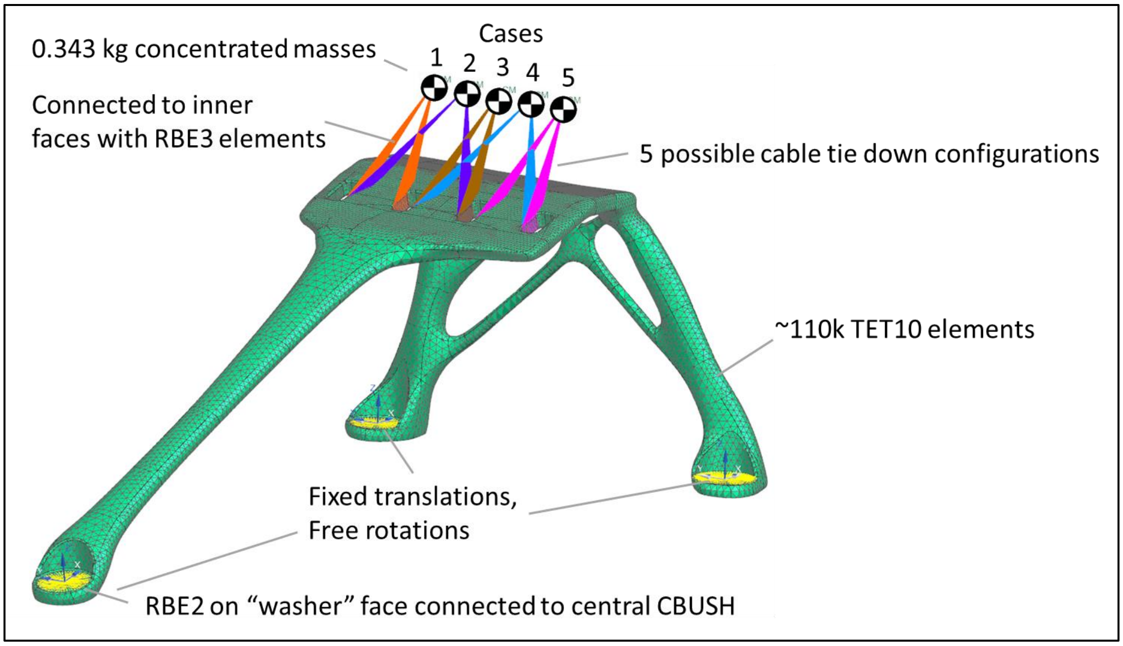

Figure 15—Structure Modeled with 3D Solid Elements and Analyzed against Five Possible Loading Configurations

To evaluate the worst-case stresses and corresponding strength margins, the 100-g load is applied to the structure in three orthogonal directions. The results are post-processed using an eigenvalue analysis to identify both the worst-case stress (eigenvalue) and corresponding load direction (eigenvector). Due to the RBE connections, there are artificially high stresses in the regions where cables will be attached to the bracket with zip ties. These artificial stresses are excluded from the MS calculations. In calculating MS, the *no-test* Yield FoS of 1.6 and Ultimate FoS of 2.0 are used. The minimum MS for the evaluated cases is $MS_{ult} = 0.72$ and corresponds to ultimate failure (see Figure 16, Stress-based Margins of Safety for 100 g Load Case Evaluated against the Enveloping Worst-case von Mises Stress Field for the 5x Loading Configurations, the Minimum of which is $MS_{ult} = 0.72$).

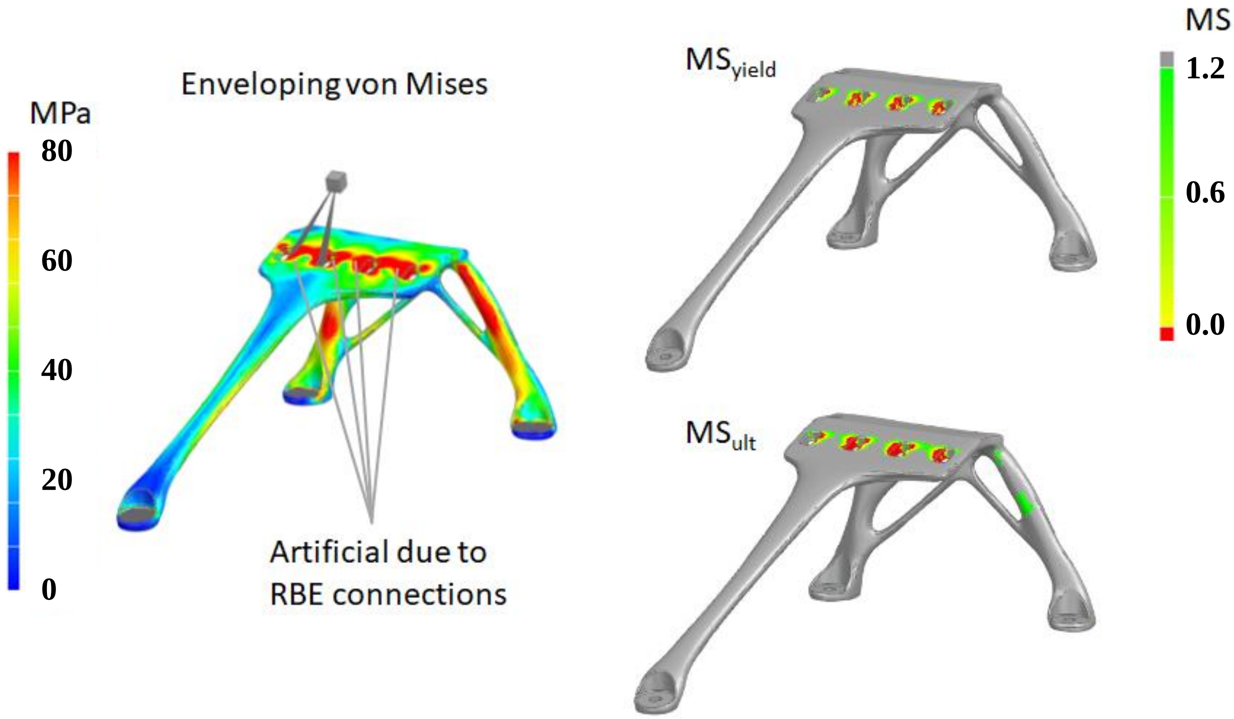

Figure 16—Stress-based Margins of Safety for 100 g Load Case Evaluated against the Enveloping Worst-case von Mises Stress Field for the 5x Loading Configurations, the Minimum of which is MS_{ult} = 0.72

Although the driving load case is defined to be quasi-static, it derives from a dynamic environment with moderate cycling and requires a fatigue analysis. Throughout the life cycle of the part, it will experience varying degrees of loading during ground testing (random vibration tests) and launch. The 100 g quasi-static load is assumed to correspond to the 3σ stress case during launch-based random vibrations and is scaled according for the varying load levels applied during ground testing. Per Table 2 of this Handbook, an additional 1.15FAF is applied to the loading. Minor’s rule is used to accumulate damage over the life cycle of the part, using the natural frequency of the part and loading environment durations to estimate cycle counts. Due to the low, elastic stresses in the part, a 4x S-Nfatigue analysis is completed per the S-N curve in Figure 14, demonstrating a MS of 1.56 (Table 8, An S-N-based Fatigue Analysis, using Minor’s Rule to Accumulate Damage, Demonstrates an MS = 1.56 against the Required 4x Life).

Table 8—An S-N-based Fatigue Analysis, using Minor's Rule to Accumulate Damage, Demonstrates an MS = 1.56 against the Required 4x Life

Max principal X stress (MPa)	Max principal Y stress (MPa)	Max principal Z stress (MPa)	Fatigue fitting factor
31.8	82.7	50.6	1.15

Event	Direction	dB	Max principal stress (MPa)	Total Cycles	Expected life	Miner's Rule Damage
RV PF/Launch	X	0	36.53	41400	1.23E+08	0.03%
RV low level		-3	25.86	13800	4.62E+08	0.00%
RV PF/Launch	Y	0	95.09	41400	7.01E+05	5.90%
RV low level		-3	67.32	13800	5.83E+06	0.24%
RV PF/Launch	Z	0	58.24	41400	1.31E+07	0.32%
RV low level		-3	41.23	13800	7.23E+07	0.02%
Acoustic	Worst	0	95.09	13800	7.01E+05	1.97%
Total				179400		8.48%
					MS	1.56

A.3 Example 3: Bracket Fatigue Analysis (Reference Witkin, et al. (2020)

A.3.1 Problem Background

AM parts subject to cyclic loading can be sensitive to fatigue due to a coarse surface finish and microscopic defects that occur near the surface. These features can act as stress concentrations that reduce fatigue life. While damage tolerance is an acceptable substitute to fatigue assessment, it is not always desirable to allow a crack to initiate in the part. The following bracket example in Figure 17, Boundary Conditions and Interactions of the AM Bracket, will be used to illustrate the application of this Handbook to verify fatigue requirements for an AM part. The bracket has an as-built surface finish, and while not preferrable, the surfaces were not machined in this application due to limitations in the manufacturing process. The part was printed in the positive 'Z' direction.

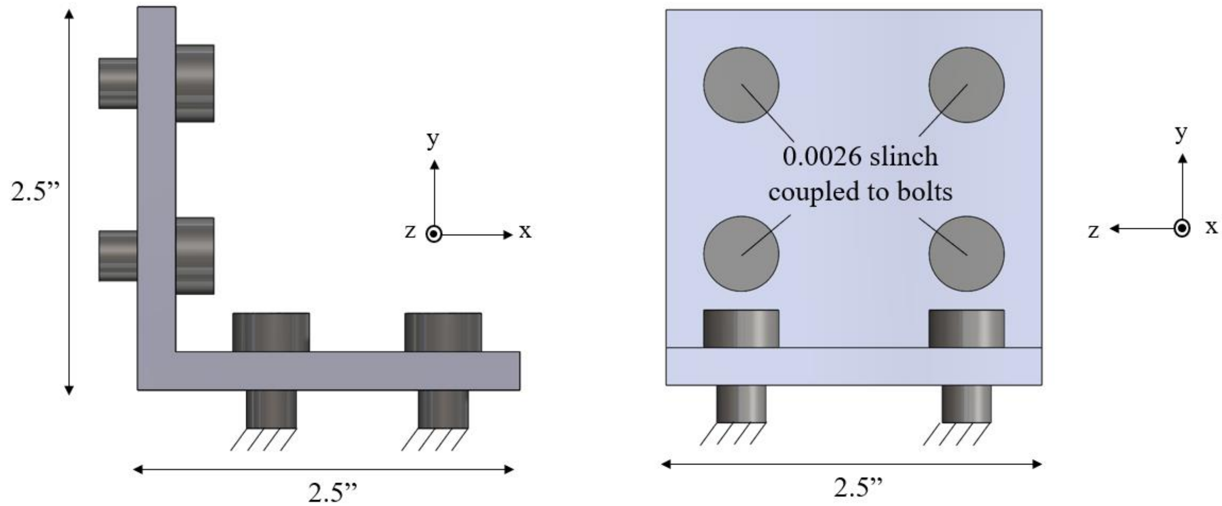

Figure 17—Boundary Conditions and Interactions of the AM Bracket

The AM bracket carries a 1-lb sensitive avionic component in the YZ plane, and it will be used in a protoflight application. The driving loads for this bracket is a random vibration excitation during a 3-minute launch, defined by the Power Spectral Density (PSD) in Figure 18, Power Spectral Density (PSD) of Base Acceleration Experienced by the Bracket during Launch. The bracket is bolted to a wall on one side (XZ plane).

