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Evaluation Mechanism for Spacecraft Pre-Launch Debris-Generation Explosion Risk Assessment: Updated 2022

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Abstract

This report discusses the methodology developed by The Aerospace Corporation to estimate the probability of an on-orbit debris generating accidental explosion for a spacecraft or launch vehicle upper-stage to demonstrate compliance with The United States Government Orbital Debris Mitigation Standard Practices and US Air Force Instruction (AFI) 91-202 The Us Air Force Mishap Prevention Program. The current assessment methodology generalizes lessons learned in past evaluations to define a general procedure that is available to evaluate future missions. This methodology covers all phases of operation of the spacecraft, including deployment, mission, disposal maneuvers, passivation, and end-of-life (disposal orbit). In general, the subsystems determined to be a source of risk for debris generation are those that contain stored energy. The stored-energy subsystems on a spacecraft may include the propulsion subsystem (liquid apogee engines, propellant and inert gas tanks, Hall current thrusters, and/or like propulsion devices), the reaction wheel assemblies, the pyro firing devices, flight termination system, the electrical power subsystem (batteries), and the thermal control subsystem. The goal is to quantify the risk of a debris-generating explosion during the spacecraft's entire orbital lifetime. This general methodology should aid in future evaluations of the explosion risk of an individual spacecraft or launch vehicle upper-stage.

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1. Introduction

1.1 Background

As space activity has increased, so has the amount of space debris in orbit. Space debris comes from many sources [1]. One source of space debris is vehicle fragmentations. From 1961 until 2007 there were 240 debris-producing fragmentation events [2]. Debris from these fragmentation events now accounts for 45% of the catalogued objects still in orbit [2]. The catalogued objects are generally greater than 10 cm in diameter. The high debris levels increase the chance of collisions between debris and active spacecraft. This risk has been realized as collisions between space debris and spacecraft have been previously observed [3].

Many of the fragmentation events were the result of explosions [4]−[6]; the most serious of these events were from nine Delta second stages. The cause of these break-ups is believed to be the failure of a common bulkhead between the residual fuel and oxidizer. To prevent the problem on subsequent missions, the contractor instituted new procedures; the second stage was maneuvered away from the payload, the engine was fired to depletion, and the tank valves were left open to vent any residual propellant [5]. The venting of propellant tanks has become a standard practice for some vehicles. To vent a pressurized subsystem, the subsystem is permanently opened to the vacuum of space. However, most spacecraft are not designed to vent because premature venting is a potential mission failure mechanism, and reliable latch valves and zero-momentum vent ports add to mission costs. Instead of venting, many vehicles and spacecraft blow-down propellant tanks at end-of-life (EOL) rather than vent them. A blow-down of a pressure subsystem temporarily opens a subsystem until only the unusable volume of the vented gas or liquid remains. For a propulsion subsystem, this generally means running a thruster until no measurable thrust is achieved. Unlike venting, blowing down a subsystem cannot guarantee a subsystem is totally empty of pressurant. Other pressurized subsystems, such as heat pipes and batteries, are generally not designed to be blown down or vented in this manner. The risks associated with either of these procedures will be assessed in this evaluation.

The United States Government Orbital Debris Mitigation Standard Practices (USGODMSP) [7][8] specifies that in developing the design of a spacecraft or upper stage, each program should demonstrate, via commonly accepted engineering and probability assessment methods, that the integrated probability of debris-generating explosions for all credible failure modes of each spacecraft and upper stage (excluding small particle impacts) is less than 0.001 (1 in 1,000) during deployment and mission operations. Other requirements such as those that flow from USGODMSP include; Department of Defense Instruction (DoDI) 3100.12, *Space Support* [9], Space and Missile Systems Center Instruction 62-110 [10], US Air Force Instruction (AFI) 91-202 Space Safety and Mishap Prevention Program [11], that requires the integrated probability of explosion for all failure modes of each spacecraft and launch vehicle to be less than 0.001 (excluding collisions) and NASA-STD-8719.14A, *Process for Limiting Orbital Debris* [12], section 4.4.1.

The formal process of failure mode analysis is covered by MIL-STD-882E [13][14]. Several previous efforts [15]–[26] at The Aerospace Corporation have focused on evaluating current spacecraft designs and operational procedures by using this method to understand how well these programs conform to the guidelines on debris-generation mitigation. The ideal scenario would be for the spacecraft contract to include a debris-generation risk assessment as part of the subsystem safety and reliability deliverables that the contractor is required to provide. However, many vehicle contracts predate space debris- mitigation efforts, and thus these programs need independent analysis, based on best available information, to determine the risk of a space debris-generation explosion from these vehicles as a function of the mission. Initial analysis efforts focused on the end-of-life phase of orbital life since it is the longest phase in the life cycle of most orbital objects. Later, evaluation efforts examined vehicles more extensively to evaluate the risk of explosion from the deployment phase to beyond end of functioning life a.k.a. EOL. These evaluations also factored in reliability of the involved subsystems to understand the risk of debris-generating explosions if the vehicle's mission had ended prematurely during different phases of operation and not as planned.

The purpose of this report is to consolidate previous explosion studies for SMC spacecraft into one general explosion risk calculation methodology to be used for evaluation of future spacecraft. This use of the methodology provides a means for future missions to establish compliance status with the AFI 91-202 requirement of section 10.8.3.4.

The causal factors under consideration will include the probability of a catastrophic and critical failure occurring during each on-orbit phase of operation that has the potential to lead to explosion. The probability of explosion will be examined for all phases of the spacecraft mission, including deployment, mission, disposal maneuvers, passivation, and EOL (disposal orbit). EOL vehicle subsystems containing stored energy are to be the focus for evaluation of this risk for debris generation. These subsystems include the following spacecraft subsystems: propulsion subsystem, reaction wheel assembly, flight termination system, electrical power subsystem, thermal control subsystem, and all pyro firing devices. Additional subsystems may need to be added in cases of special mission risk due to unusual payload. This study assumes the vehicle is built per design and reliability as stated by the manufacturer and that this information is readily available for review and evaluation from either the customer or the contractor. When this information is not available, best assumptions to estimate these values are discussed.

1.2 Scope of Generated Report

The generated report provides analyses of probability of accidental explosion for all relevant failure modes of the spacecraft and provides the integrated probability for accidental explosion of the spacecraft. This probability can be used by program manager to evaluate compliance with AFI91-202, sections 10.8.3.4.1. for the probability limit (excluding collisions) of less than 0.001 (one-thousand in one-million).

The scope covers the spacecraft from the separation of the last launch vehicle component through EOL disposal and 200-years of disposal life (AFI91-202, section 10.8.3.4.1.).

The scope does not include risk response, risk monitoring, risk control, orbit mechanics, intentional breakups, collision, operational debris release, launch vehicle components, ground support equipment or manned spacecraft.

