23 Space Logistics and Manufacturing

23.4 Space Mission Verification and Validation

23.4.5.1 Environmental Testing and Defect Screening

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It is rarely practical to exercise an end item or test article in the full mission environment and duration before committing it to space deployment and fielded service. (One counter-example is the Pioneer Jupiter series, which encountered Jupiter in 1973, path-finding the Solar System grand tour nearly six years before the 1979 launches of Voyagers 1 and 2. Lessons learned were used to modify equipment on the subsequent spacecraft, especially for radiation hardness.) More commonly, proving suitability for deployment and use in the space environment requires subjection of the flight article to ground test conditions that represent stresses of transportation, launch, and space flight. Table 23-10 describes a number of test regimens used to qualify and accept space flight equipment. Practice has shown that these methods can be used to promote timely qualification through accelerated life testing.

Accelerated life testing, as used here, is a quantitative approach to predicting life of a fielded space asset, by subjecting that article and similar ones to stresses exceeding those expected to be experienced during the mission. Traditional methods of reliability testing, qualification, and acceptance, have seen innovations such as test-to-failure and *highly accelerated life testing* (HALT) enter as viable alternatives or supplements [Meeker, 1985]. Suitability of these latter methods depends on inherent product robustness, technology maturity, and other considerations [Weibull, 2007].

In addition to demonstrating hardware and software robustness, a primary objective of space hardware environmental testing is to demonstrate ability to perform as required while operated on orbit, after experiencing the rigors of ground processing and transportation, launch, atmospheric flight, and orbital maneuvering and deployment (see Sec. 23.3). It is therefore desirable to subject units, subsystems, payloads, and vehicles to environmental levels, durations, and sequences similar to those expected for the mission. The intuitive construct of this approach is to *Test Like You Fly* (TLYF) [White and Wright, 2005] built on methods usually associated with system validation. Ground testing results are used to characterize, and confirm predictions for, such parameters as post-launch alignment and settling for line of sight (LOS), integrity of moving mechanical assemblies, rf and optical transmission, and thermal control and monitoring. While predictions and measurements are used for final vehicle verification and sell-off, characterization provides data to be used during on-orbit operations. For example, parameter variation under well-chosen temperature conditions will be used to calibrate data for each specific build of the vehicle and payloads during mission life. See the book website for a more extensive discussion of both environmental testing and qualification and acceptance testing, for space and ground systems.

A closely related objective is that of model validation, often key to end item verification, already discussed in Sec. 23.4.5. This includes the ubiquitous practice of thermal balance testing, a crucial phase of thermal vacuum test at the subsystem and element level. This is where final adjustments are made to account for end item thermal interfaces, view factors, gradients, power dissipation, heater control duty cycles, and current draw. Test cases are chosen to best utilize available test stimuli emulating on orbit conditions. Equipment limitations typically limit simulation of important on-orbit effects such as changes in beta angle of incident sunlight over time, largely due to facility constraints of cost, schedule, and physical realizability. For these reasons, it is essential that thermal engineers on the program are intimately involved in definition of test facility capabilities and test entry criteria, to assure that available methods will provide relevant data for extrapolation to such phenomena. Other aspects of space equipment model validation are also important: electro-optic path loss and noise signature, vibro-acoustically induced misalignment or vibration resonances. Each of these is a sophisticated discipline that requires collaboration between designers, analysts, facility personnel, and test engineers.

Defect Screening: Design vs. Workmanship

Typically, defects are categorized as either, inherent and repeatable due to flaws in design or build process; or, as specific to a particular build, indicating flaws in material or workmanship. A *design defect* is a flaw contained in the engineering data, rendering equipment built and tested to that data unsuitable for fielded use over mission life. Depending on severity of the defect, every such build is susceptible to premature failure during test or fielded use. The flaw is not necessarily a drawing error, and in fact often is not. Examples of design defect are: violation of safe operating area on a transistor; interface incompatibility such as *electromagnetic interference* (EMI) between adjacent equipment sets; insufficient radiator margin for expected degradation of thermal emissivity over life; inadequate stress margin on a structural element; stress induced in piece-parts due to poor lead forming process; excessive workmanship screen vibration levels resulting in breakage or unacceptable structural degradation; and other weaknesses in design or process likely to result in failure of an element or interface. As the name implies, a *workmanship defect* refers to a flaw introduced by faulty practice in the assembly, integration, handling, storage, or application of a product. Failure resulting from such defect is not always readily distinguishable from those reflecting poor design (see Sec. 24.1).