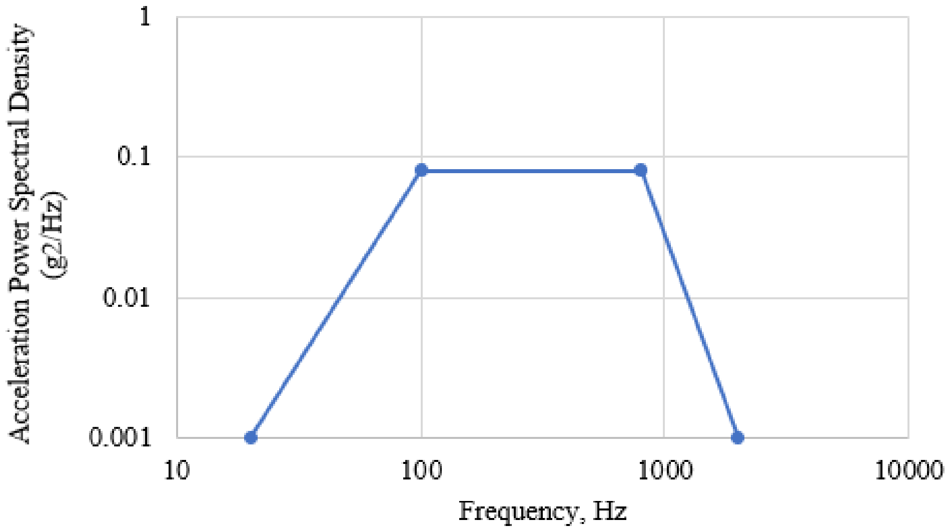

Figure 18—Power Spectral Density (PSD) of Base Acceleration Experienced by the Bracket during Launch

A.3.2 Recommended Assessments

Through a failure modes effects and analysis, it is determined that failure of this structure will result in high-consequence mission loss. The bracket is classified as a Class A part, following the guidance in Figure 2. Before stepping into the fatigue assessment, it is useful to revisit the requirements for a Class A part:

1. Strength assessment: The structural margin assessment is performed using Equations 2 and 3 with an A-Basis allowable in accordance with NASA-STD-6030, section 2.1.1; using the FoS for yield and ultimate of 1.25 and 1.4, respectively (Table 1); and using the additional AM factor for Class A parts of 1.2 and 1.3 corresponding to yield and ultimate respectively (Table 1). Recall, the additional factor is to account for risk and uncertainty in AM parts and are derived based on NASA-STD-6030. A qualification test is required for Class A AM parts, and the test factor is 1.2.
2. Workmanship assessment: A proof test factor of 1.2 is required for this Class A AM part per Table 1. Further, witness coupon fatigue testing for the Class A part is performed per NASA-STD-6030. Build orientation and surface finish of these witness coupons and microstructural features near the surface such as porosity were consistent with the manufacturing of the bracket.
3. Fatigue assessment: All AM parts receive a fatigue assessment and are shown to be capable of surviving four service lives in cases of LCF or surviving 10 service lives in cases of HCF per NASA-STD-5012 and a FAF in Table 2, which is applied to the magnitude of cyclic loading.
4. Damage tolerance assessment: Class A AM parts should follow the guidance in sections 6 and 7.

A.3.3 Fatigue Assessment

The fatigue assessment requires the stresses produced from a FEA, the load spectra based on the PSD input and launch duration, and the S-N Curve for the AM material. The fatigue assessment also accounts for the scatter fatigue factors required by the standards, and the FAFs specified in Table 2. The orthotropic material properties for the FEA are provided in Table 9, AM Bracket Material Properties. The bolts in the XZ plane are represented as fixed boundary conditions, and the mass is represented as a point mass connected to the four bolts in the YZ plane using distributed coupling. Unit errors tend to be one of the major errors in fatigue dynamic analysis. For consistency, the one-pound weight is divided by 386.09 lbs to convert this to 0.0026 slinch (or lbm), which is the consistent unit with
Table .

To determine the response to random vibration, there are two steps required in FEA: first, a modal analysis to determine the natural frequencies and mode shapes of the structure, and second, a random response analysis that uses the set of modes extracted from the previous step to characterize the response to random excitation.

The boundary conditions and interactions depicted in Figure 19, Boundary Conditions and Interactions of the AM Bracket, are applied in both the modal and random response steps.

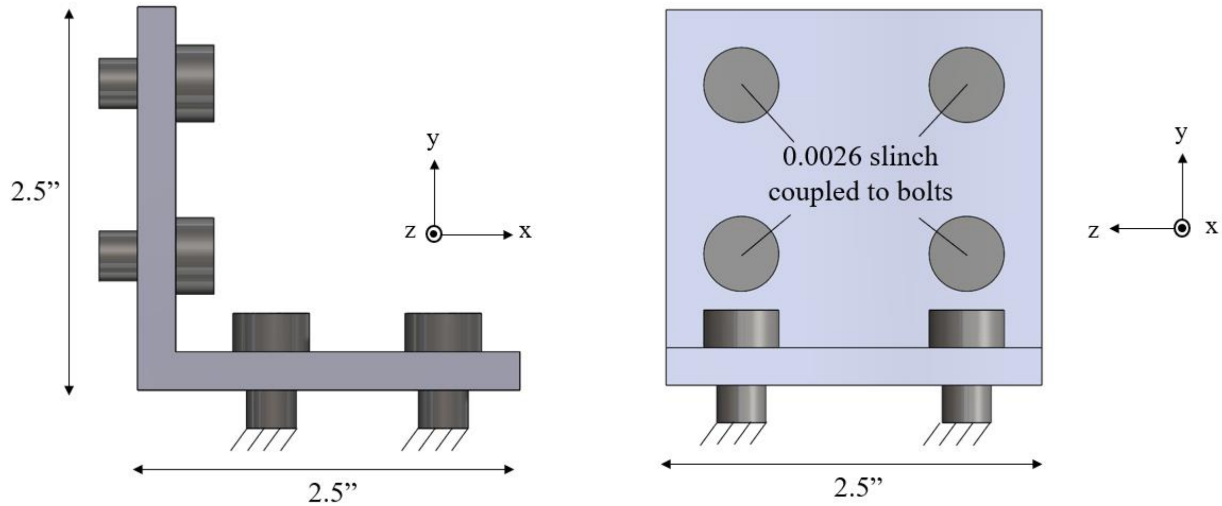

Figure 19—Boundary Conditions and Interactions of the AM Bracket

Table 9—AM Bracket Material Properties
(The properties are fabricated for the sake of example.)

Property	Value in XY Plane	Value in Z Direction
Young's Modulus, E	29,000 ksi	28,000 ksi
Shear Modulus, G_{XY}	11,000 ksi	-
Shear Modulus, G_{XZ}, G_{YZ}	-	10,000 ksi
Poisson's Ratio, ν_{XY}	0.33	-
Poisson's Ratio, ν_{XZ}, ν_{YZ}	-	0.3
Density	0.001 slinch/in^3	

The density is specified in the FEM using slinch/in^3 units to have a consistent unit of mass. Density is often provided in lb/in^3, but it can be converted from lb/in^3 to slinch/in^3 by dividing by 386.09 lbs. The density of the AM material in this example is 0.001 slinch/in^3.
The modal analysis is performed to extract modes between 20 Hz and 2,000 Hz, as this captures the frequency range provided by the loads group. The random response analysis is run over this same range.

A 0.025 direct modal damping ratio is used for the entire frequency range based on tap testing data for this bracket material.

In the random response step, an additional boundary condition is added for the random excitation. The PSD is defined and correlated to this boundary condition. The PSD includes the root mean square (RMS) vibration levels at various frequencies. It is dependent on the application and often defined by the launch vehicle provider. The PSD used in this example is shown in Figure 18. The base motion is applied in the x-direction because the bracket stresses are highest when loaded in the x-direction.

The PSD in Figure 18 represents vibration levels within one standard deviation of the average. The 1-sigma PSD is applied in the FEA. Based on the results from 1-sigma level, the stresses for the 2-sigma and 3-sigma levels can be easily determined.

The random analysis is now performed. The first step contains the results of the modal analysis. There are only three modes between 20 Hz and 2,000 Hz, as pictured in Figure 20, Mode Shapes and Natural Frequencies of the Bracket between 20 and 2,000 Hz. The first is a deflection in the x-direction at 314.2 Hz, the second is a twisting about the y-axis at 831.8 Hz, and the third is a bending about the z-axis at 1,374.5 Hz.

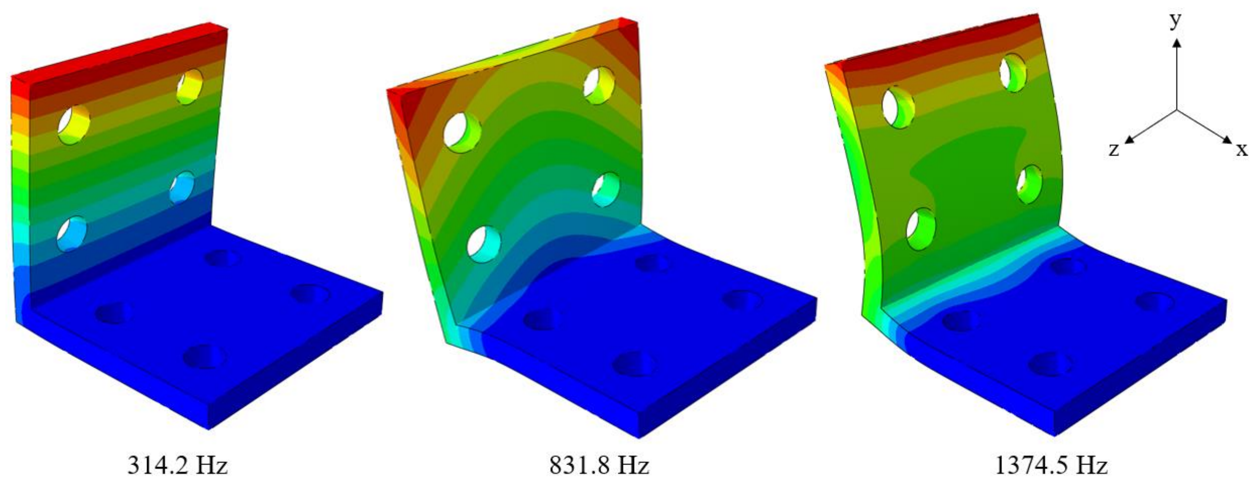

Figure 20—Mode Shapes and Natural Frequencies of the Bracket between 20 and 2,000 Hz

In the random response step, the random excitation is applied at the existing boundary condition in the four bolts in the XZ plane. The acceleration in the x-direction as a function of frequency is output at the two points shown in Figure 21, Points on the Bracket where the Response is Output. Point 1 is on the portion of the bracket attached to the point mass where the response will be measured. Point 2 is located on one of the fixed bolts. To verify that the PSD is correctly applied to the fixed bolts, the acceleration in the x-direction as a function of frequency is compared to the input in Figure 22, Input PSD Compared with the Acceleration in the x-direction as a Function of Frequency for a Point on the Boundary Condition of the Bracket. The response at the fixed bolts matches the input PSD shape, indicating that the random vibration was applied correctly.