The scope of this report is currently limited to an EOL duration of 200 years for several reasons, the main one being the uncertainty in the estimates involved in the current analysis. Humans began launching spacecraft roughly seventy years ago, thus the 200 years projection is a least three times what any orbital spacecraft has endured to date. There are many sources of uncertainty. The orbit of the spacecraft cannot be projected accurately for 200 years, and it will likely drift before this time is complete. Space debris has increased dramatically in the last seventy years and is likely to change significantly in the next 200 years. The ongoing process of climate change is likely to affect satellites in low earth orbit, as it is known to affect the height and density of the upper atmosphere. This will likely change drag and affect lifetimes of satellites beyond the scope of this document. Many space programs are also discussing active debris removal and deorbit projects; it is entirely plausible that spacecraft in EOL may be deorbited within the 200 years analyzed. Given all the uncertainties affecting the spacecraft and the space environment, a projected explosion risk for an EOL of greater than 200 years does not have much meaning. The prudent course would be to reevaluate the explosion risk at the end of 200 years, or after any major on-orbit debris generating accidental explosion failure.

The current methodology finds that the accidental explosion risk generally decreases as a spacecraft proceeds through its phase of operation. Current ODMSP requires all spacecraft to be passivate before entering EOL, but does not ask for a risk evaluation for accidental explosion after this passivation. However, the risk of explosion of the vehicle or spacecraft at extreme long lifetimes that will be experienced for many EOL spacecraft pushes this spacecraft beyond common experience. It is possible explosion risk increases again at these very long lifetimes. Thus, the information needed to understand this potential risk due to the remaining stored energy on-board even after passivation is still to be evaluated.

1.3 Information and Data Needs

The following information and data must be obtained for optimal evaluation of the spacecraft or upper-stage. This information may be made available to The Aerospace Corporation from the customer or contractor. In the absence of specific information the evaluation will have to rely on estimates, modeling or test campaigns.

1.3.1 Phases of Operation

A description of spacecraft deployment, mission, and EOL plan including passivation, disposal maneuvers and reentry or EOL re-orbit is required. This description should include phase duration and altitude/orbit information. For the purpose of this evaluation, operation begins when the spacecraft separates from the final launch vehicle stage.

1.3.2 Subsystems Description

A description of all potential sources of stored energy on the spacecraft must be included, e.g., propulsion subsystem(s), reaction wheel assemblies, electrical power subsystem, heat pipes, flight termination system, and pyro firing devices. The description of each subsystem should include the chemical identity of all gases, liquids, reactive solids and mixtures associated with that subsystem e.g. xenon, helium, monomethyl-hydrazine, ammonia, mixed oxides of nitrogen. The description should include physical and chemical properties of any new or proprietary substance or mixture. The qualification burst (qual-burst) pressure, design-burst pressure, proof pressure, maximum allowable working pressure (MAWP), volume, amount and type of material stored and operational pressures of all tanks on the spacecraft that will contain stored gases or liquids are needed.

1.3.3 Propellant Usage Estimate

A table of the fuel aboard the spacecraft and the standard deviation of the fuel used during each phase of operation, including an estimate of trapped propellant after blow-down at EOL is needed. For best results in evaluating explosion concerns for each phase of operation, an independent evaluation by the contractor of subsystem reliability and fuel usage estimates as a function of the spacecraft's phase of operation should be used. This information is generally available from the contractor as part of the contract but should be included as a deliverable on all future spacecraft contracts. The contractor may not deliver all the data needed for each phase of operation. This, in general, may not matter so long as the highest risk phase, highest propellant load, is obtained for evaluation since the highest risk often occurs in a propellant tank when there is the lowest ullage space. Once any propellant has been consumed the risk typically goes down considerably because of the subsequent increase in ullage space within the propellant tank. However, it is important to remember that reliability also factors into risk evaluation and this typically decreases with time. Thus, to best understand risk throughout the lifetime of a spacecraft a full set of propellant usage estimates at each phase of operation is preferred when it can be obtained.

1.3.4 Reliability

For best results in evaluation, the spacecraft contractor's assessment of subsystem reliability as a function of the spacecraft's phase of operation should be used. This information is generally available from the spacecraft customer and should be a required deliverable on future contracts. If a subsystem reliability isn't known the total reliability of the spacecraft will be used as a conservative estimate.

1.3.5 Thermal Model

The spacecraft contractor's thermal model should be used if available. When not available section 1.4 will discuss alternative estimates.

1.3.6 Other Relevant Information, New Design, Modifications

Is there anything else that the contractor or customer has identified as relevant for review? Does the spacecraft have a new design, design change, materials change in any subsystem including propulsion, reaction wheels, pyrovalve or electric power system that would require a re-evaluation of these subsystems or other sources of stored gases or liquids at pressure? If such information becomes available after an evaluation has been completed the evaluation should be updated.

1.4 Calculated Thermal Model

When the spacecraft contractor's thermal model is not available, a nominal temperature value of 272 K with $\sigma T = \pm 11$ K has been used previously as an estimate based on analysis of several current spacecraft. [15]–[26]. This estimate is based on a worst case EOL heating scenario for components within the spacecraft. The spacecraft was assumed to be oriented to receive direct sun in a 78° orbit (no eclipse) and the Winter Solstice solar flux was used to provide maximum temperatures.

- Simple Model – Spherical Object
  - Absorbed power from Sun = 1361 $W/m^2 * pR^2$
  - Emitted power = $sT^4 * 4pR^2$ (Stefan-Boltzman Law)
  - $s = 5.67e\text{-}8\ W/m^2\,K^4$
  - Emitted power = Absorbed Power
    - T= 278 K
  - Model is for skin temperature of a good conductor
    - For a tank internal to a spacecraft with insulation the temperature will be lower
    - Model assumes black body with absorptivity/emissivity of 1; lowers numbers could lower temperature
    - Experience with other spacecraft indicates a temperature of ~272 +/- 11 K

A consideration must be made when using the above thermal model of spacecraft for upper stages where components are not inside a thermal blanket. In addition, satellites during EOL also have passive thermal control, i.e., heat pipes, to transfer thermal energy within the spacecraft, but the upper-stages lacks passive thermal control. The thermal model for an uncontrolled satellite determined the skin temperature of the satellite could reach temperatures of 348 K. A thermal model by McCall et al. predicted the surface temperature of 10 cm blocks in various orbits. The worst-case scenarios predicted temperatures that fluctuate during their annual orbit between 280 K and 320 K [27]. Another report from 1982 Gumpel et al. [28] did a thermal model of Delta A upper stages in sun synchronous disposal orbit. They determined a maximum of ~400 K could be reach for short time periods at worst case conditions, i.e., 100% sun exposure to the upper stage broadside to the sun. The predicted 400 K maximum temperature is significantly higher than others reported in the literature. These short periods of high temperature could be a potential hazard to common wall tanks and propellants with low auto ignition temperatures.

2. Phases of Operation

A spacecraft or upper-stage is designed to operate for a designed lifetime as designated by the mission profile. Regardless of the type of spacecraft, the lifetime is broken down into several general spacecraft operational phases. The phases include deployment from the launch vehicle, mission operations, disposal operations (maneuvers and passivation) after completion or termination of the mission, and EOL. Individual spacecraft however may have additional steps such as passivation or several disposal maneuvers prior to entering into EOL. Disposal operations may include de-orbit or moving the spacecraft to a disposal orbit depending on the original orbit of the spacecraft. This report is broken into phases of operation because each phase of operation has varied amounts of stored energy that must be considered individually. Also, each phase of operations has varied subsystem reliability concerns. In general, subsystem reliability decreases as the spacecraft subsystems' age increases while in orbit.