Defect screening is an expensive process. The most cost and time effective method for screening defects is to prevent them from occurring in the first place. A substantial body of literature and practice has arisen to describe methods of manufacture and material control that have proven effective in prevention and early correction of workmanship defects in space flight hardware. Section 23.5 summarizes principles and specifics of methods in common use. Defect discovery and correction at the assembly level is the final screen, the lower the level of assembly the better. Otherwise a defect lies latent while value-added steps are executed until discovery, at which point much of this work must be reversed and done again.

The type of defect to be screened depends on the types of assembly steps used to fabricate a particular end item. Judicious selection of environmental test regimen (or combinations thereof) has been shown effective in screening specific classes of defect. For example, a 1982 Navy study [NAVMAT, 1982] concluded that a combination of thermal cycling and random vibration is a highly effective method for exciting failure modes in electronic equipment, particularly solder joints. Hardware defect screening is a rich field of study and practice. Useful references have been included at the end of this chapter.

Defects in an end item deliverable (EID) eventually manifest themselves in outright failure of the affected element or interface. Robustness of equipment over mission life is frequently characterized by *failure rate*, *FR* (or λ), typically expressed as the reciprocal of *mean time between failure* (MTBF):

$$FR = \text{average failure count per unit time} = 1/\text{MTBF} \quad \text{(23web-1)}$$

A useful illustration of failure rate as a function of time across product life cycle, is the bath-tub curve, given general form in Fig. 23web-3. As experience consistently demonstrates, once a unit's workmanship defects have been eliminated during manufacture and acceptance testing, it enters the region of its useful operating life, during which failure rate achieves a low, roughly constant value. At the end of its useful operating life, failure rate increases as the fielded item begins to wear out.

Screening for workmanship is facilitated by environmental stress screening (ESS) of equipment assemblies for a sufficient time to provide reasonable assurance that those items which have survived have reached the point in their service lives that correspond to the flat section of the bathtub curve. The defects/failures are detected, assessed, and fixed (or screened), in some cases requiring an additional design iteration. Having passed such *acceptance tests*, these units are ready for fielded service. Elimination of design defects through ESS requires that a test persist throughout its required mission duration. This is the purview of *qualification testing*.

There are many important types of equipment for which it is necessary to conduct full life testing, often on multiple test articles, in order to obtain practical failure statistics and achieve final qualification. For items such as bearings, gyros, certain types of high voltage modules, etc., there is no credible substitute for a test that replicates the mission life profile. Such items need to be identified early in a program so as to complete a maximum amount of life testing on qualification articles, reducing risk to acceptable levels prior to committing fielded units to launch and flight service.

For a majority of complex assemblies this direct approach is neither practicable (since they are built late in the development cycle) nor desirable. Even for lower level assemblies, piece-parts, and materials there is a desire to retire risk associated with their use in build of the EID, and hence to obtain qualification and reliability data as early as possible. This is the purview of *accelerated life testing*, as a part of qualification. Exposure to