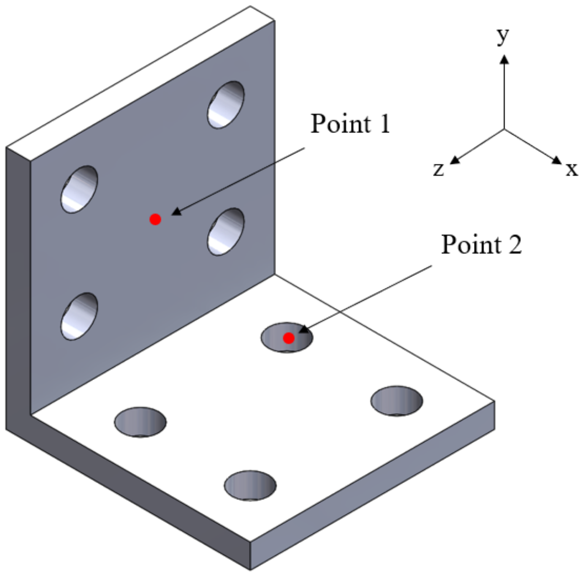

Figure 21—Points on the Bracket where the Response is Output

(Point 1 is on the portion of the bracket that holds the mass and point 2 is on one of the fixed bolts.)

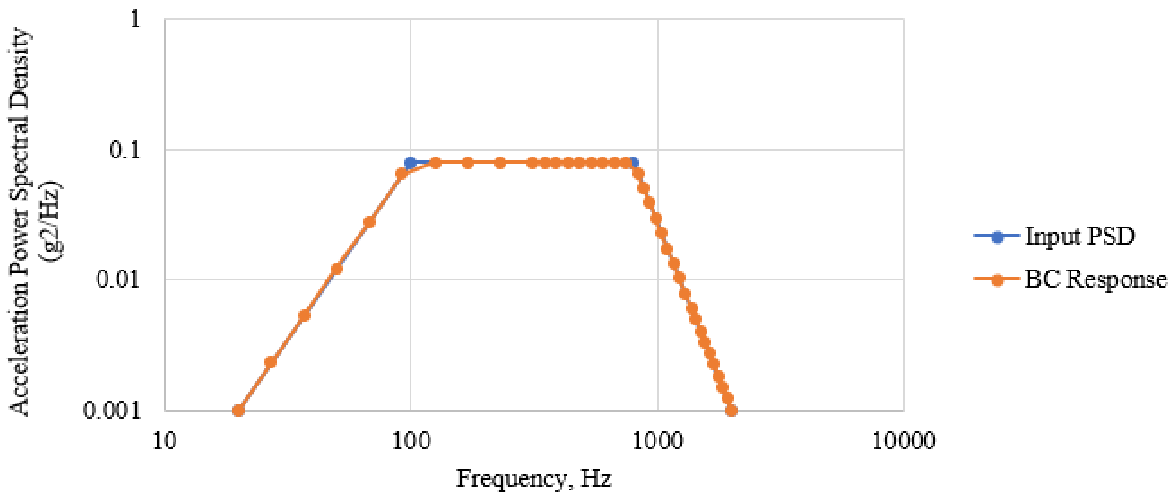

Figure 22—Input PSD Compared with the Acceleration in the x-direction as a Function of Frequency for a Point on the Boundary Condition of the Bracket

The response of the structure to vibrations can be visualized by plotting the acceleration as a function of frequency for point 2, as seen in Figure 23, Acceleration in the x-direction as a Function of Frequency for a Point Connected to the Unconstrained Bolts. As expected, the bracket is very sensitive to vibration at 314.2 Hz, which corresponds to the first mode shown in Figure 20.

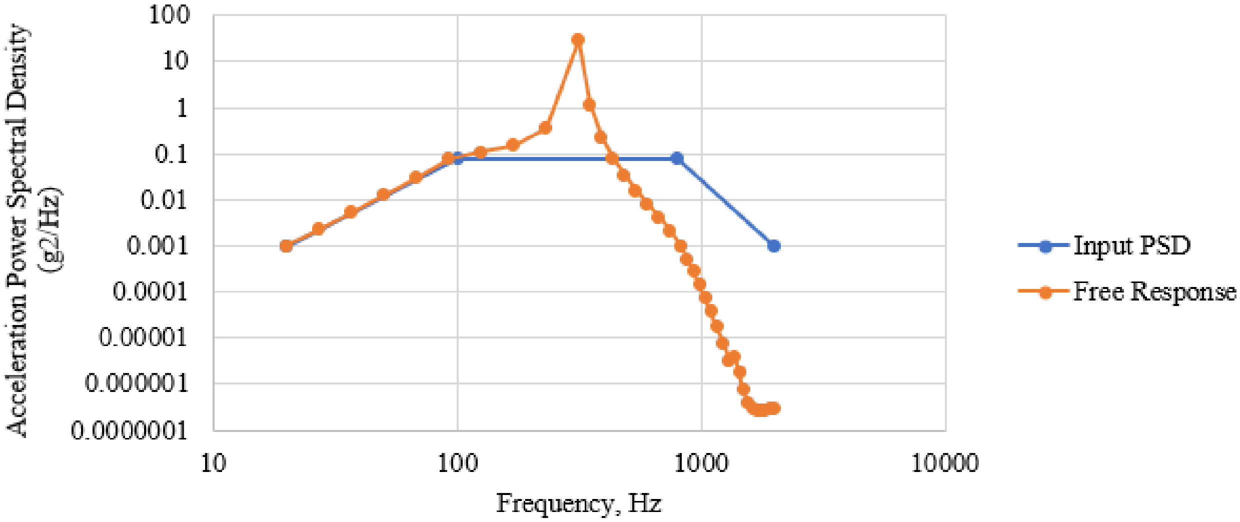

Figure 23—Acceleration in the x-direction as a Function of Frequency for a Point Connected to the Unconstrained Bolts

To complete the fatigue analysis, the RMS stress over the frequency range is calculated. The RMS von Mises stress distribution in the bracket from 20 to 2,000 Hz is shown in Figure 24, Distribution of the RMS of the von Mises Stress from 20 to 2,000 Hz. The peak stress of 27.5 ksi occurs next to the fixed bolts on the base. This RMS stress distribution represents the response to 1-sigma random vibrations. It can be linearly increased to obtain the RMS stress distribution for 2-sigma and 3-sigma random vibrations.

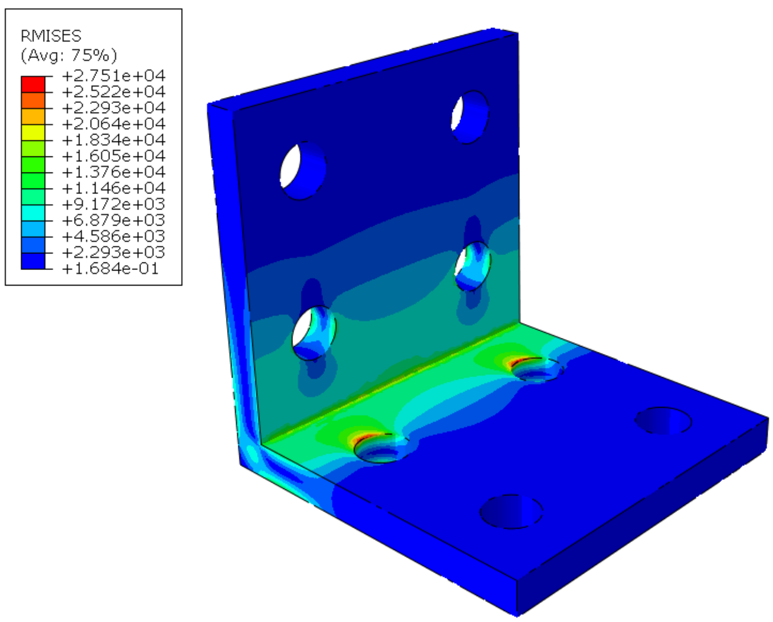

Figure 24—Distribution of the RMS of the von Mises Stress from 20 to 2,000 Hz
(The values are reported in psi.)

Vibration levels up to 3-sigma are considered in the fatigue analysis. Assuming the amplitude of vibrations follows a normal distribution, 68.27% of vibration levels are within 1-sigma, 95.45%

are within 2-sigma, and 99.73% are within 3-sigma. Based on these values, 68.27% of cycles are assumed to be at 1-sigma random vibration, 27.18% at 2-sigma, and 4.28% at 3-sigma.

To calculate the total cycles, the driving frequency of 314.2 Hz is used. This frequency is multiplied by the time of random vibrations to determine the total number of cycles. The bracket sees 56,556 cycles at 314.2 Hz for a 3-minute launch. The cycles are broken down into sigma levels in Table 10, Breakdown of Cycles Experienced by the Bracket. The maximum 1-sigma stress of 27.5 ksi is multiplied by a factor of 2 and 3 to get the peak RMS stress for 2-sigma and 3-sigma vibrations, respectively. Note that residual stresses are not included because the bracket is annealed and confirmed not to have residual stresses with X-ray diffraction.

Table 10—Breakdown of Cycles Experienced by the Bracket

Sigma Level	Cycles	Peak RMS Stress (ksi)
1	38,618	27.5
2	15,375	55.0
3	2,421	82.5

The fatigue analysis is completed in the critical locations of the bracket, which in this case occurs at the edge of the fixed bolts. The fatigue guidance of section 5.4 of this Handbook is followed. A fatigue assessment that demonstrates four service lives for LCF or 10 service lives for HCF is required. Additionally, a FAF is applied to the loads in the fatigue analysis. As per Table 2 in this Handbook, a 1.5 FAF is used for the fatigue assessment of Class A parts.

To perform the fatigue assessment, it was necessary to understand the microstructural features and surface features of the bracket. This is consistent with section 5.4.3 of this Handbook. Before stepping into the assessment, a few pathfinder brackets were printed to characterize the surface finish using various measurement techniques near the critical location of interest. Profilometer measurements reported a peak-to-valley worst-case lower-bound condition of 110 µm in this region and were further confirmed with computer tomography. Two (2)D imaging technique and scanning electron microscopy were used, which found a near-surface porosity (i.e., LOF). The surface roughness and porosity features were then replicated in hourglass fatigue tension specimens. These fatigue specimens were printed in the same orientation as the maximum principal stress orientation in the bracket and employed the same printer, process parameters,powder, and software. Christensen, R. M. (2013) provides the test configuration geometry (Figure 25, Hourglass Specimen used in the Fatigue Test Campaign to Characterize the S-N Curve) and the test procedure for the load-controlled fatigue tests.

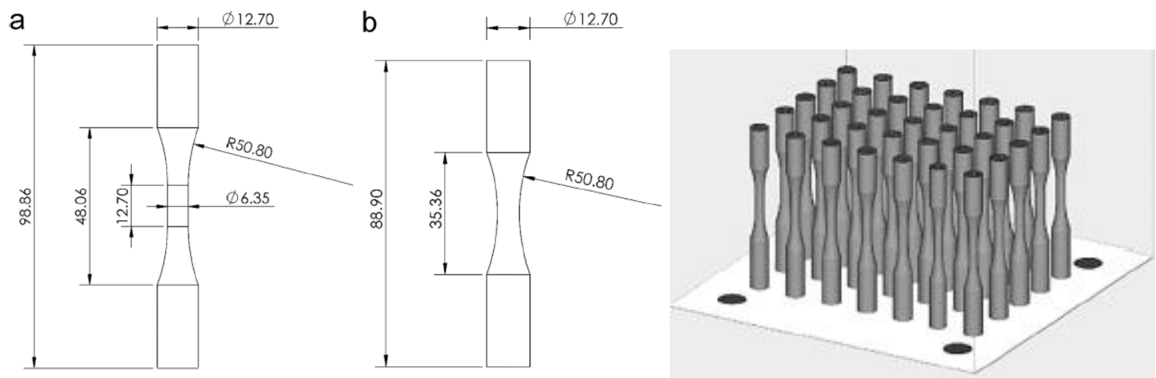

Figure 25—Hourglass Specimen used in the Fatigue Test Campaign to Characterize the S-N Curve

To illustrate, a representative S-N curve (see Figure 26, S-N Curve for the Bracket Material at a 0.1 R Ratio) derived from tests was used in the prediction of fatigue life. The S-N curve should be adjusted to account for other Marin factors such as temperature effects. Note, in this example, the Marin factor associated with surface finish was 1.0, as the S-N curve already accounts for those effects.