2.1 Deployment

Spacecraft life begins with the deployment phase. Spacecraft are transported into their final orbit or, a transfer orbit, by the upper stage of a launch vehicle. The time from separation from the upper stage of the launch vehicle until the vehicle reaches final station orbit is called the deployment phase. Spacecraft that do not separate from the launch vehicle do not achieve orbit. The deployment phase is generally a relatively short time compared to active mission life. Deployment phase is the phase most likely to use bipropellant hypergolic thrusters to orbit raise and to circularize the orbit. Deployment is the phase when the spacecraft will likely start using devices such as batteries, reaction wheels, and pyrovalves. The pyrovalves, if a vehicle has them, are likely to be fired to open or close propellant lines.

In general, since the spacecraft is at the beginning of active life, reliability of subsystems is highest. During this phase, the spacecraft's thermal profile and attitude are actively controlled. This mission control means that deviations in temperature, and hence the probability of an overheating event leading to an accidental explosion, will be smaller than for an uncontrolled object.

Upper stages are typically deployed from lower stages while still in suborbital space. However, unique upper stages such as cube satellite space ferries or taxis, may deploy from a launch vehicle upper stage and then begin deploying other spacecrafts. Stages that finish their lifetime while still in suborbital space cannot create space debris from accidental explosion and thus the risk is eliminated for them.

2.2 Mission

The mission phase of the spacecraft is expected to last throughout its designed mission profile. Based on operational performance, a mission may be extended beyond the designed operational life. In general, however, the report should assume the designed mission profile will be followed with designated uncertainties. These uncertainties are usually included as part of a contractor's reliability and fuel usage estimates. During the mission the spacecraft is involved in active station keeping of its orbit. This phase is likely to use chemical monopropellant thrusters or xenon ion thrusters. During the mission phase the spacecraft's thermal profile and attitude are actively controlled.

A typical upper-stage, has several controlled burns of its rocket motor which occur relatively quickly after deployment. Thus, the mission phase is typically much shorter than a spacecraft.

2.3 Disposal Maneuvers

Following mission completion or termination, the spacecraft or upper-stage is re-orbited either to directly reentered into the Earth's atmosphere, maneuvered into an orbit in which atmospheric drag will reenter the subsystem within 25 years, or maneuvered into a disposal orbit that is shown to avoid interference with other operational spacecraft. The designed disposal maneuvers are dependent on the spacecraft's initial lifetime profile and heavily dependent on its initial orbit. For spacecraft in low Earth orbit (LEO), the disposal maneuver will often be to de-orbit and allow it to re-enter into the Earth's atmosphere within a short time from end of mission life. For Geostationary orbits (GEO) and other high altitude orbits the propellant requirements to de-orbit the spacecraft makes de-orbit often not feasible. Thus, these spacecrafts are often boosted into a higher orbit that is called the disposal orbit. This disposal orbit is designed to be sufficiently higher than GEO so that more than 100 years is required before the disposal orbit will decay into the operational band of GEO orbiting spacecraft. The disposal operations phase often uses the spacecraft thrusters if a vehicle has them. During the disposal operations phase the spacecraft's thermal profile and attitude are actively controlled.

2.4 Passivation

Passivation is the phase of operation in which the spacecraft or upper-stage is prepared for the EOL phase. All on-board sources of stored energy of a spacecraft or upper stage should be depleted or safed when they are no longer required for mission operations or post-mission disposal. Depletion should occur as soon as such an operation does not pose an unacceptable risk to the payload. Propellant depletion burns and compressed gas releases should be designed to minimize the probability of subsequent accidental collision and to minimize the impact of a subsequent accidental explosion. Passivation typically involves depleting the tanks of propellant and other stored fluids if possible. Usually a spacecraft or upper-stage burns to completion or blows down propellant tanks. This leaves a residual amount of pressurant and propellant onboard after passivation. Electrical passivation is achieved by turning off the payload and telemetry, stopping all equipment in motion (flywheels, momentum wheels, gyroscopes, earth sensors, and solar panels), safing flight termination systems, discharging the batteries and disconnecting their charge network. The goal of this phase is to minimize the risk of post mission break-up resulting from stored energy. After this phase of operation, the spacecraft will no longer be able to actively control its thermal profile and attitude.

Since the goal of passivation is to leave the spacecraft in a state with a lower probability to generating space debris, these things should be considered as part of passivation;

1. Does the propellant being blown-down have the physical properties such that it will readily sublimate in space? Most standard propellants are unlikely to survive more than 25 years in orbit [30]. However, low vapor pressure ionic liquids and dissolved solids can survive for these time periods [30]–[32]. Burning these propellants to completion is a better alternative than a strict blow-down.

2. Propellant release doesn't leave the spacecraft in a flat-spin from which centripetal forces damage the EOL spacecraft.

3. Since the reliability of systems in EOL are known, care should be taken to avoid leaving the spacecraft in a Lazarus mode where the spacecraft power systems can recharge bring the vehicle back to life if it tumbles into direct sunlight. A disposal plan that requires a sub-system to remain functioning at EOL to prevent mishap is not truly entered an EOL state.

2.5 End-of-Life

The EOL of a spacecraft nominally begins when the mission lifetime has ended, the disposal procedures have been completed successfully and the subsystems have been adequately passivated. Once entering

EOL the spacecraft will be an uncontrolled space object. There will be no active temperature control or attitude control, and no communication with the spacecraft.

There is always a chance that an anomaly could cause the spacecraft to enter into EOL prior to mission completion. Irrespective to the cause, once a spacecraft has become uncontrollable and useful operation of the spacecraft is no longer possible, the spacecraft has essentially entered EOL. This means that an evaluation of explosion risk needs to assume that during each phase of operation the mission may suddenly come to an unexpected end and the spacecraft may enter EOL not in its designed fashion. To establish the risk contributed by these scenarios the probability of explosion due to the spacecraft not being able to finish its phase of operation and the likelihood of the event, established with subsystem reliability data, must be taken into account.

2.5.1 Orbit Lifetime

For spacecraft that are actively de-orbited, the EOL may be relatively short before the spacecraft reenters and thus the orbital lifetime estimate will be used to determine the amount of time the spacecraft will spend in orbit after mission completion or termination. For spacecraft or upper-stages with lifetime in disposal orbits > 25 years, a 200 year assessment for accidental explosion is conducted.

3. Failure Mode Analysis

Each spacecraft subsystem that potentially contains stored energy during any phase of operation of the spacecraft or upper-stage must be evaluated during each of the different phases of operation for explosion risk. Thus, the design of the subsystems must be known to understand whether the spacecraft has the following devices; propulsion subsystem(s), reaction wheel assemblies, electric power subsystem, flight termination system, heat pipes, and pyro firing devices. Some spacecraft may have more than one of each type or possibly a unique source of stored energy. The report should address each subsystem's failure modes as a function of the phase of operation of the spacecraft separately.