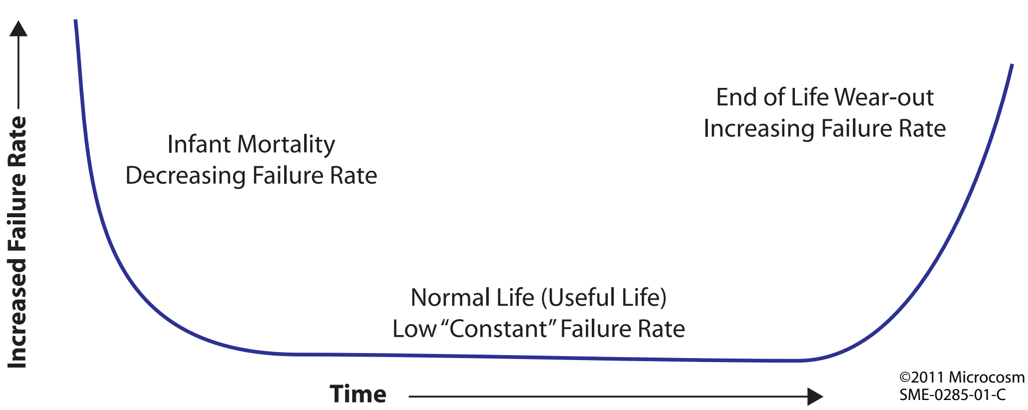

Fig. 23web-3. Bathtub Curve Failure Rate vs. Time for Equipment. Illustrates trend of failure rate over life cycle of fielded product, with higher rates early and late in life giving the "bath tub" shape.

increased environmental stress hastens failure mechanisms in a way that can be directly correlated to mission life in the predicted operational environment. A number of such accelerations are well understood, such as the following relationship based on the Arrhenius model for processes directly dependent on temperature:

$$T_O = \mathrm{AF}T_S \qquad \text{(23web-2)}$$

where T_S is time span of stress test, AF is the Arrhenius acceleration factor, and T_O is the resulting predicted operating time before failure based on that test. AF is a function of temperature and activation energy of the failure mechanism. This relationship is widely used to set conditions for tests qualifying materials and parts for adverse effects such as electromigration, corrosion, charge injection, crystalline growth, and gate-oxide defect.

Accelerated ESS is common in temperature cycling, random vibration, acoustics, and ionizing radiation. In the latter case, it is a simple process of increasing the dose rate to achieve total dose in a relatively short time. Not all environmental test acceleration factors can be so neatly stated as the Arrhenius relation. One such in common use [DoD, 1999] relates to temperature cycle count during unit thermal cycling or thermal vacuum test:

$$N_C = N_b\,(125/\Delta T_C)^{1.4} \qquad \text{(23web-3)}$$

where N_C is the number of complete temperature cycles, N_b is the basic number of cycles, and ΔT_C is the span between the minimum and maximum temperature in deg C. Here the story is more complicated since the most effective screening aspects of temperature cycling are: temperature and dwell time at hot and cold plateaus, revealing coefficient of thermal expansion (CTE) mismatches and growth phenomena; and temperature transition rate (should be a value between 1 C and 10 C per minute), which tends to provoke failure of faulty solder joints in electronics and similar failure mechanisms. Temperature cycle count scaling is valid only if these quantities are properly set. For electronic units, N_b is 8 for acceptance; 25, for qualification.

In all tests used to show design life, the intent is to sufficiently exercise and fatigue the *unit under test* (UUT) as to demonstrate ability to stand up to environments expected throughout its mission life, bringing it arbitrarily close to wear-out. The explicit objective of design *qualification testing* is to achieve a multiple of mission *fatigue equivalent duration* in order to prove robustness of UUT design. (MIL-STD-1540C recommends a multiple of four in defining thermal qualification.) For this reason, qualification units are not considered suitable for subsequent flight service.

23.4.5.2 Qualification and Acceptance at All Levels of Assembly

In certifying engineering data for space equipment, the test article must be built using the same drawings, software, specifications, procedures, tooling, test equipment, facilities, personnel training, and assembler skill levels as will be used for end item deliverable (EID) production. Flight qualification is a dominant aspect of design verification at all levels of assembly, representing the principal proof of performance and equipment robustness for space use. As described in Sec. 23.4.5.1 above, this is accomplished using accelerated environmental stress screening (ESS) techniques, involving exposure to environments exceeding those expected during flight.