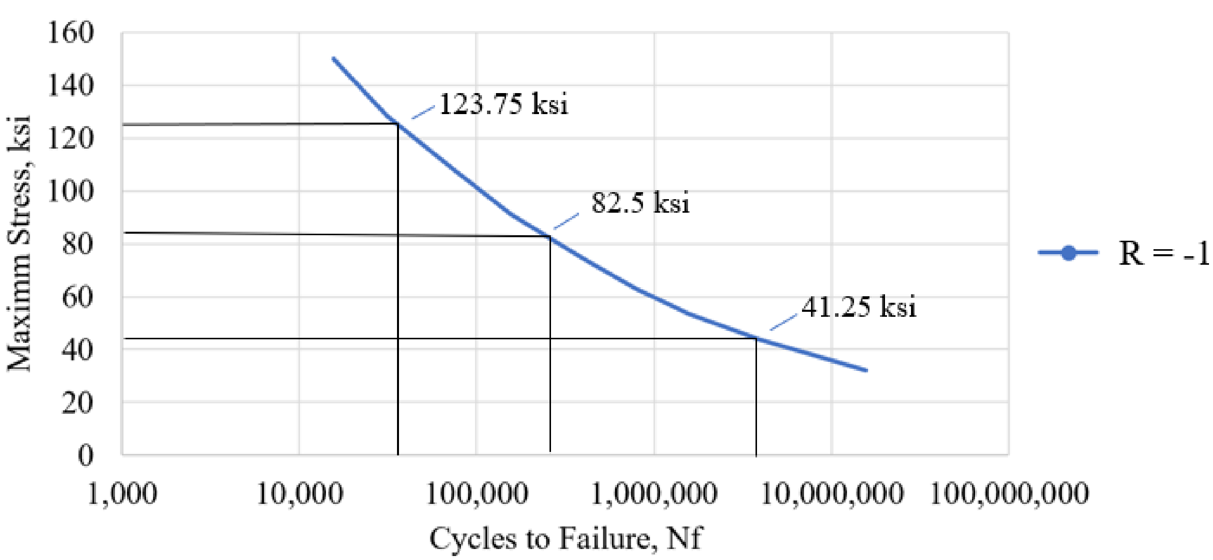

Figure 26—S-N Curve for the Bracket Material at a 0.1 R Ratio

The variation in AM fatigue life should be considered, and a higher scatter factor should be used if a factor of four is observed to be insufficient to adequately capture the scatter in material fatigue capability.

Miner's rule cumulative fatigue damage approach can be used to evaluate the fatigue life based on the distribution of loads in Table 10. The bracket in this example only sees fully reversed loading due to the random vibrations. The amplitude of each cycle is defined by the RMS stress. The maximum stress is multiplied by a 1.5 FAF, and the cycles at each level are multiplied by a scatter factor of 4.0.

The AM fatigue analysis approach in section 5.4 of this Handbook is followed. The percent life used at each load is summed and added in accordance with the Miner's rule cumulative fatigue damage approach. The percent life at each load corresponds to the cycles with the four times the scatter factor divided by the cycles to failure at that load. The breakdown is displayed in Table 11, Load Distribution and Demonstration of Miner's Rule Cumulative Fatigue Damage Approach. The bracket satisfies the requirements with 43.0% total percent life used.

Table 11—Load Distribution and Demonstration of Miner's Rule Cumulative Fatigue Damage Approach

Sigma Level	Maximum Stress (ksi)	R	Maximum stress with 1.5x FAF (ksi)	Cycles at this load	Cycles with 4x scatter factor	Cycles to failure at this load	Percent life used
1	27.5	-1	41.25	38,618	154,472	3,750,000	4.1%
2	55.0	-1	82.50	15,375	31,500	240,000	13.1%
3	82.5	-1	123.75	2,421	9,684	37,500	25.8%
-	-	-	-	-	-	-	Total: 43.0%

A.4 Example 4: Bracket Distortion and Residual Stress Assessment

A.4.1 Introduction

The purpose of this example is to provide background on the prediction of part distortion and residual stresses. AM process simulation tools, using model-informed physics-based methods, provide a method to predict thermomechanical response of the part during manufacturing. The method allows simulation of the layer-by-layer additive process and simulates transient heat transfer response to predict the microstructural and mechanical response.

Part distortion is important even if the part is heat treated, as the distorted part may violate functional requirements (e.g., tolerancing, aerodynamics). Most AM parts are post-processed using a combination of stress relief, HIP to reduce internal voids, and solution heat treatment to target optimal microstructural features. There are situations in which post-processing may not be performed due to time and cost considerations. This is analogous to composite materials, where usually post-processing is not performed to relieve the built-in residual stresses. Understanding residual stresses becomes important if the part is not heat treated, as it can reduce part life. Generally, residual stresses and distortions can be challenging to predict.

Other benefits of performing process simulations include preventing build failures, optimizing designs, understanding where support structures need to be incorporated, providing guidance on the build plate layout, evaluating non-conformances, and aiding in qualification assessments.

AM process simulations are evolving rapidly to address discrepancies observed between the printing process and software models.

A.4.2 Overview of Process Simulations

AM is a process that makes physical parts from digital 3D models. Materials are deposited layer by layer from a tool head following a predetermined scanning path that is generated by slicing the digital model.

A commonly used metal AM process to make liquid rocket engines is PBF. The PBF process starts from spreading a thin layer of powder on the build platform. The cross section of the part is melted layer by layer with a focused heat source such as a laser (selective laser melting [SLM]) or electron beam (electron beam melting [EBM]). The build chamber is filled with inert gas to prevent oxidation or undesired alloying. The localized melting followed by rapid cooling and densification introduces thermal residual stresses and distortion in the parts. If distortion is not controlled during the printing process, unfinished parts could collide with the recoater blade, which would result in a costly build failure and schedule delay. Unrelieved residual stress can negatively affect part strength and safe life. These issues motivate using AM process simulation to study parts behavior before printing to improve manufacturing success and parts performance in service.

AM process simulation models the layer-by-layer building of AM parts, generally in a finite element solver which is the focus going forward. The geometry of the 3D model, support structures, and build plates are represented by finite element meshes. Elements are progressively activated following the laser tool path. Two methods are available to discretize part geometry. One is to use the traditional finite element mesh to model details of the part geometry. Elements of various types are required to simulate the manufacturing of a part with complex geometry.

Voxel meshing, where hexahedron elements are stacked together to approximate the geometry (similar to building parts from interlocking plastic toy bricks), is another possible method.

The following are modeling techniques that have been used:

1. Melt-pool modeling: In this approach, a physics-based model of the melt-pool is developed. A detailed simulation could predict grains or keyhole defects and can be used to create surrogate models. These models can rely on techniques besides or in addition to finite elements.

2. Pseudo melt-pool models: These are finite-element based thermomechanical models that can perform continuum analysis as time and length scales approach melt pools. It can be useful for predicting initial cooling rate and resultant microstructure; however, a full part model can take weeks or months.

3. Inherent strain approach: This approach requires calibration samples to be built and tested to determine principal strains during manufacturing. It requires that the scan pattern, laser power, machine, and other key parameters be held fixed post-calibration. No thermal predictions are generated, which allows solutions in minutes or hours, and it has been found suitable for design (include optimization in the loop).

   The inherent strain approach provides fast predictions based on an estimation of residual strain either from physics or from a micro-welding model (reference Keller [2014]). When inherent strain values are calibrated from a specific machine setup and printing process, results can be reasonably accurate. A simplified formulation called assumed strain approach is shown in Equation 7:

   $$\varepsilon = \mathbf{SSF} * \frac{\boldsymbol{\sigma}_{\mathbf{Y}}}{\mathbf{E}} \qquad \text{(Equation 7)}$$

   where ε is the initial strain applied to the part as it is printed, $\boldsymbol{\sigma}_{\mathbf{Y}}$ is the yield strength, E is the elastic modulus, and SSF is the strain scale factor (reference ANSYS 2021: Users Guide). SSF is solved iteratively to match distortion predicted by simulation with distortion measured from printed coupons.

4. Thermomechanical modeling (reference Dassault Systèmes 3DExperience 2021): A sequentially coupled thermal and mechanical analysis is performed using laser path and power as a function of time. A variable time step size and mesh resolution are chosen depending upon the purpose of the simulation. Typically, calibration tools are used to improve predictions.

A transient heat transfer analysis is performed to solve the thermal history of the part as shown by the balance of thermal energy in Equation 8:

$$\boldsymbol{\rho}\mathbf{Cp}\dot{\mathbf{T}} = -\mathbf{div}\,\mathbf{q} + \mathbf{r} \qquad \text{(Equation 8)}$$

Conduction, convection, and radiation effects are captured during this step. Based on the temperature history, a series of nonlinear static analyses are performed to solve the residual stress and part distortion. This approach is more time consuming but can be more accurate than the inherent strain approach. The following balance of forces in Equation 9 play a key role in the simulations:

$$\boldsymbol{\rho}\ddot{\mathbf{U}} = \mathbf{div}\,\mathbf{T} + \boldsymbol{\rho}\mathrm{b} \qquad \text{(Equation 9)}$$

Several commercial AM process simulation software packages at the time of this writing are available, including ANSYS Additive Suite™ (reference ANSYS 2021: Users Guide), Dassault Systèmes 3DExperience™ (reference Dassault Systèmes 3DExperience 2021: User Guide), Altair™ Inspire (reference Altair Inspire 2021: Users Guide), and MSC Simufact Additive™ (reference MSC Simufact Additive 2019: User Guid**e)**. As tools mature, focus is shifting to how to use these tools in practice: (1) Develop ground rules for how contractors validate analyses so they can facilitate AM adoption, and (2) Identify areas that need development and work with the broader industrial community and software vendors to target those. Analytical predictions using these tools require validation, and the results produced by software should be used with caution as the results can be sensitive to many processing parameters.

A.4.3 Additive Manufacturing (AM) Process Simulation Illustration of a Bracket

The main purpose of this section is to illustrate the application of AM process simulation to predict residual stress and distortion. The bracket is to be built by the PBF process. Detailed thermal-mechanical finite element modeling to predict residual stress is a challenging problem. AM benchmarks are performed from time to time to find best tools and practice to predict residual stress (reference Yang, et al. [2019] and Levine, et al. [2018]). It is not used here due to the amount of computation effort required. Instead, inherent strain approach in ANSYS Additive Suite™ is adopted for this application. The selection of this software is for the purposes of illustrating this example.

Three types of inherent strain simulations are available in ANSYS Additive Suite™ (reference ANSYS 2021: Users Guide). The assumed strain simulation employs an isotropic initial strain assumption per Equation 7 and is the fastest version to run a part simulation. It does not use process parameters but incorporates them indirectly by calibration of the SSF. The scan pattern strain simulation uses scan vector orientation to divide assumed uniform strain into anisotropic initial strain. Scan vectors are either user defined or from a machine's build file. Scan pattern strain simulation takes slightly longer to run. It is more accurate than assumed strain when the scan vector pattern is not randomized but aligned with part geometry. Both simulations are only a mechanical simulation. In addition to anisotropic initial strain, the thermal strain simulation accounts for the effects of thermal cycling on strain accumulation, where heating above temperature threshold increases strain and remelting reduces strain to base strain. It uses process parameters such as laser power, spot size, and scanning path to perform thermal simulations to obtain thermal history. The temperature field is applied to subsequent thermal mechanical analysis to perform mechanical prediction. It takes much longer to run than the previous two types of simulations but provides the highest degree of accuracy. All three methods require calibration of SSFs by matching distortion prediction with measurements from printed coupons.