MIL-STD-882E [13][14] is the formal department of defense document used to evaluate and conduct the failure mode analysis. The probability levels of accidental failure for the mode analyzed are addressed in Table 1. For compliance with accidental explosion risk a mode's risk must be remote. The severity of the accidental explosion is listed on Table 2. Although, compliance doesn't rely on severity. This is due to the nature of accidental explosions. The accidental explosion of an EOL asset or a small cube satellite has relatively little monetary value. However, that explosion event may through subsequent collisions with its debris jeopardize a much more valuable asset later on.

Table 1. Probability Levels

Description	Level	Specific Individual Item	Fleet or Inventory	Probability
Frequent	**A**	Likely to occur often in the life of an item	Continuously experienced.	Probability of occurrence greater than or equal to 10^{-1}.
Probable	**B**	Will occur several times in the life of an item	Will occur frequently.	Probability of occurrence less than 10^{-1} but greater than or equal to 10^{-2}.
Occasional	**C**	Likely to occur sometime in the life of an item	Will occur several times.	Probability of occurrence less than 10^{-2} but greater than or equal to 10^{-3}.
Remote	**D**	Unlikely, but possible to occur in the life of an item	Unlikely but can reasonably be expected to occur.	Probability of occurrence less than 10^{-3} but greater than or equal to 10^{-5}.
Improbable	**E**	So unlikely, it can be assumed occurrence may not be experienced in the life of an item	Unlikely to occur, but possible.	Probability of occurrence less than 10^{-5}.
Eliminated	**F**	Incapable of occurrence. This level is used when potential hazards are identified and later eliminated.	Incapable of occurrence. This level is used when potential hazards are identified and later eliminated	

Table 2. Severity Categories

SEVERITY CATEGORIES		
Description	**Severity Category**	**Vehicle Mishap Result Criteria**
Catastrophic	1	Could result in one or more of the following: death, permanent total disability, irreversible significant environmental impact, or monetary loss equal to or exceeding $10M.
Critical	2	Could result in one or more of the following: permanent partial disability, injuries or occupational illness that may result in hospitalization of at least three personnel, reversible significant environmental impact, or monetary loss equal to or exceeding $1M but less than $10M.
Marginal	3	Could result in one or more of the following: injury or occupational illness resulting in one or more lost work day(s), reversible moderate environmental impact, or monetary loss equal to or exceeding $100K but less than $1M
Negligible	4	Could result in one or more of the following: injury or occupational illness not resulting in a lost work day, minimal environmental impact, or monetary loss less than $100K

3.1 Propulsion Subsystem

The propulsion subsystem consists of engines e.g. Liquid Apogee Engine (LAE), Rocket Engine Assemblies (REAs), Hall current thrusters (HCTs), gridded ion engines, as well as tankage, e.g., the fuel tanks, the oxidizer tanks, the Xenon tank, and pressurant tanks.

Although many spacecraft may use common buses and common propellants, there is typically less design agreement from program to program as to the propellant subsystem's operational propellants, volume of propellants, and quantity of propellant as function of phase of operation than for any other subsystem on the spacecraft. Thus, the propellant subsystem is the most likely subsystem for which detailed information from the contractor or customer is needed to complete the evaluation of explosion risk.

The explosion risk analysis for the propulsion subsystem should be conducted for the deployment, mission, disposal maneuvers and passivation phases, and an explosion risk analysis for the thrusters must also be conducted. The analysis of the risk of temperature excursion causing a propellant tank overpressure should be conducted for a full tank. Statements of thermal runaway, decomposition and freeze-thaw cycles may also be included in this section depending on the propellants used by the spacecraft.

The main explosion risk for tanks is a temperature excursion that increases the pressure beyond the capability of the tank. The tanks on a spacecraft in a disposal orbit have a typical predicted temperature mean and standard deviation (σ). An estimate of this can be obtained from the thermal model analysis of the spacecraft.

SMC STD-S-005 [33] establishes the baseline requirements for the design, fabrication, assembly, installation, test, inspection, operation, and maintenance of pressure systems used in spacecraft and launch vehicles. Each tank has several limiting pressures, which are characterized by the difference between internal and external pressure.

- The qual-burst pressure is the pressure difference applied to a qualification tank causing it to actually burst.
- The design burst pressure is the applied pressure difference above which analysis shows the tank may burst.
- The proof pressure, defined by SMC STD-S-005 section 4.1.1.8, is the applied pressure difference the tank will sustain without distortion, damage, leakage, or loss of functionality.
- Maximum allowable working pressure (MAWP) is the maximum pressure difference that is permissible in the vessel under normal operating conditions. Typically organizations outside of The Aerospace Corporation use the term MEOP for the Maximum Expected Operating Pressure instead of MAWP.

The design burst pressure for the propulsion subsystem typically includes a factor of safety of 1.5 times MAWP. The proof pressure generally includes a factor of safety of 1.25 times MAWP. Other subsystems (thermal control subsystem, electric power subsystem) generally have higher factors of safety. For the tanks in the propulsion subsystem the limiting pressures in descending order are the qual-burst, design burst, proof pressure, and MAWP. These pressures are different for each spacecraft and values from the contractor should be used to produce the most accurate evaluation. The values desired for these tanks are given in Table 3.

Table 3. The MAWP, Proof and Burst pressures for Fuel, Oxidizer, Pressurant Tank, and Heat Pipes

Tank (psia)	MAWP (psia)	Proof (pisa)	Burst (psia)
Fuel	x	x	x
Oxidizer	x	x	x
Pressurant	x	x	x
Heat Pipe	x	x	x

3.1.1 Tanks Storing Gaseous Compounds

The main contributor to the explosion potential of the spacecraft is an overpressure of a pressure storage vessel. The pressure in the tanks is generally dictated by the temperature of the storage vessel, the volume of the storage vessel and the amount of material in the storage vessel. For the gas-filled tanks, tanks with inert gases such as helium, nitrogen or xenon, or nearly empty propellant tanks near EOL, the pressure is driven by the gas law, given in Eq. (1), where P is the pressure, n is the number of moles of propellant in the gas phase, T is the temperature and V is the volume of the tank.

$$P = \frac{nRT}{V} \tag{1}$$

For slightly more accuracy, the van der Waals equation can be used instead at the high pressures inside the xenon and helium tanks. This is needed as pressurized gases tend to be at very high initial pressures in propellant tanks for in-space usage.

$$\left(P + \frac{an^2}{V^2}\right) * (V - nb) = nRT \quad (2)$$

where a and b are constants that depend on the gas. For helium, a and b are 0.03412 L^2-atm/mol^2 and 0.02370 L/mol. For xenon, a and b are 4.19442 L^2-atm/mol^2 and 0.05105 L/mol. For Krypton, a and b are 2.325 bar L^2 mol^{-2} and 0.0396 L mol^{-1}.