Though similar to one another, distinct practices have evolved in segments of the space system development and acquisition community, around the selection and application of enhanced environments during qualification and acceptance of flight equipment and hardware. For DoD procurements, MIL-STD-1540 has been the standard of choice and is used to guide discussions for the remainder of this section. (Note: MIL-STD-1540 is being superseded by SMC Standard SMC-S-016.) NASA programs utilize standards specific to each center, which should be applied on relevant programs [GSFC, 2005]. Other standards have sprung up in commercial space venture practice, often embodied in documents produced by national and international engineering societies. Chiefly, the differences lie in the margins used to define environment intensity and duration. Though what follows is specific to MIL-STD-1540, the general principle applies.

Environmental test margins often drive design because allowed electrical and mechanical stress capability, or thermal management features, must accommodate them. This is a critical consideration, since robustness usually comes at the expense of weight and cost. Space launch is an expensive process and each kilogram is a precious commodity. On the other hand, the application of larger stresses can significantly accelerate test time, which saves cost.

The discussion of margin begins with uncertainty in *maximum predicted environment* (MPE). Experience has shown that flight environments often exceed even the most conservative predictions, usually due to resonance or coupling effects, and variations in thermal properties. A practice has grown up around the use of specific values of *uncertainty margin* to be used at various stages of program development. During test, uncertainty margin is directly summed with analytical predictions to yield MPE, to give the acceptance margin. Alternatively, uncertainty can be accounted for by using sophisticated statistical models. This method of deriving random vibration test amplitude spectrum from flight predictions is shown in Fig. 23web-4. Fatigue equivalent duration is then calculated by summing periods of greatest excitation for a given mission profile, as indicated by the annotated bars in Fig. 23web-5.

For qualification, further margins—known as design, acceleration, or qualification margin—are imposed above MPE in environmental intensity and duration, as illustrated for temperature limits in Fig. 23web-6. Table 23web-1 summarizes some important environmental test margins stipulated by MIL-STD-1540 for units and other complex assemblies, and how they vary by development maturity. These qualification and accep-

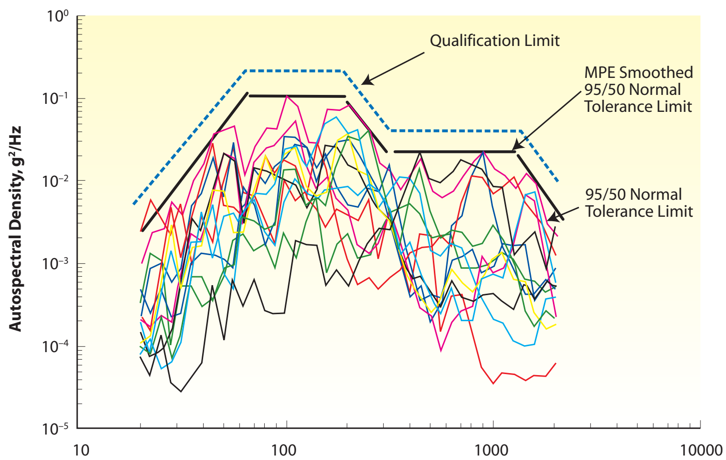

Fig. 23web-4. Typical Space Vehicle Unit Qualification and Acceptance Test Vibration Levels. The levels envelope those analytically predicted using spatial and time-averaging techniques (adapted from Fig. 6.2, NASA HDBK 7005). Here MPE is directly calculated, using statistical techniques instead of log margin standards, to determine acceptance level.