Both assumed strain and thermal strain simulation in ANSYS Additive Suite™ were performed, with the assumed strain approach an order of magnitude faster than the other approach. Residual stresses and distortion predictions were similar for the simple bracket geometry. For large parts, thermal strain simulation may take weeks or months to finish, which can be longer than machine printing time to make these parts. On the other hand, assumed strain simulation can provide quick predictions before printing, which give engineers opportunities to evaluate printing success and part performance, and to redesign parts if necessary. Therefore, results from assumed strain approach are shown here for illustration purposes, while also noting that these types of predictions do require test validation.

The bracket computer-aided design (CAD) model is imported into ANSYS Additive Suite™. It is positioned directly on the build plate, shown by Figure 27, Build Orientation of Inconel® 718 Bracket. The CAD model is discretized into voxel elements of 0.5 mm to perform simulation, shown by Figure 28, Voxel Mesh of Inconel® 718 Bracket. Elasto plastic properties of Inconel® 718 material provided by the software are used in the simulation. The SSF is used to scale the amount of initial strain applied to the structure. A calibration of SSF should be performed so that distortion prediction matches distortion measurement of coupons printed with the same process parameters as those used for the bracket.

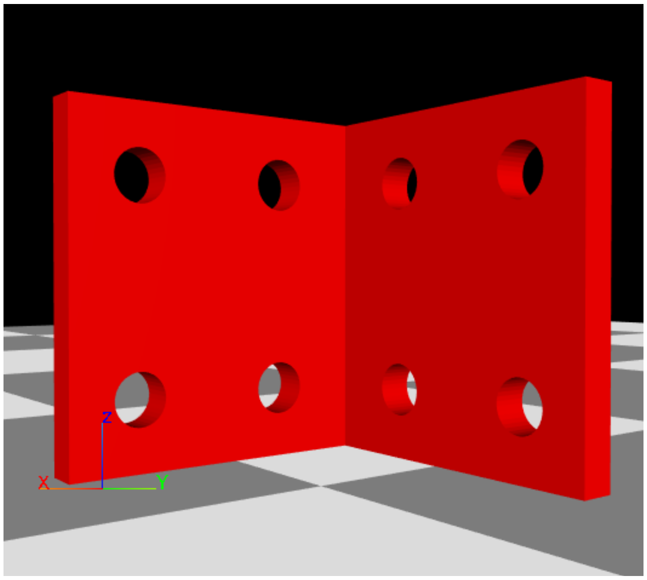

Figure 27—Build Orientation of Inconel® 718 Bracket

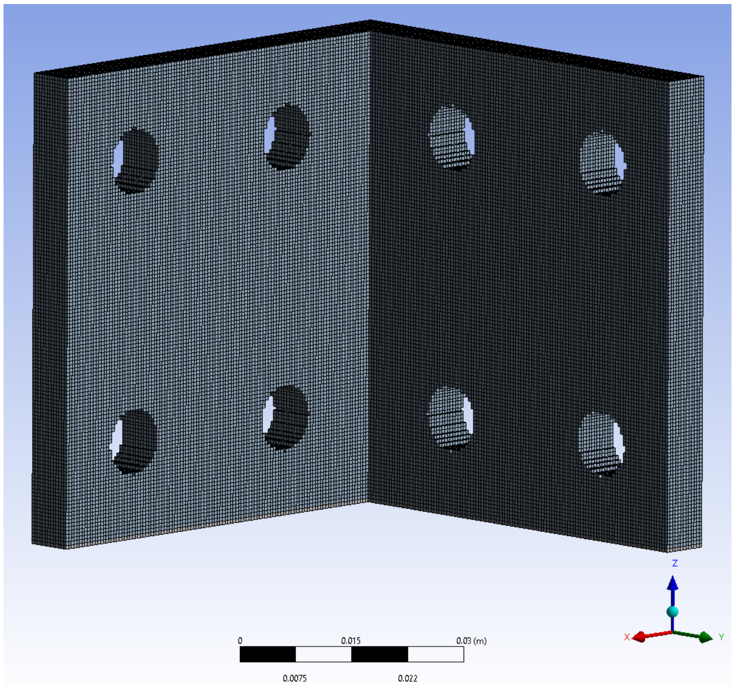

Figure 28—Voxel Mesh of Inconel® 718 Bracket

All parts “virtually” manufactured in this study use Concept Laser M2 instrument parameters: Nitrogen environment at a laser power of 370 W, scan rate of 900 mm/s, and focused spot size of 50 μm.

The printing process is simulated in ANSYS Additive Suite™. Figure 29, Simulated Printing Process of the Bracket, shows an intermediate stage of the printing process and distortion contour. Figure 30, Distortion, shows the distortion after printing finished. Figure 31, Residual Stress Predictions, shows residual stresses in the bracket.

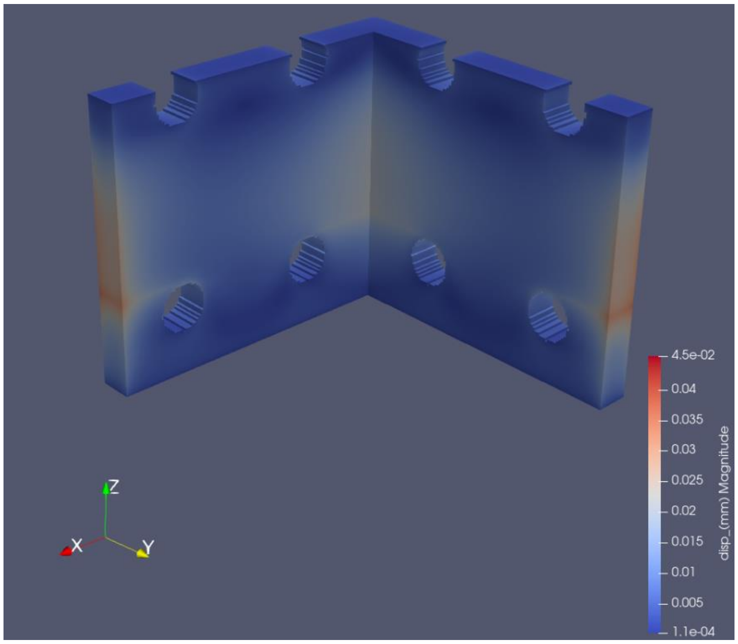

Figure 29—Simulated Printing Process of the Bracket

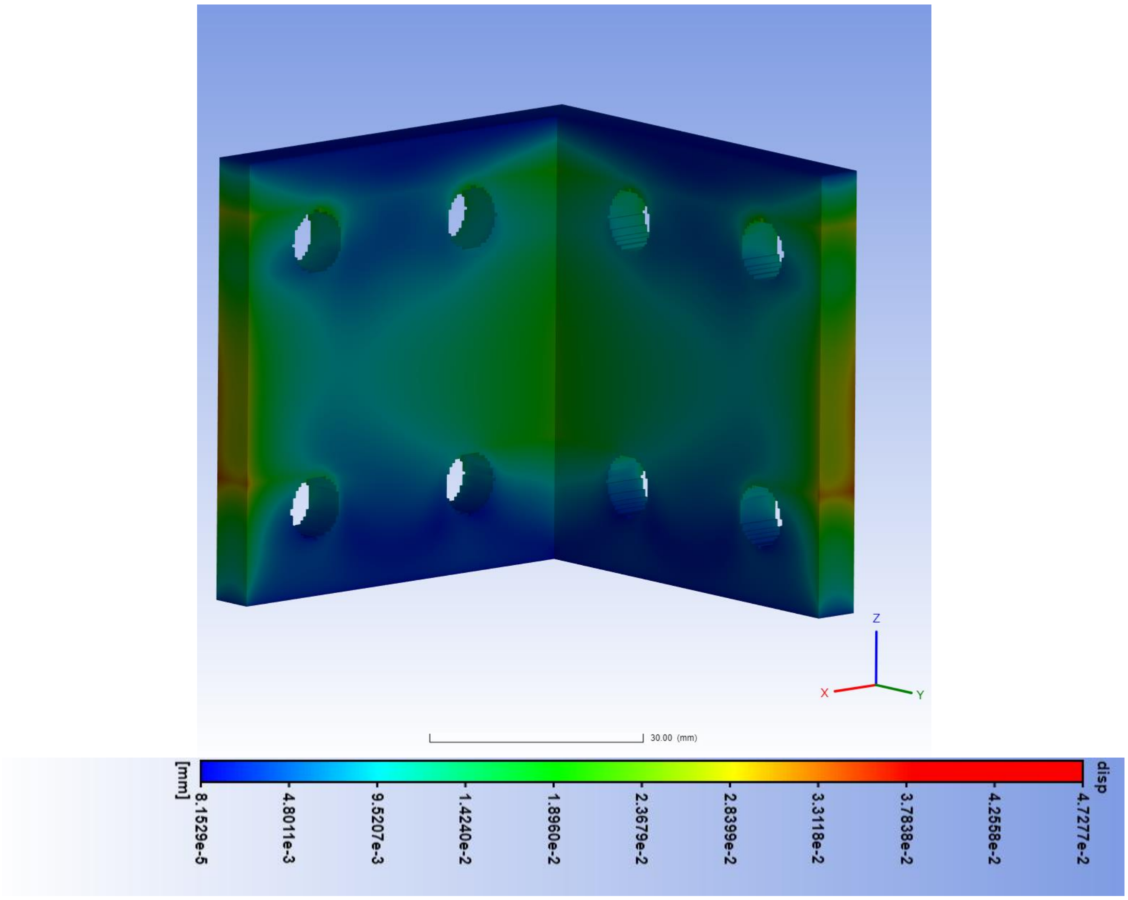

Figure 30—Distortion

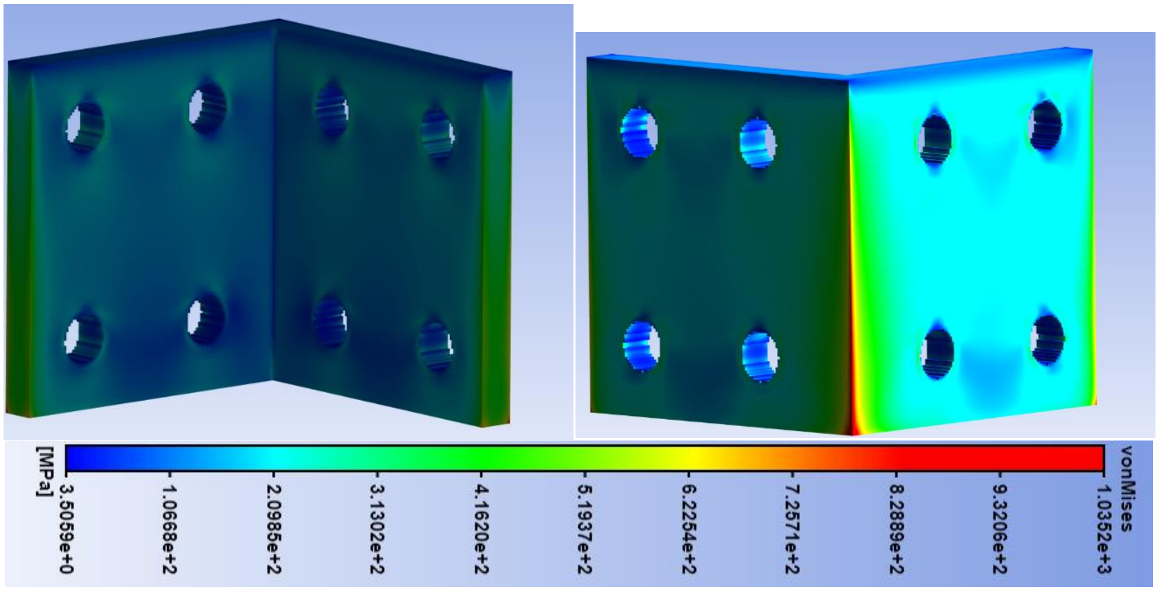

Figure 31—Residual Stress Predictions

To predict life of the part, typically the stress tensors due to the residual stresses from the printing process are linearly superimposed with those stemming from the loading environments in service as shown in Equation 10:

$$\begin{bmatrix}\sigma_{xx} & \sigma_{xy} & \sigma_{xz}\\ \sigma_{xy} & \sigma_{yy} & \sigma_{yz}\\ \sigma_{xz} & \sigma_{yz} & \sigma_{zz}\end{bmatrix}_{Total\ Stress} = \begin{bmatrix}\sigma_{xx} & \sigma_{xy} & \sigma_{xz}\\ \sigma_{xy} & \sigma_{yy} & \sigma_{yz}\\ \sigma_{xz} & \sigma_{yz} & \sigma_{zz}\end{bmatrix}_{Residual\ Stress} + \begin{bmatrix}\sigma_{xx} & \sigma_{xy} & \sigma_{xz}\\ \sigma_{xy} & \sigma_{yy} & \sigma_{yz}\\ \sigma_{xz} & \sigma_{yz} & \sigma_{zz}\end{bmatrix}_{In-Service\ Stress} \quad \text{(Equation 10)}$$

The approach can be conservative in the cases where the total stress exceeds the yield stress of the material. An approach that is less conservative is specifying the predicted residual stress as the initial stress prior to analysis of in-service environments. The strength, buckling, fracture, and fatigue assessments can then be performed in a straightforward manner following the procedures outlined in this Handbook. Residual stress predictions require validation. Methods to measure residual stresses are discussed in the next section.

A.4.4 Residual Stress Measurements in Additive Manufacturing (AM)

Large temperature gradients and high cooling rates associated with AM processes often result in residual stresses building up during production, resulting in part warpage or even complete build failure. Measuring the magnitude of residual stress is an important capability that can help inform design decisions to minimize negative effects on AM builds and can be used to validate process simulation models.

Residual stress measurement techniques for AM parts include nondestructive and destructive testing, which covers a wide range of length scales, depths of penetration, and measurement accuracies. Popular nondestructive techniques include ND (reference Wu, et al. [2014] and Masoomi, et al. [2017]) and UT (reference Acevedo, et al. [2020]). ND testing utilizes a neutron source and diffractometer to measure internal residual stresses and strains. These measurements are based on the diffraction of neutrons and are sensitive to the lattice parameters of the medium through which they are passed, which will vary based on the presence of internal residual stresses and strains. The standard method for ND testing, ISO 21432:2019, Non-destructive testing - Standard test method for determining residual stresses by ND, outlines in detail the procedures that should be used to acquire accurate measurements. ND has excellent spatial resolution and is a very accurate test technique but is expensive relative to other residual stress measurement techniques due to the highly specialized and complex equipment necessary. UT measurements analyze mechanical waves as they propagate through a medium at frequencies typically greater than 20 kHz. This technique relies on acoustoelastic theory, which relates the velocity of elastic wave propagation to mechanical stresses in a body (reference Acevedo, et al. [2020]). With a varying amount of residual stress in a given component, the corresponding wave propagation velocity will change. While this technique is highly versatile for use on multiple materials and geometries, it offers limited spatial resolution and cannot separate multi-axial stresses.

Destructive test methods for determination of residual stresses are typically used to measure macro-levels of residual stress, which result in the bulk deformation of samples. These techniques include hole drilling (reference Swain, et al. [2019]) and bulk deformation measurement (reference Mugwagwa, et al. [2018]). The hole-drilling strain gauge method follows ASTM E837-13a, Standard Test Method for Determining Residual Stresses by the Hole-Drilling Strain-Gage Method, and can be used to determine near-surface residual stresses. Strain rosettes are attached to the sample surface, and a hole is drilled at their geometric center. The resulting relieved strain can be correlated to the residual stress that was present at the surface based on linear elasticity equations. Bulk deformation measurements can be made in multiple ways, sharing the common thread of measuring macro-scale plastic deformation in a sample caused by residual stresses in the parts. Typically, measurements are made on a specified geometry while it is still attached to the build substrate. Next, the part is excised from the plate incrementally or all at once, and a complementary set of measurements is made to determine the distortion that occurs due to the relief of residual stress that resided in the part. These destructive methods of measuring residual stresses can provide users with clear results of bulk deformation at a fraction of the cost of some of the more expensive nondestructive techniques.

Both nondestructive and destructive techniques offer distinct advantages and disadvantages to be considered by users to determine what the most appropriate measurement technique is for their specific application. While this is not an exhaustive list of residual stress measuring techniques, it provides a brief sampling of what is used in the AM industry today.

See reference for methods to predict residual stresses in welds (reference Schajer [2013]. While weld modeling is not the same manufacturing method as AM, it is useful to understand residual stress measurement approaches for welds. The U.S. Nuclear Regulatory Commission and the Electric Power Research Institute funded a series of mock-up problems where different modelers made blind predictions of weld residual stress (WRS) fields. Mock-up welds were made on a number of different geometries, and measurements were taken using four different measurement techniques, including deep hole drilling (invented by VEQTER Ltd), contour method (Hill Engineering), ND, and several surface measurement methods (X-ray, hole drilling, etc.). There were scatter and differences between measurements made from the different methods. The conclusions were that the contour method and deep hole drilling were best, and ND was adequate but less accurate. The latter is because the focus was dissimilar metal welds between ferritic to stainless pipe/plate using Alloy 182 weldment and lattice spacing was difficult to measure. Comparing surface measurements (hole drilling, X-Ray, etc.) to model predictions was not a good indicator of model accuracy because through thickness WRS fields are key to fatigue and SCC crack growth predictions. It is unknown whether these conclusions translate to AM processes.

A.4.5 Summary

The purpose of this example is to illustrate the state-of-the-art in process simulation of AM parts, explain the motivation and challenges of process simulation, and illustrate the simulation process using two different methods. These analytical methods are capable of predicting part distortion and residual stresses but should be validated against experimental measurement techniques before evaluating space hardware. These validated models can be used to prevent build production failures, assess non-conforming conditions, and evaluate tolerance stack-up issues and other functional requirements. As numerical methods evolve and computational power increases, these analytical tools will become essential in ensuring optimal builds and lower scrap rates.

A.5 Example 5: Bracket Strength and Fatigue

This example outlines the application of this Handbook to assess the strength and fatigue requirements for a flight-critical Inconel® 718 bracket produced using AM techniques (reference Christensen, R. M. (2013), Chapters 4, 5, and 11**)**.

A.5.1 Dimensions and Loads

The bracket dimensions and loads are shown in Figure 32, Bracket Dimensions and Design Loads. The bracket is bolted in the y-z and x-z planes. In the design limit load case, the bolts in the y-z plane undergo a small displacement in the negative x-direction relative to the bolts in the x-z plane. Additionally, 400 lbf in the z-direction are distributed among the bolts in the y-z plane.

The bracket is printed such that layers are added in the z-direction, as shown in Figure 33, Bracket Print Orientation.

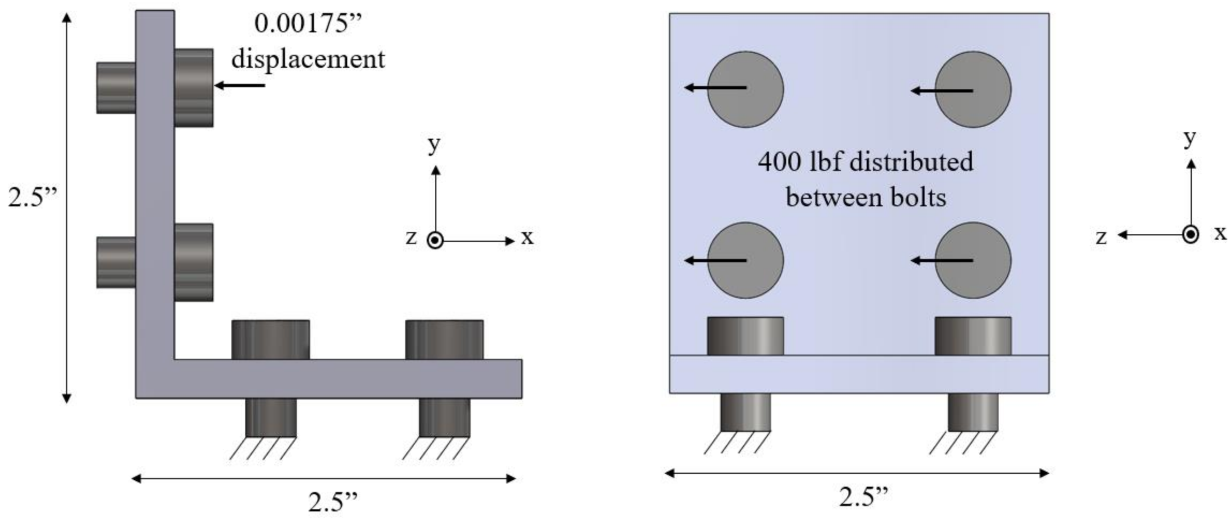

Figure 32—Bracket Dimensions and Design Loads

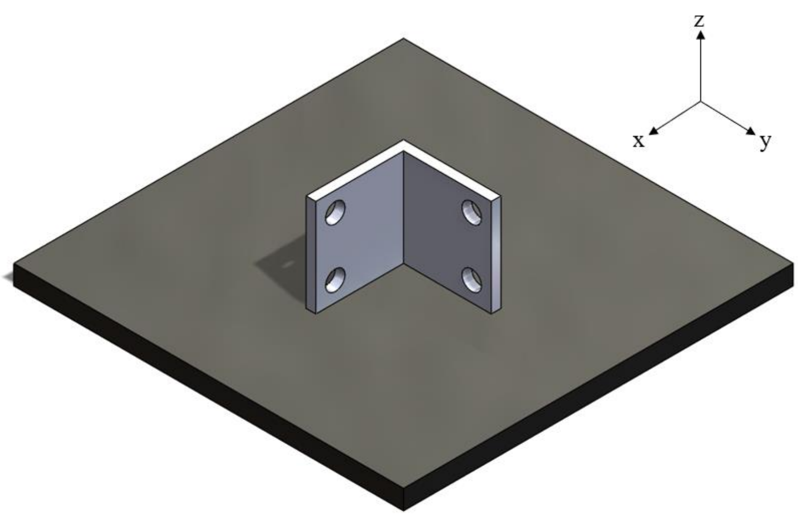

Figure 33—Bracket Print Orientation
(The z-axis is aligned with the print direction.)

A.5.2 Additive Manufacturing (AM) Classification

First, the AM part classification is established in accordance with section 5 of this Handbook. Figure 2 of this Handbook provides a visualization of the part classification breakdown for AM parts. The bracket has a high consequence of failure and therefore is classified as a Class A part.