The actual amount of material in the tank at a given time in the mission profile has an uncertainty from propellant usage, and this information is needed from the contractor to complete the evaluation. The uncertainty of the volume of the tank includes the uncertainty due to the volume of lines attached to the tank as well as a minor amount of thermal expansion or contraction. A conservative volume uncertainty of 0.3 % has previously been used by Aerospace to account for volume uncertainty. The uncertainty in the amount in the tank is generally provided by the contractor. For uncorrelated terms, the cumulative error can be determined by Eq. (3).

$$\frac{\sigma^2(p)}{P^2} = \frac{\sigma^2(T)}{T^2} + \frac{\sigma^2(V)}{V^2} + \frac{\sigma^2(n)}{n^2} \quad (3)$$

The Central Limit Theorem [32] says that the distribution of an average of many independent, identically distributed random variables tends toward the famous bell-shaped normal distribution with a probability density function as given in Eq. (4).

$$Probability = \frac{1}{\sigma * \sqrt{2\pi}} e^{\frac{(-(P(T)-P)^2)}{2\sigma^2}} \quad (4)$$

The worst-case scenario is the maximum amount of propellant in the tank including uncertainty and the minimum tank size including uncertainty. An increase in pressure can come from an increase in the number of moles of gas, a decrease in the volume of the tank, or an increase in the temperature. The tank volume is fixed at manufacturing, and if there is no supply of additional gas on orbit, then temperature increases are the only viable failure mode for these subsystems. Typically, the error (fluctuations in) associated with temperature at EOL is the biggest driving factor for failure. Error analysis is driven by uncertainty in the EOL temperature and often uses the worst-case number of moles and several standard deviations of temperature to analyze risk.

For each gas storage tank on the spacecraft a table should be constructed (example in Table 4) as part of the evaluation that shows the temperatures required to reach the limiting pressures in the spacecraft tanks and the associated probabilities that those tanks reach those temperatures (an example is shown in Table 3). The temperature is calculated using Eq. (1). The probability for the standard deviations from nominal temperature is given by the normal distribution divided by two since deviation of temperature on the low side cannot cause overpressure of the tank.

The standard deviation will depend on the spacecraft and the phase of operation. During the active mission, the spacecraft's thermal controls are operational; during EOL the spacecraft's thermal controls are no longer active and the value in section 1.3.5 or 1.4 is used.

Table 4. Probability of Temperature Reaching a Critical Pressure in a Given Spacecraft Tank Used to Store a Gaseous Component

Phase of operation	Pressure (psi)	Description of pressure	Temperature required to reach pressure (K)	Std Deviations from nominal operating temperature	Probability	Total Reliability	Total probability
Deployment	x	Design Burst	x	x	x	x	x
Deployment	x	Proof	x	x	x	x	x
Deployment	x	MAWP	x	x	x	x	x
Mission	x	Design Burst	x	x	x	x	x
Mission	x	Proof	x	x	x	x	x
Mission	x	MAWP	x	x	x	x	x
Disposal maneuvers	x	Design Burst	x	x	x	x	x
Disposal maneuvers	x	Proof	x	x	x	x	x
Disposal maneuvers	x	MAWP	x	x	x	x	x
Passivation	x	Design Burst	x	x	x	x	x
Passivation	x	Proof	x	x	x	x	x
Passivation	x	MAWP	x	x	x	x	x
EOL	x	Design Burst	x	x	x	x	x
EOL	x	Proof	x	x	x	x	x
EOL	x	MAWP	x	x	x	x	x

3.1.2 Tanks Storing Liquid Compounds

The main risk for liquid tanks is similar to that of gaseous tanks: an overpressure of a pressure storage vessel on the spacecraft. The pressure in these tanks is generally dictated by the temperature of the storage vessel, the volume of the storage vessel and the amount of material in the storage vessel. The propellant tank pressure is controlled by the vapor pressure of the liquid in the tank. The vapor pressure (P) of the liquid can be predicted at a given temperature (T). Equations for selected propellants are shown below. For hydrazine tanks [35][36]:

$$P(atm) = e^{(58.7582 \; - \; \frac{0.707\times10^4}{\mathrm{T}} - 7.088\ln(\mathrm{T}) + 0.457\times10^{-2} * \mathrm{T})} \tag{5}$$

For monomethyl hydrazine (MMH) tanks [35]

$$P(Torr) = 10^{(7.11158 - \frac{1.1045711\times10^3}{\mathrm{T}} - \frac{1.522277\times10^5}{\mathrm{T}^2})} \tag{6}$$

For unsymmetric dimethyl hydrazine (UDMH) tanks [35]

$$P(Torr) = 10^{(6.73578 - \frac{6.5510134\times10^3}{\mathrm{T}} - \frac{4.247009\times10^5}{\mathrm{T}^2})} \tag{7}$$

In the deployment phase xenon is pressurized into the liquid state. The vapor pressure of xenon is given by the following equation [37].

$$P(10^{-5}Pa) = 10^{(\frac{-8485.83984}{\mathrm{T}} + 269.6183136 - 3.94126562 * \mathrm{T} + 0.03332691*\mathrm{T}^2 - 0.000175307022*\mathrm{T}^3 + 0.000000586334808*\mathrm{T}^4 - 0.000000001215310816*\mathrm{T}^5 + 1.424073676\mathrm{E}{-12}*\mathrm{T}^6 - 7.2009907\mathrm{E}{-16}*\mathrm{T}^7)} \tag{8}$$

The critical temperature of xenon is 290 K and the critical pressure is 5,840 kPa [37]. At temperatures above the critical temperature the xenon will either behave as a supercritical liquid or gas depending on the pressure in the tank. For the colder range of the EOL temperature, xenon will follow this equation. However, the proof and burst pressures can only be obtained above this temperature where xenon will act as a supercritical fluid, and Eq. (2) must be considered for evaluation purposes.

In order to burst the tank the mass of the xenon will need to be heated to the critical point and then vaporized. The heat required for vaporization of xenon is 12.636 kJ mol^{-1} [37]. The total energy available to vaporize the liquid xenon has previously been determined as part of an earlier evaluation. The energy needed would be determined by the following equation:

$$energy\ (kJ) = heat\ of\ vaporization\ xenon * mass\ xenon\ available * \frac{mole\ xenon}{131.293\ \mathrm{g}} \tag{9}$$

where the mass of xenon available will depend on the mission and the phase of operation.

The most common oxidizer in spacecraft is nitrogen tetroxide (NTO). The vapor pressure of nitrogen tetroxide can be determined by the following equation [38];

$$P(Torr) = 10^{(-81.086+0.81558*\mathrm{T}-0.0027*\mathrm{T}^2+3.05\mathrm{E}{-06}*\mathrm{T}^3)} \tag{10}$$

NTO gas is in equilibrium with the NO_2 gas. The vaporization and dissociation of oxidizer into nitrogen dioxide gas is very endothermic, requiring more heat input than the other propellants (86.69 kJ/mole). The nitrogen tetroxide critical temperature is 431.2 K.