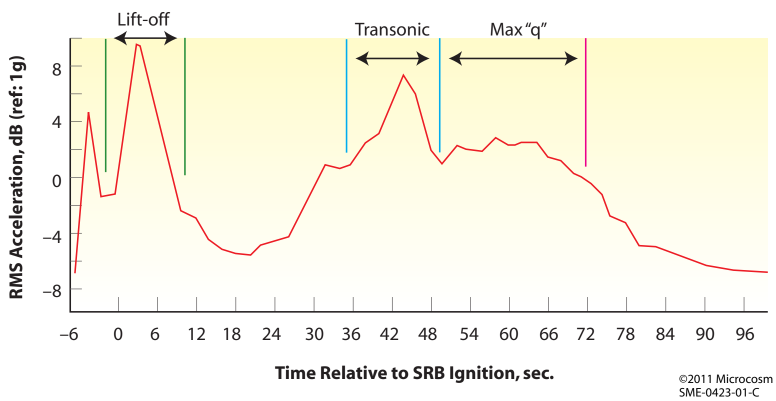

Fig. 23web-5. Time Plot for RMS Value of a Typical Vibration Measurement During Launch and Atmospheric Flight. This mission phase usually contributes the most stressing quasi-static and dynamic environments experienced by a space vehicle (adapted from Fig. 7.6, NASA HDBK 7005). Potential exceptions include missions that involve atmospheric re-entry, landing, or impact.

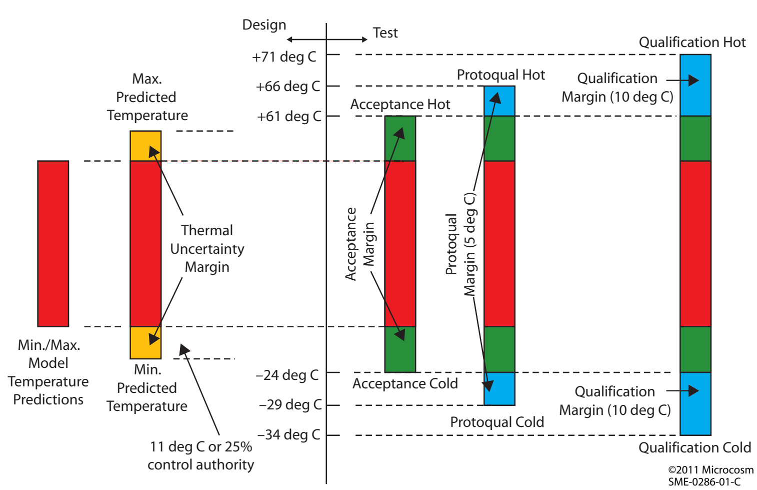

Fig. 23web-6. Thermal Design and Test Limits for Maximum Predicted Environment (MPE), Acceptance, and Qualification.

Table 23web-1. Environmental Test Margins for Electronic Units from MIL-STD-1540 and MIL-STD-1541. Uncertainty margins are greater during preliminary design phase, reflecting the lack of maturity in system design and consequently the engineering team's knowledge of component interaction and, boundary conditions. For example, EMI critical circuits use a total margin of 12 dB at preliminary design review (PDR)—summing preliminary design uncertainty margin with qualification margin—but preserve only 9 dB during production to account for variation in manufacturing tolerances.

Test Quantity	Uncertainty, Preliminary Design	Uncertainty, Acceptance Margin	Acceptance Duration	Qualification Intensity Margin	Qualification Duration
Random Vibration	6 dB rms	3 dB rms or statistical	1 min / axis	6 dB rms	Exposure duration ÷ 15
Thermal Cycling	11 C	11 C	17 cycles	10° C	53.5 cycles
Thermal Vacuum	18 C	11 C	8 cycles	10° C	25 cycles
Electromagnetic Interference	6 dB	3 dB	N/A	6 dB	N/A

tance limits represent the environments to which DoD space flight equipment is designed. Any factors of safety (see Sec. 22.1) required by the program design standards are over and above this set of quantities. Similar margins are imposed at the element and vehicle level of assembly.

The space system qualification and acceptance program starts at the lowest level of assembly: piece-parts and materials. Along with reliable production methods associated with defect prevention (Sec. 23.5), this is often considered the most important line of defense, since early detection avoids potential impacts to cost and schedule that accrue when discovered in later stages of test. Qualification and acceptance testing of spaceflight units and major subassemblies follows some subset of the tests listed in Table 23-10 titled "Environmental Tests Used for Spaceflight Hardware and Equipment," depending on assembly type —electronic, optical, rf, moving mechanical assembly.