The part’s secondary classification is determined by the structural demand, which is outlined for metallics based on NASA-STD-6030, Table 2; and the AM risk, which is determined using NASA-STD-6030, Table 4. The bracket has a high structural demand because it does not meet multiple criteria in NASA-STD-6030, Table 2. The AM risk is low because the bracket has a simple design and is easily inspected, yielding an AM risk score of zero from NASA-STD-6030, Table 4. With a high structural demand and low AM risk, the bracket falls under Class A2.

A.5.3 Material Properties

The AM part material property data is established in accordance with NASA-STD-6030. The material properties are obtained from an MPS in compliance with both NASA-STD-6016 and NASA-STD-6030. The AM bracket is a Class A part and subject to the mechanical property testing per NASA-STD-6030, section 5.5.3.5. AM parts often demonstrate anisotropy, which should be reflected in the material properties.

The properties in this example are given in Table 12, AM Bracket Material Properties Representative of an AM Material. The values are representative of an AM part and represent realistic trends in AM parts. The allowable values are taken from the worst-case thermal environment allowed during flight.

Typically, a test program is needed to characterize the modulus, strength, fracture toughness, and elongation. AM parts tend to exhibit a degree of anisotropy so directional mechanical properties need to be characterized, similar to composite materials. Many AM parts exhibit transverse isotropy such that the properties are isotropic in-plane but differ out-of-plane. In this example, the bracket material is assumed to be transversely isotropic.

Table 12—AM Bracket Material Properties Representative of an AM Material

Property	Value in XY Plane	Value in Z Direction
Young's Modulus, E	29,000 ksi	28,000 ksi
Shear Modulus, G_{XY}	11,000 ksi	-
Shear Modulus, G_{XZ}, G_{YZ}	-	10,000 ksi
Poisson's Ratio, ν_{XY}	0.33	-
Poisson's Ratio, ν_{XZ}, ν_{YZ}	-	0.3
Tensile Yield Strength	130 ksi	120 ksi
Compressive Yield Strength	140 ksi	135 ksi
Ultimate Tensile Strength	175 ksi	160 ksi
Ultimate Compressive Strength	190 ksi	180 ksi
Fracture Toughness	25 ksi $\sqrt{\text{in}}$	20 ksi $\sqrt{\text{in}}$
Elongation	10 %	11 %

A.5.4 Structural Analysis

The bracket is evaluated with the minimum FoS and test factors in section 5.1.2 and Table 1 of this Handbook. This is a metallic structure, so the corresponding row of Table 1 is chosen.

The structural analysis is completed in accordance with section 5.2 of this Handbook. If the metal is ductile and has little or no anisotropy, the von Mises failure criteria can be used for the stress analysis of AM parts. NASA-STD-6030 specifies that anisotropy is typically considered negligible if there is less than a 5% difference in properties by orientation.

In this example, the bracket material properties vary by more than 5%, so a maximum principal stress criterion or an alternative failure criterion can be used.

To evaluate this bracket, the maximum principal stress is determined and compared to the minimum allowable strength. The stresses are analyzed with a FEM. If necessary, a buckling analysis that accounts for anisotropy should be completed, but the bracket in this example is not susceptible to buckling.

Section 5.2.3 of this Handbook calls for mitigation of residual stresses. The analysis should include the residual stresses, or it should be demonstrated that the residual stresses are enveloped by the analysis method. The bracket in this example is annealed and confirmed not to have residual stresses with X-ray diffraction.

A linear static analysis is completed with the design load. The maximum absolute principal stress distribution is shown in Figure 34, Contour and Vectors of the Maximum Absolute Principal Stresses in the Bracket, and Figure 35, Contours of Directional Stresses in the Bracket. The bracket sees a maximum tensile principal stress of 52.1 ksi and a maximum compressive principal stress of 81.1 ksi.

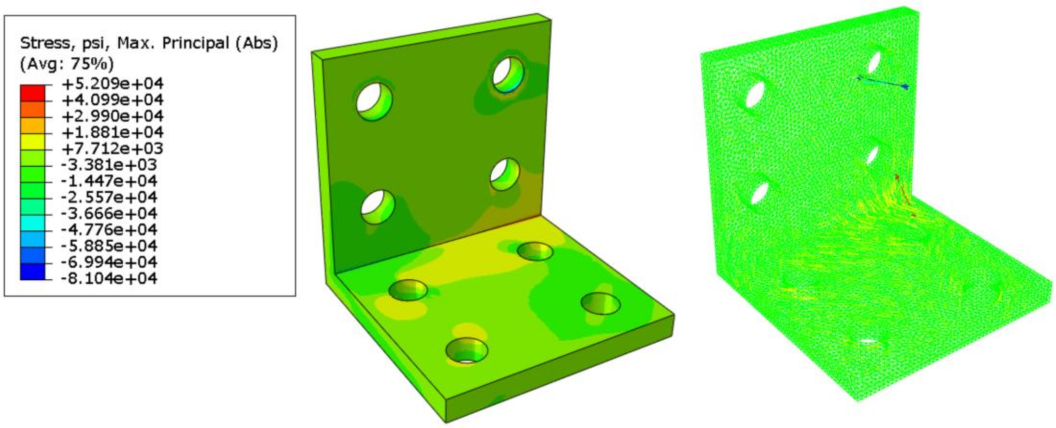

Figure 34—Contour and Vectors of the Maximum Absolute Principal Stresses in the Bracket

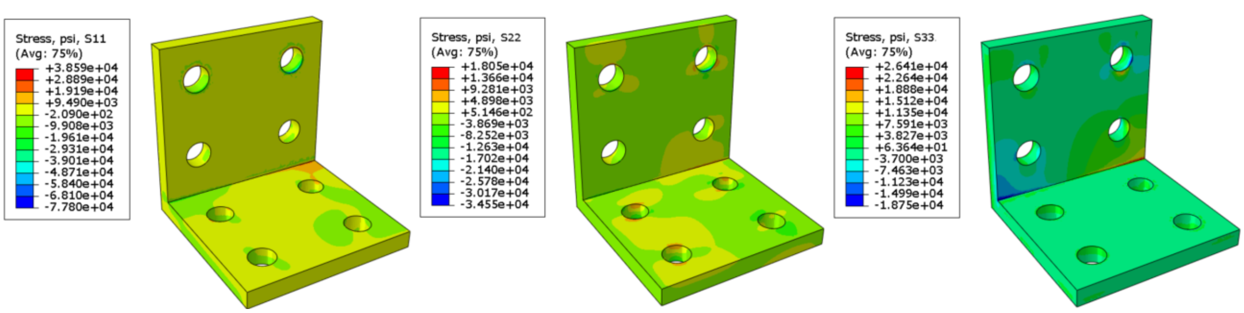

Figure 35—Contours of Directional Stresses in the Bracket
(The contours for the x, y, and z stresses are shown from left to right.)

The margins are calculated for the maximum tensile and compressive principal stresses using Equations 2 and 3 from section 5.1.3 of this Handbook. The bracket is a Class A part, so an AM factor is used as shown in Table 13, Margin of Safety Calculations for the Transversely Isotropic Bracket. The allowable yield and ultimate stresses are taken from the direction with the minimum values. In this case, all the minimum allowable values are in the z-direction. The allowable values are taken from the worst-case thermal environment allowed during flight.

The MS corresponding to the maximum principal stresses are presented in Table 13. The yield and ultimate MS meet the requirements, as they are greater than or equal to zero.

Table 13—Margin of Safety Calculations for the Transversely Isotropic Bracket

Stress	Value	Factor of Safety		AM Factor For Class A Parts		Allowable Values in Minimum Strength Direction		Margin of Safety	
		Yield	Ultimate	Yield	Ultimate	Yield	Ultimate	Yield	Ultimate
Maximum Tensile Principal Stress	52.1 ksi	1.25	1.4	1.2	1.3	120 ksi	160 ksi	0.53	0.69
Maximum Compressive Principal Stress	81.1 ksi	1.25	1.4	1.2	1.3	135 ksi	180 ksi	0.11	0.22

A sensitivity study with isotropic bounding stiffness was completed. This step may not be necessary in every case, but it increases confidence if testing shows a large variation in modulus from build to build. As part of the manufacturing readiness, it was found that the transverse modulus varied up to ±20% during initial builds. Therefore, the analysis considered these variations to understand impact of modulus variations on the margins. While the additional AM factor could help cover these types of uncertainties, the analysis was performed to increase confidence in the design despite modulus variations. Two fully isotropic models were run: one with an isotropic elastic modulus 20% greater than the transversely isotropic model, and one with an isotropic elastic modulus 20% less than the transversely isotropic model. The margins are compared in Table 14, Comparison of Margins of Safety for the Transversely Isotropic Model and the Bounding Isotropic Models.

The von Mises stress margins are lower than the principal stress margins for the transversely isotropic because the largest magnitude principal stress is compressive. The compressive principal stress margins are calculated with the compressive allowables, which are larger than the tension allowables. The von Mises stress margin is calculated with the lowest yield and ultimate strengths, which are the tensile in the z-direction.

The isotropic modulus models with ± 20% modulus bound the transversely isotropic results. The negative yield margin for the +20% modulus case is deemed acceptable with non-detrimental yielding, as it is localized to the edge of a bolt hole and does not adversely affect the form, fit, function, and integrity of the bracket. The bounding isotropic modulus cases have acceptable margins, which gives confidence that the bracket will have positive margins from build to build.

Table 14—Comparison of Margins of Safety for the Transversely Isotropic Model and the Bounding Isotropic Models

Model	Stress	Value	Factor of Safety		AM Factor For Class A Parts		Allowable Values in Minimum Strength Direction		Margin of Safety	
			Yield	Ultimate	Yield	Ultimate	Yield	Ultimate	Yield	Ultimate
Transversely Isotropic	Max Compressive Principal Stress	81.1 ksi	1.25	1.4	1.2	1.3	135 ksi	180 ksi	0.11	0.22
Transversely Isotropic	Maximum Tensile Principal Stress	52.1 ksi	1.25	1.4	1.2	1.3	120 ksi	160 ksi	0.53	0.69
Transversely Isotropic	von Mises	75.3 ksi	1.25	1.4	1.2	1.3	120 ksi	160 ksi	0.06	0.17
Isotropic with Modulus +20%	von Mises	81.1 ksi	1.25	1.4	1.2	1.3	120 ksi	160 ksi	-0.01	0.08
Isotropic with Modulus -20%	von Mises	64.5 ksi	1.25	1.4	1.2	1.3	120 ksi	160 ksi	0.24	0.36

Alternative failure criteria that better represent the behavior of the AM material can be used. Similar to composite materials, it is essential that an appropriate failure criterion is used and validated by test if the material exhibits anisotropy. Incorrect failure criteria, such as the maximum principal or Hoffman, can potentially lead to an under-designed part. In this example, the PIFC can be used to account for a difference between tensile and compressive strengths. The PIFC is a failure criterion similar to the von Mises criterion, but it accounts for a difference between tensile and compressive strengths (reference Keller [2014]. Note that the PIFC reduces to the von Mises failure criterion when the tensile and compressive strengths are equal.