The oxidizer has a higher vapor pressure than the fuel and a lower temperature is required to reach design burst pressure in general for the oxidizer tank, the actual heat input required to achieve the burst temperature is much higher for the oxidizer. This is once again because the process of vaporizing the oxidizer absorbs a significant amount of heat [39].

$$N_2O_{4(L)} \xrightarrow{28.66\,kJ/mol} N_2O_{4(g)} \tag{11}$$

$$N_2O_{4(g)} \xrightarrow{58.03\,kJ/mol} 2NO_{2(g)} \tag{12}$$

$$NO_{2(g)} \xrightarrow{57.1\,kJ/mol} NO_{(g)} + \frac{1}{2}O_{2(g)} \tag{13}$$

Reactions 11 and 12 are necessary to reach the tabulated vapor pressure of the oxidizer; vaporization of the remaining oxidizer would require the input of a large amount of energy that can be determined from the equations above and the amount of propellant remaining. The energy (E) needed can be determined by the following equation:

$$E\ (kJ) = heat\ of\ vap + dis\ N_2O_4 * mass\ N_2O_4\ available * \frac{mole\ N_2O_4}{92.011\ \mathrm{g}} \tag{14}$$

Where the heat of vaporization and dissociation is 86.69 kJ/mol and the mass of N_2O_4 available will depend on the mission and the phase of operation. The amount of dissociation of the gaseous N_2O_4 will depend on the equilibrium constant.

$$K_c(RT)^{\Delta n} = K_p = e^{-58.03\frac{kJ}{mol}+T*0.176\frac{kJ}{mol}}/RT \tag{15}$$

The uncertainty of the pressure in these subsystems is generally dependent on the temperature of the tank. During the operational lifetime the temperature is controlled, the spacecraft thermal controls are operational, and the standard deviation is relatively small. At EOL the temperature is no longer controlled, thermal controls are no longer active, and the temperature has greater uncertainty. Section 1.3.5 or 1.4 can be used to determine EOL temperature.

Another potential satellite propellant that is not used much at the moment is hydrogen peroxide. Hydrogen peroxide can be used as both a monopropellant, or as an oxidizer for a fuel. The vapor pressure of hydrogen peroxide depends on the concentration of the peroxide used [40]–[42].

Additionally, ionic liquids are starting to be used for in-space propulsion purposes. In theory ionic liquids vapor pressure is so low that for AFM-315E tanks the vapor pressure is going to be close to the mole fraction of the water concentration [43][44].

$$P(bar) = 10^{(}6.20963 - \frac{2354.731}{\mathrm{T}+7.559}) * (mole\ fraction\ water)\ ) \tag{16}$$

For LMP-103S tanks the vapor pressure is shown in Table 5 [46].

Table 5. The Vapor Pressure of LMP-103S at a Function of Temperature

Temperature (K)	Pressure (bar)
298.15	0.125
325.15	0.5
345.15	1
358.15	1.5
368.15	2
375.15	2.5
383.15	3
393.15	4

Another alternative propellant in development is iodine in electric propulsion thrusters. Iodine tanks the vapor pressure is given by [46];

$$P(bar) = 10^{(3.36429 - \frac{1039.159}{T-146.589})} \quad (17)$$

Another alternative electric propulsion propellant under consideration in krypton. In the deployment phase krypton is pressurized into the liquid state. The vapor pressure of krypton is given by the following equation [46].

$$P(bar) = 10^{(4.2064 - \frac{539.004}{T+8.855})} \quad (18)$$

The critical temperature of krypton is 209.46 K and the critical pressure is 55.2019 bar [46]. At temperatures above the critical temperature the krypton will either behave as a supercritical liquid or gas depending on the pressure in the tank. For the colder range of the EOL temperature, krypton will follow this equation. However, the proof and burst pressures can only be obtained above this temperature where krypton will act as a supercritical fluid, and Eq. (2) must be considered for evaluation purposes.

The upper-stages typically use cryogenic fuels such as liquid hydrogen and oxygen. The vapor pressure of these fuels when they warm up near 298 K will be vastly higher than the tank burst pressure. Thus, cryogenic tanks always contain latch valves that actuate and release excess pressure. So long as the latch valves are in working condition, the tanks cannot over pressure due to temperature rise. The reliability of the latch valves is thus critical to tank integrity.

Some upper-stages also use kerosene. The vapor pressure of kerosene can be determined as follows [46];

$$P(Pascals) = 10^{(3.185 - \frac{61770.125}{T-52793})} \quad (19)$$

As with the gas tanks, the liquid tanks should have a table documenting the calculated probability that the temperature during each phase of operation reaches pressures of concern. Table 4 shows an example. The temperature is calculated using the equations in this section. The probability is based on standard probabilities of temperature.

No upper-stages use methane as of yet. The vapor pressure equation for methane is [47];

$$P(bar) = 10^{\left(4.22061 - \frac{516.689}{T+11.223}\right)} \quad (20)$$

The critical pressure for methane is 1190.6 K and critical pressure is 46.1 bar [46].

Table 6. Probability of Temperature Reaching a Critical Pressure in a Given Spacecraft Tank Used to Store a Liquid Component.

Phase of operation	Pressure (psi)	Description of pressure	Temperature required to reach pressure (K)	Std Deviations from nominal operating temperature	Probability	Total Reliability	Total probability
Deployment	x	Design Burst	x	x	x	x	x
Deployment	x	Proof	x	x	x	x	x
Deployment	x	MAWP	x	x	x	x	x
Mission	x	Design Burst	x	x	x	x	x
Mission	x	Proof	x	x	x	x	x
Mission	x	MAWP	x	x	x	x	x
Disposal maneuvers	x	Design Burst	x	x	x	x	x
Disposal maneuvers	x	Proof	x	x	x	x	x
Disposal maneuvers	x	MAWP	x	x	x	x	x
Passivation	x	Design Burst	x	x	x	x	x
Passivation	x	Proof	x	x	x	x	x
Passivation	x	MAWP	x	x	x	x	x
EOL	x	Design Burst	x	x	x	x	x
EOL	x	Proof	x	x	x	x	x
EOL	x	MAWP	x	x	x	x	x

3.1.3 Pressure Regulators

One way to increase the number of moles (get more gas) into a propellant tank while in space is due to failure of another subsystem. One of the most intense in-space explosions of a U.S. propellant tank was due to a faulty regulator valve that allowed high pressure (more gas) to enter into a storage tank from a high pressure pressurant tank. This explosion was due to a regulator failure that allowed the design burst pressure to be exceeded. This design is no longer flown and the evaluator should verify that similar subsystems, if they exist on the spacecraft, have mitigating valve designs to prevent such occurrences. Most spacecraft use a redundant passive pressure regulator system. The regulator is use early in mission (when reliability is high) and then sealed off from the propellant tanks. Thus, eliminating the failure mode for the rest of the spacecraft's life. However, some spacecraft used active valving to regulate pressure drops between pressurant tanks and propellant tanks (or similar low pressure tolerant tubing) or leave the double redundant passive regulator active throughout mission. In such case a failure of the regulator which would case an over pressure must be evaluated. Table 7 shows an example.