For low-rate production space missions (some are virtually one-of-a-kind), a third alternative is often followed, that of proto-qualification. For such programs, it is not cost effective to build both a qualification unit and a small number of acceptance units for flight. In such cases, a flight unit is subjected to less strenuous environments than qualification, but greater than acceptance. This has the advantage of screening for workmanship defects as well as some portion of the design margin, while leaving sufficient useful life to justify commitment of the UUT to flight use. Note this implies that the design activity must still use qualification limits to define the design environments.

Qualification and acceptance testing at the element (bus, payload, or launch vehicle stage) and space vehicle level is required to ensure that interactions between lower level assemblies do not introduce defects that could not be experienced at lower levels of assembly. Here efforts to emulate mission environments are more complex and costly. The qualification test sequence more or less tracks the expected flight sequence:

- ***Temperature Cycling***—imposes predicted temperature extremes (with added margins for uncertainty and design robustness) under normal atmospheric pressure, preferably in the stowed configuration emulating boundary conditions of air flow and conduction expected in the launch vehicle fairing, as shown in Fig. 23web-4
- ***Vibro-Acoustics***—reproduces an amplified version (scaled by uncertainty and design margins) of stimuli predicted for launch pad acoustics and vibration, and vibration induced during atmospheric flight and engine operation—refer to Fig. 23web-4
- ***Pyro-Shock***—usually involves demonstration firing of on-board ordnance and non-explosive release mechanisms, along with confirmation of associated deployments, largely relying upon lower level assembly testing to qualify with margin for mechanical shock events associated with ignition, staging, separation, and deployment
- ***Thermal Vacuum***—approximates conditions of vacuum, view factor, and boundary conditions expected during on-orbit use, in all modes of operation, while imposing analytically predicted temperature extremes with additional margins in uncertainty (MPE) and design robustness
- ***Electromagnetic Compatibility***—checks for interference between subsystems and sub-elements, often utilizing special test circuits to confirm EMI safety margin; often includes radiated and conducted emissions; usually relies upon susceptibility data collected for lower level assemblies

A typical sequence is discussed in greater detail in Sec. 23.3.

Note the use of data from lower levels of assembly to complete the qualification story, providing another illustration of the roll-up of verification products as described in Fig. 23-9. Sometimes the arrow points the other way, completing qualification of a lower level assembly at the element or subsystem level. Such cases are the exception and should only be used when it is either impractical to create adequate test conditions at the lower level, or schedule pressures justify the risk of proceeding with unqualified hardware to further value-added processing and integration. This risk should not be minimized and is worthy of a formally tracked risk item at the program risk board.

The process for planning a spacecraft qualification program is given in Table 23web-2. A similar process is followed for planning acceptance testing for elements and space vehicles. Scaled-down versions of both should be applied to test planning for units and subsystems. Section 23.3 gives typical sequences and environmental test profiles for spaceflight units.

Table 23web-2. Steps in Architecting a Spacecraft or Element Qualification Test Program.

Step	Comments and Required Information	References
1. Identify Spacecraft and Payload Functions.	Test each spacecraft and payload function for proper operation. Identify the top functional requirement of the spacecraft in the top system specification and CONOPS, and glean subsystem functions from the subsystem specifications	Secs. 5.5, 6.4
2. Identify Environments.	Environments for transportation and storage, launch, and orbit including vibration, shock, rf, temperature, vacuum, and radiation	Chap. 7, 22 Sec. 26.3
3. Correlate Functions and Environments.	During transportation the spacecraft is off, although sensitive components may be powered. During launch, some equipment will be in standby and some will be operating. As a minimum, test the operating equipment during spacecraft vibration and check all modes of on-orbit operation	Sec. 5.5
4. Identify Main Configurations.	Include boost configuration and one or more orbital configurations. Assess needs for qualification data not obtained at lower levels of assembly.	Sec. 5.5, 6.5
5. Devise Functional Tests for Each Major Configuration.	Test each function appropriate to a particular configuration, including all equipment and software	Sec. 23.3
6. Lay Out the Sequence of Functional Tests and Environmental Exposures.	Sequence considerations/philosophies: • Test like you fly (TLF) • Optimize defect excitation and exposure • Order testing to retire greatest risks first	Secs. 23.4.5.1, 23.4.5.3
7. Identify Span Times and Special Requirements for Support Equipment and Facilities.	Create exit and pass/fail criteria for each environmental test regimen. Assess support equipment and facilities for measurement error, environmental control tolerances and limits, availability, consumables	Sec. 23.3, SMC-S-016