The isotropic form of the PIFC model is as shown in Equation 11:

$$\left(\frac{1}{T}-\frac{1}{C}\right)(\sigma_1+\sigma_2+\sigma_3)+\frac{1}{2TC}\left[(\sigma_1-\sigma_2)^2+(\sigma_2-\sigma_3)^2+(\sigma_3-\sigma_1)^2\right]\leq 1 \quad \text{(Equation 11)}$$

To include the FoS and AM factor, the tension and compression allowable strengths can be divided by both factors. This yields the version of the PIFC in Equation 12:

$$(FS*AM\ Factor)\left(\frac{1}{T}-\frac{1}{C}\right)(\sigma_1+\sigma_2+\sigma_3)+\frac{(FS*AM\ Factor)^2}{2TC}\left[(\sigma_1-\sigma_2)^2+(\sigma_2-\sigma_3)^2+(\sigma_3-\sigma_1)^2\right]\leq 1$$

(Equation 12)

When the ratio of tensile strength to compressive strength (T/C) is less than ½, the material is considered brittle. The PIFC is further restricted as shown in Equation 13:

$$if\ \ \frac{T}{C}\leq\frac{1}{2}\ \ then\ \ \sigma_1,\sigma_2,\sigma_3\leq T \quad \text{(Equation 13)}$$

This tensile to compressive strength ratio should be completed with the minimum value direction tensile strength and maximum value direction compressive strength. In this case, the ratio is above ½ for both yield and ultimate strengths, so the material is not brittle.

The PIFC equation can be expanded to account for orthotropic or transversely isotropic material properties. In this example, the isotropic PIFC equation is evaluated with different combinations of directional tensile and compressive strengths to verify that the PIFC is not violated. The maximum value of the left side of the PIFC equation was calculated and verified to be less than or equal to one. Note that the PIFC should be evaluated throughout the bracket, not just at the location of the maximum principal or von Mises stress. In this example, a custom output for the left side of the PIFC equation was created in the FEM results to visualize the PIFC results throughout the bracket. The values in Table 15, Verification of the Polynomial Invariant Failure Criterion (PIFC), show the PIFC equation left side for all combinations of tensile and compressive strengths. The maximum value occurs at the location of the maximum absolute principal stress, where the principal stresses are 12.6 ksi, -32.9 ksi, and -81.1 ksi.

The margin can be calculated as the inverse of the left side of the PIFC equation subtracted by one. In all cases, the margin is greater than zero, so the bracket passes the PIFC with the appropriate FoS and AM factor.

In this example, the PIFC margins are higher than those calculated with the maximum principal stress and von Mises failure criteria. This is because PIFC accounts for the increased compressive strength. Depending on the stress state, PIFC margins can be higher or lower than those calculated with other failure criteria. The appropriate failure criterion should be determined and validated through test.

Table 15—Verification of the Polynomial Invariant Failure Criterion (PIFC)

Strength	Factor of Safety	AM Factor	Tensile Strength (ksi)	Compressive Strength (ksi)	Maximum value of PIFC Equation Left Side	Margin for Each Combination	Minimum Margin
Yield	1.25	1.2	120 (Z)	135 (Z)	0.77	0.30	0.25 Yield
			120 (Z)	140 (X,Y)	0.70	0.43	
			130 (X,Y)	135 (Z)	0.80	0.25	
			130 (X,Y)	140 (X,Y)	0.73	0.37	
Ultimate	1.4	1.3	160 (Z)	180 (Z)	0.63	0.59	0.52 Ultimate
			160 (Z)	190 (X,Y)	0.54	0.85	
			175 (X,Y)	180 (Z)	0.66	0.52	
			175 (X,Y)	190 (X,Y)	0.57	0.75	

A.5.5 Qualification and Proof Testing

Structural integrity qualification should be completed in accordance with section 5.3 in this Handbook. The bracket is a Class A structure and therefore is qualified by testing. Additionally, the flight component should undergo proof testing. Table 1 of this Handbook is used to determine a 1.2 proof test factor for the bracket. Additionally, room temperature tests include an ECF. Multiple ECFs should be calculated based on the knockdown of strength and fracture toughness at design limit, with anisotropy considered. This process is outlined in Table 16, Environmental Correction Factor (ECF) Calculations. For the bracket material, the ECF is controlled by the fracture toughness in the z-direction, which is as low as 20 ksi $\sqrt{\mathbf{in}}$ during flight. The minimum flight fracture toughness in the z-direction is 1.1x less than at room temperature. This is the greatest knockdown of the allowable values, so a 1.1 ECF is required. The 1.2 proof test factor and 1.1 ECF combine for a 1.32 factor during test.

Table 16—Environmental Correction Factor (ECF) Calculations

Property	Minimum Value in Operating Temperature Range	Room Temperature Value	ECF
Fty,xx/Fty,yy	130 ksi	136 ksi	1.05
Fty,zz	120 ksi	125 ksi	1.04
Fcy,xx/Fcy,yy	140 ksi	143 ksi	1.02
Fcy,zz	135 ksi	138 ksi	1.02
Ftu,xx/Ftu,yy	175 ksi	185 ksi	1.06
Ftu,zz	160 ksi	168 ksi	1.05
Fcu,xx/Fcu,yy	190 ksi	195 ksi	1.03
Fcu,zz	180 ksi	185 ksi	1.03
K_{IC},xx/K_{IC},yy	25 ksi $\sqrt{\text{in}}$	27 ksi $\sqrt{\text{in}}$	1.08
K_{IC},zz	20 ksi $\sqrt{\text{in}}$	22 ksi $\sqrt{\text{in}}$	1.10
			Maximum ECF: 1.10

If the qualification test cannot be loaded or fixed in the same manner as flight, a different load and fixture can be used; however, the load should be increased to bound the expected structural response of the component under nominal design loads. For example, if the 0.00175-in displacement cannot be easily applied to the bracket, the 400 lbf can be increased such that the test factor is met. A test model should analytically predict that the stresses meet the required levels throughout the bracket.

Alternatively, if it is found that the test load cannot be altered to envelop the flight case, a qualification by similarity rationale can be used based on a critical section of the bracket that meets the 1.32 factor. The qualification by similarity rationale should consider anisotropy, as the directional stresses are not necessarily enveloped if the von Mises stress is enveloping. Any deviation from testing in the same manner as flight is approved by the delegated NASA Technical Authority.

When performing a qualification test, it is key to measure performance of the qualification article to validate the analysis models. Minimizing the error between the model and testing increases confidence in analytical assessments. Errors in the models can impact fatigue analysis and can result in orders of magnitude error in life predictions. Analysis is also relied upon for evaluating some non-conforming conditions that occur after proof testing and to formulate robust acceptance and qualification programs. Depending on the program, analysis correlations may not be required. When correlation is required, displacement and/or strain correlation should meet the specified criteria. In this example, the deflection corresponding to the force applied was measured during qualification testing. The deflection was observed to be greater than 10% but less than 20% of the deflection in the analysis. Confidence was gained in the sensitivity analysis varying the modulus ± 20%.

A.5.6 Fatigue

The fatigue guidance of section 5.4 of this Handbook is followed to evaluate the bracket. The bracket undergoes LCF, so a fatigue assessment that demonstrates four service lives is performed. In addition, a FAF is applied to the loads in the fatigue analysis. As per Table 2 in this Handbook, a 1.5 FAF is used for the fatigue assessment of Class A parts.

In this example, it is assumed that the bracket has a machined surface. Other considerations are needed for as-built AM surface finishes.

The fatigue life is very sensitive to the AM process and build orientation. The variation in AM fatigue life should be considered, and a higher scatter factor should be used if a factor of four is observed to be insufficient to adequately capture the scatter in material fatigue capability. Compressive stresses do not drive a crack and are ignored. As demonstrated in the structural analysis, the maximum tensile principal stress at the design load case is 52.1 ksi. The 52.1 ksi stress occurs when the mean stress is combined with worst-case dynamic loads. A separate analysis determined that the mean stress is 40 ksi. The worst-case design load includes 12.1 ksi amplitude dynamic stress.

The bracket sees 10,000 cycles during flight, so a fatigue assessment shows a service life of at least 40,000 cycles. During flight, the dynamic stresses vary from cycle to cycle. Only a fraction of cycles has a maximum stress close to the 52.1 ksi maximum stress. Miner’s rule cumulative fatigue damage approach can be used to evaluate the fatigue life based on a load spectrum. The load spectrum is determined with a statistical analysis of the structural dynamics. An example load spectrum for this bracket is presented in Table 17, Load Spectrum and Demonstration of Miner’s Rule Cumulative Fatigue Damage Approach.

The amplitude of each cycle is determined with a statistical analysis. The 500 cycles have an amplitude between 7.5 ksi and 12.1 ksi. In the fatigue assessment, the amplitude for these cycles is conservatively assumed to be 12.1 ksi, which corresponds to a 52.1 ksi maximum stress. With the 1.5 FAF and 4x scatter factor, this corresponds to 2,000 cycles with a 78.2 ksi maximum stress. Table 17 provides the breakdown for the maximum stresses for the 10,000 cycles.

Table 17—Load Spectrum and Demonstration of Miner's Rule Cumulative Fatigue Damage Approach

Amplitude	Maximum stress (ksi)	R	Maximum stress with 1.5x FAF (ksi)	Cycles to failure at this load	Cycles at this load	Cycles with 4x scatter factor	Percent life used
12.1	52.1	0.54	78.2	22,000	500	2,000	9.1%
7.5	47.5	0.68	71.3	31,000	750	3,000	9.7%
5	45	0.78	67.5	40,000	1,250	5,000	12.5%
2.5	42.5	0.88	63.8	47,000	7,500	30,000	63.8%
-	-	-	-	-	Total: 10,000	-	Total: 95.1%

A S-N curve is used to demonstrate the fatigue life for the bracket. The S-N curve should be obtained from a NASA-approved source or with test coupons from the same AM machine and process as the flight part. All directions are tested with the data from the lowest life orientation. The data should consider anisotropy, and the material source or tests comply with NASA-STD-6030 requirements.

The S-N curve in Figure 36, S-N Curve for the Bracket Material at a 0.5 R Ratio, is used to evaluate the bracket. Note that the R ratio is above 0.5 for all load cases. It is conservative to use a S-N curve derived from a lower R ratio, so the curve in Figure 36 is appropriate to analyze this load spectrum.

The AM fatigue analysis approach in section 5.4 of this Handbook is followed. The percent life used at each load is summed and added in accordance with the Miner's rule cumulative fatigue damage approach. The percent life at each load corresponds to the cycles divided by the cycles to failure at that load. The bracket satisfies the requirements with 95.1% total percent life used, as shown in Table 17.

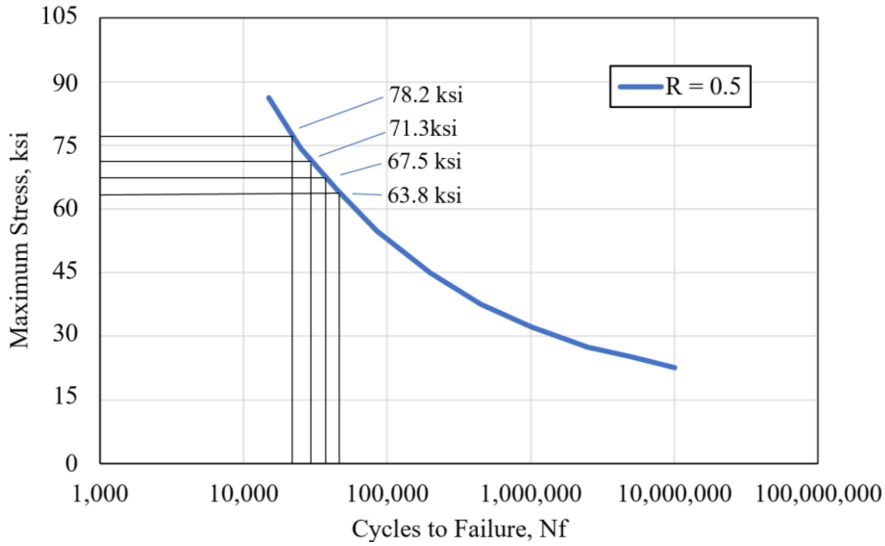

Figure 36—S-N Curve for the Bracket Material at a 0.5 R Ratio