Table 7. Probability of Helium Regulator Failure Causing an Overpressure of Propellant Tanks

Mission	Phase of Operation	Pressure (psi)	Description of Pressure	Resulting Pressure If Regulators Failed @ 298 K (psi)	Probability of Pressure Exceedance Due to EOL Temperature	Total Reliability of One Regulator	Total Probability of Exceeding Pressure
x	Deployment	x	Design burst	x	x	x	x
x	Deployment	x	Proof	x	x	x	x
x	Deployment	x	MAWP	x	x	x	x
x	Mission	x	Design burst	x	x	x	x
x	Mission	x	Proof	x	x	x	x
x	Mission	x	MAWP	x	x	x	x
x	Disposal maneuvers	x	Design burst	x	x	x	x
x	Disposal maneuvers	x	Proof	x	x	x	x
x	Disposal maneuvers	x	MAWP	x	x	x	x
x	Passivation	x	Design burst	x	x	x	x
x	Passivation	x	Proof	x	x	x	x
x	Passivation	x	MAWP	x	x	x	x
x	EOL	x	Design burst	x	x	x	x
x	EOL	x	Proof	x	x	x	x
x	EOL	x	MAWP	x	x	x	x

3.1.4 Decomposition

Hydrazine propellant has an additional consideration because it has a natural decay rate of 0.015 % per year at 310 K in storage [35][47]. The decomposition products are mainly nitrogen and ammonia. Over time these decomposition products can build up in the tank and cause the tank pressure to increase. The additional amount of pressure in a propellant tank that has hydrazine decomposition occurring is to be tabulated as part of the evaluation when hydrazine propellant is present. The decay rate is slightly temperature dependent and the decomposition rate is adjusted for temperature by the following equation:

$$decay\ rate(\frac{\%}{yr}) = 0.015\frac{\%}{yr}e^{(\frac{1}{316K}-\frac{1}{T})*66.3\frac{kJ}{mol}}/R \tag{21}$$

Liquid hydrazine decomposes into mostly gaseous products; hydrogen, nitrogen and ammonia via two separate reaction channels.

$$3\,N_2H_4 \rightarrow 4\,NH_3 + N_2 \tag{22}$$

$$N_2H_4 \rightarrow 2\,H_2 + N_2 \tag{23}$$

Reaction 17 is thought to be the dominant reaction pathway near room temperature. The ammonia (NH_3) and nitrogen (N_2) produced does not immediately go into the gas volume (particularly in a propellant tank with a roll down bladder) as both gases have a natural solubility in the liquid hydrazine. The solubility constant for ammonia dissolved in hydrazine is calculated from the following [48];

$$K_s = e^{-(-5204\frac{cal}{mol}+22.53\frac{cal}{mol\,K}*T)/(1.987\frac{cal}{K\,mol}*T)} \tag{24}$$

The solubility constant for nitrogen dissolved in hydrazine is calculated from the following [49];

$$K_s = 10^{(2.5322-516K/T)*0.000001))} \tag{25}$$

The moles dissolved will then be determined by:

$$moles\ dissolved = K_s * moles\ of\ hydrazine(L) * tankpressure\ (atm) \tag{26}$$

If the amount of ammonia and nitrogen exceeds the amount of substance that can be dissolved the remaining gas will be available to increase the pressure in the tank. The pressure rise can be determined by:

$$\Delta P = \frac{moles\ gas\ not\ dissolved}{total\ volume-volume\ hydrazine(L)} * T * 0.08206\frac{L\ atm}{K\ mol} * 14.7\frac{psi}{atm} * 1000) \tag{27}$$

The uncertainty in the amount of hydrazine that decomposes will depend on the uncertainty of the temperature. The probability that the tank overpressures due to hydrazine decomposition is thus dependent on this temperature uncertainty. The standard deviation of the temperature will depend on the spacecraft and the phase of operation. During the active mission the spacecraft's thermal controls are operational; during EOL, the spacecraft's thermal controls are no longer active. This is the worst-case scenario. The EOL thermal model is covered in section 1.3.5 or 1.4. For missions that have hydrazine tanks, the liquid tanks should have a table that shows the probability that the pressure exceeds design proof pressure for EOL starting during each phase of operation. The following table (Table 8) shows, for each phase of operation, the pressure rise at up to 200 years EOL at the temperatures needed to reach proof pressure, probability of reaching proof pressure is also indicated for each phase. In general, the probability is usually highest during the deployment as none of the propellant has been consumed by operations. There is no phase studied before deployment.

Table 8. Probability of Reaching Proof Pressure Due to Hydrazine Decomposition for 200 Years

Mission	Phase	Temperature Needed to Reach Proof Pressure K	Standard Deviation from EOL Temperature	Probability of Reaching Temperature That Will Reach Proof Pressure	Reliability of Applicable Subsystems or Total Bus	Probability Total
x	Deployment	x	x	x	x	x
x	Mission	x	x	x	x	x
x	Disposal maneuvers	x	x	x	x	x
x	Passivation	x	x	x	x	x
x	End of life	x	x	x	x	x

Thermal runaway, the process wherein decomposition is accelerated by increased temperature, in turn releasing more energy that further increases temperature, is evaluated but not considered a credible source of explosion. The heat released by a full hydrazine tank decomposing at this rate is less than 0.003 Watts. This is an insignificant heat source compared to other thermal loads on the spacecraft, e.g. thousands of Watts of solar radiation.

Monomethyl-hydrazine (MMH) has a natural decay rate of only around 10 % of hydrazine at 473 K [35]. The pressure rise due to decomposition of MMH is determined similarly to hydrazine except that MMH decomposition products are significantly different from hydrazine. MMH mainly decomposes into nitrogen, methane, methylamine and ammonia. The products will likely dissolve to some extent at high pressure. The solubility of these gaseous products is not well understood and a conservative estimate of low solubility (similar to nitrogen in hydrazine) should be used. This estimate is extremely conservative since nitrogen gas is very likely to have lower solubility than these hydrocarbon-based products. For hydrazine 1/3 of the moles produced by decomposition are nitrogen-like gaseous product, for MMH it is assumed 1 moles decomposes to produce 1.23 moles of these type of products. The decay rate is slightly temperature dependent and the decomposition rate is adjusted for temperature by the following equation:

$$decay\ rate\left(\frac{\%}{yr}\right) = 0.0015\,\frac{\%}{yr}\,e^{\left(\frac{1}{473K}-\frac{1}{T}\right)*66.3\,\frac{kJ}{mol}/R} \tag{28}$$

The determination and evaluation table are similar to the procedure used for hydrazine above (Table 8).

The slow decomposition rate of other propellants is possible. Hydrogen Peroxide is known to decompose and produce oxygen gas. The decomposition rate is highly dependent on the pressure vessel material. Likely hydrogen peroxide tanks will need a pressure relief valve for long term storage in space. In any event, the observed pressure rise and decay rate should be measured and known for the flight tank for tanks without relief valves.