At the element and subsystem levels, this still leaves topics of flight and ground software qualification, and earth station qualification. The process of incremental software integration, test, and qualification is more fully addressed in Chaps. 20 and 28. Table 23web-3 summarizes test objectives for software verification and validation events culminating in Software Qualification Test (SQT). Like any other subsystem, flight software must be in its final flight configuration, fully operating during space vehicle and element qualification testing. In addition to the machine code loaded onto the host processor, this includes specifications, design documentation, *software development library* (SDL), software test reports at all levels of integration, compiler, emulator, archival protocols and resources, configuration management, k-parameters, upload equipment and procedures, maintenance manuals, and anything else that will be used to upload, checkout, maintain, update, or regression test the flight software over mission life. See [Halang, 1992], [MITRE, 1998], [Kaner, Falk, and Nguyen, 1993], [Santoni, 1997], and [Stankovic, 1992]. Exceptions to this regimen require potentially extensive regression testing and delta qualification.

To discuss earth station qualification is to address a broad array of facility types. The term, *earth station*, is used to refer to such a diversity of space mission assets such as control centers, fixed and mobile data relay terminals, user terminals, and *ground truth* calibration support sites. These are described in detail in Chaps. 15, 17, 28, and 29. The process of qualifying an earth station for use in the space system has a number of elements largely common across the spectrum, with unique requirements for each in performance, environment, and portability. Some of the more common parameters are given in Table 23web-4. Facilities and equipment comprising such stations undergo periodic overhaul and calibration.

Table 23web-3. Space System Software Verification and Validation Objectives.

- Software unit testing: lowest entity of software decomposition
  - Statement coverage: each statement executed at least once
  - Branch coverage: every possible outcome of each branch tested
  - Execution cycles and operation counts for real-time systems
- Integration testing: aggregation of software units and components
  - Test by logical grouping of units and components
  - For concurrent processing, exercise all concurrent task interfaces
- Includes real-time systems (i.e. embedded controllers)
  - Execution cycles while hosted on target processor/emulator
- Software configured item testing (a.k.a. SQT)
  - Functional testing
- Interfaces and constraints
- Event sequencing and operating modes
  - Load/stress testing (esp. real-time systems)
  - Performance testing
- Quantization error, variable type mis-match
- Convergence and numerical instability
- Response time for real-time systems (i.e. vehicle flight control)
- Data latency
  - Final bug detection and correction
- Variable type mis-match
- Address confliction: registers, flags, data, executables
- Data/bus collision
- Error detection and correction faults
- Overflows
- Watch-dog expirations and processor cycle hangs

When all elements have achieved qualification, there is still the question of verifying system level compliance to formally specified requirements. Much of this will be a matter of rolling up verification products from the ele-

Table 23web-4. Typical Earth Station Qualification Parameters. (Adapted from [Elbert, 2008])

- Radio systems
  - Antenna pattern, sidelobe levels
  - Cross-polarization isolation
  - EIRP and G/T
  - Carrier frequency and modulation bandwidth
  - Bit Error Rate (BER) performance
  - Channel amplitude response
  - Threshold performance
  - Frequency stability
  - Burst timing stability and channel group delay response (as applicable)
  - Anti-jamming (as applicable)
- Data systems
  - Land line data rates, BER, channel capacity, availability
  - Data storage and backup capacity and transfer rate
  - Data processing
- Power systems
  - Utility power phasing, peak and average current, volt-amp rating, line protection
  - Uninterruptable power system (UPS) response
  - Grounding and lightning protection
- Security measures
  - Physical security: doors, fences, alarms, posts, walls, ciphers and locks
  - Electromagnetic emissions (as applicable)
- Operator control
- Contact information and protocol
- Environmental controls (HVAC)
- Fault response and alarms (e.g. fire, power loss, computer failure, operator error)
- Site EMC survey and certification

ment level and lower, assuring satisfaction of allocation budgets and direct flowdowns. What remains is the purview of system simulation (discussed in Sec. 23.4.4) and *system integration test* (SIT). Because much of this latter activity is executed as a subset of space system validation, the details are discussed in (Sec. 23.4.5.3). As always, the principal distinction is between that of demonstrating compliance with requirements (verification) and of exercising the system through operational threads to show satisfaction of user needs (validation). For reasons of risk and affordability, most verification objectives are satisfied through ground-based testing, analysis, and simulation, prior to committing space assets to launch. On-orbit verification is expensive and usually too late to prevent loss of major mission objectives and timelines, in the event of failure.

23.4.5.3 Space System Validation and Final Certification

System validation is the process for assuring that the system will provide needed services in the mission operational environment. System validation involves the exercise of simulators, emulators, and development hardware and software, using mission operators, infrastructure, and procedures, through progressive levels of integration and fidelity as the program matures. Activities focus on exposing system elements and subsystems to interfaces (including users and operators), data interchange, available heritage system elements, and event sequences representative of the operational mission and environment. Detailed mission scenarios (or, threads) from system concept of operations (CONOPS) are enacted to capture incompatibilities that may have escaped the requirement derivation process. System validation is distinct from both system verification and from model validation to be described later in this section. It is also distinct from the process of requirement validation (Sec. 23.6), though this process is often incorporated as an early part of the system validation plan. This is the final line of defense against previously undiscovered mission hazards related to sneak paths, software bugs, and missed hand-offs.

A commonly used method for planning validation testing follows a simple mantra: *First Day, Best Day, Worst Day, and Last Day*. Each scenario represents a standard mission operational thread or set of threads, with actual commands and messaging, operator displays, communications links and relays, preferably at projected durations. First Day threads emulate launch day operations, and post-launch deployment and initialization for the spacecraft. This is almost always the most hazardous part of the mission, with success hinging on real-time decisions by ground crews, as well as flight and ground processors. It is essential that fault response scenarios be enacted multiple times in the months leading up to this mission phase (also falls under category of Worst Day).

Best Day scenarios, more often referred to as *day-in-the-life* (or, DITL, pronounced like fiddle), represent the operations for which the mission was desired in the first place. For many missions, this is simply continuous operation, with occasional interruptions for calibration or maintenance activities, such as orbital stationkeeping. More complex missions involve intricate autonomous or operator-directed orbital rendezvous and docking, real-time tasking, planetary encounter, surface roving, sampling and other in situ measurement. In such cases, it is crucial that potentially debilitating phenomena are accounted for, such as celestial body eclipse, communication lag, Doppler effects, solar activity disruptions, and other inherent aspects of the mission. A subset of these scenarios is typically run with on-orbit assets after launch, as a part of final *system certification* for operational use.

Equally critical is the exercise of anomalous scenarios (Worst Day). Here the system equipment and operators conduct normal operations which are interrupted by the occurrence of a fault. This fault can be a hardware failure in space or on the ground, single event effect (SEE), or even natural or man-made catastrophe. The aggregate space system must demonstrate ability to respond to and correct the fault, and return to normal (or pre-defined degraded) operations.

Finally, all space missions are required to provide the means for retiring space and ground assets, as part of mission termination (Chap. 30). For space assets in particular, this is critical to preserve availability of orbits and orbital stations that are free of debris or other hazards. Interplanetary missions are required to assure elimination of materials that could harm as-yet-undetected life forms. Exercise of realistic operational scenarios must demonstrate ability to de-orbit or otherwise safe the asset in question, even in the presence of faults. In some cases, mission planners desire the capability to go into dormant or follow-on mission bridging states, and this also will need validation.