Similarly, LMP-103S has shown stable pressure over storage times of eight months to five years. Longer tests simulating 200 years have not been conducted. Since LMP-103S can thermally breakdown into known gasses such as N_2O, NO_2 etc these gasses could potentially build up in a long disposal orbit and rupture the propellant tank. No data is available to evaluate this risk for long disposal. Similar data is also missing for AFM-315E.

3.1.5 Freezing of Propellant

For certain propellants a line rupture is theoretically possible in a propellant line if freezing occurs. For instance, hydrazine, if stored below the freezing point of 275 K [50], may freeze. If freezing occurs, the hydrazine volume will decrease because the hydrazine will shrink. A line rupture can then occur during thaw cycle if the line increases in temperature above 275 K as liquid fills the volume behind the frozen hydrazine and is trapped. Freeze/thaw cycles can (and have in the past) led to line rupture when thermal control is eliminated [35]. However, passivation is conducted in the disposal orbit and non-explosive fragmentations such as a line rupture have always generated less than ten fragments [51]. Remaining tubing and components would still be attached to the vehicle. Under normal operation phase the thermal control subsystem keeps the propellant tank more than 5 standard deviations above the freezing temperature. Thus, the risk of propellant tank rupture due to freezing is less than one in a million during deployment, mission, disposal maneuvers and passivation. Most other common in-space propellants freeze at vastly lower temperatures than hydrazine and thus their risk is similar.

At EOL, the predicted propellant tank temperature is lower than the freezing point of hydrazine. A standard deviation of 11 K would mean the hydrazine potentially could freeze and thaw in a line and cause a rupture. Since the rupture would occur after passivation the rupture would occur with the tank at the vapor pressure of the hydrazine. The vapor pressure at the freezing point is 0.05 psi, unlikely to result in an explosion. If the spacecraft failed prior to normal EOL with high pressure in the propellant tank, it is possible a larger pressure release event could occur due to a freeze/thaw rupture. The probability of a spacecraft failure in an earlier operational phase is based on the reliability of the spacecraft and the possibility the spacecraft fails in a way propellant tanks cannot be vented. Even without venting, propellant has already been shown to be unlikely to overpressure. Therefore, a freeze/thaw rupture is not equivalent to an overpressure explosion and would not result in an explosion risk.

3.1.6 Explosion in the Thrusters

The probability of explosion in the thrusters needs to be considered. Different types of thrusters may be used during several different phases. For example, in the deployment and disposal maneuvers phases, the monopropellant and bipropellant thrusters are likely to be used. During the mission phase the station keeping thrusters such as the electric propulsion thrusters and monopropellant thrusters are more likely to be used.

Chemical explosions are known to occur in bipropellant engines such as the LAE, and also for monopropellant thrusters such as the REAs, but are impossible for the HCTs. Bipropellant thrusters, which run on hydrazine and NTO, ignite hypergolically. The fuel and oxidizer ignite on contact without the need for a separate igniter. Occasionally during startup, or short pulse operation, explosions of accumulated hydrazine vapor can occur. In some cases, these explosions can damage the thruster, or lead to propellant leaks. The Aerospace Corporation has investigated several such anomalies; but there are no examples of such anomalous explosions leading to spacecraft fragmentation or debris generation.

Monopropellants such as the REAs can also have explosions. These are believed to be caused by voids in the catalyst bed. Voids can come from errors in the initial packing, or from the degradation of the catalyst through operation. Liquid hydrazine can accumulate in these voids and ignite suddenly when heated sufficiently. These explosions can be damaging to the catalyst bed and to the engine. There is no evidence, however, of debris-generating explosions occurring in REAs.

An anomalous deployment does not necessarily increase the risk. For example, the loss of a single thruster may make the deployment phase longer but does not change the magnitude of the temperature excursion necessary to cause an explosion or increase the probability of a thruster explosion leading to a debris causing event.

The reliability of the thruster is generally provided by the customer or the contractor for the evaluation. Once again thruster failure does not assume a debris-generating failure. As stated above there is no example of thruster failure on-orbit of this nature.

3.1.7 Autoignition of Propellant

The autoignition temperature of a substance is the temperature at which it will spontaneously ignite in a normal atmosphere without an external source of ignition. For most substances used as propellants and inert gases a greatly elevated temperature beyond nominal operating temperature of the spacecraft is required to reach autoignition temperature. Table 9 has autoignition temperatures for common substances found in spacecraft subsystems. Not all of these substances necessarily will be found on a specific spacecraft and if other potentially explosive substances are present then they must be assessed. The probability is dependent on the EOL temperature as explained in section 1.3.5 and section 1.4.

Table 9. The Autoignition Temperature of Selected Substances

Substance	Autoignition temperature (K)	Probability
Hydrogen [46]	809	X
Hydrazine [35]	543	X
MMH [35]	423	X
UDMH [35]	499	X
Ammonia [46]	903	X
Ethane [46]	745	X
N_2O [46]	598	X
LMP-103 [45]	458	X
AFM-315E [43]	413	X
Methane [46]	830	X
Kerosene [46]	483	X

The exact autoignition temperature often varies with purity, pressure and the type of material in contact with the substance. Also, most measurements of autoignition are done either in oxygen or air. The presents of oxygen and air in general lowers the autoignition temperature of most substances, but on spacecraft they're unlikely to be in contact with either air or oxygen while stored on-orbit. Most tank materials were previously tested with autoignitable liquids such as hydrazine and air and showed no ignition at 403 K [35].

3.1.8 Propellant Usage During Operational Phases

The most important information needed to begin an evaluation of the propulsion subsystem is a contractor- or customer-provided table of the fuel aboard the spacecraft with the standard deviation of the fuel used during each stage, including an estimate of trapped propellant after blow-down at EOL.
Table 10 shows an example of a propellant usage table.

Table 10. Propellant Usage Table

Phase of operation	Mean Fuel (lbm)	σ Fuel (lbm)	Mean Oxidizer (lbm)	σ Fuel (lbm)
Deployment	x	x	x	x
Mission	x	x	x	x
Disposal maneuvers	x	x	x	x
Passivation	x	x	x	x
EOL	x	x	x	x

In addition, for systems with pressurant tanks, the tank pressure at the beginning of each phase is needed. Table 11 shows and example of a table of pressurant quantities.

Table 11. Quantities of Pressurant at Beginning of Phase

Phase	Pressurant 1 (psi)	Pressurant 2 (psi)
Deployment	x	x
Mission	x	x
Disposal maneuvers	x	x
Passivation	x	x
End of Life	x	x

3.1.8.1 Deployment

The one difference between deployment and EOL that tends to increase risk in deployment is that the propellant and pressurant tanks are full in deployment, whereas they are empty or nearly empty at EOL. Based on the design requirements, the design burst is at least 50% higher in pressure than the MAWP. The MAWP is expected to occur in deployment when the tanks are full. Sections 3.1.1-section 3.1.3 handle how to determine probability of failure due to propellant tanks during this phase of operation and the following phases as well.

3.1.8.2 Mission

The mission phase can be significantly different than the deployment phase for the propulsion subsystem. Typically, station keeping uses different engines on the spacecraft than the deployment phase. Often a significant percentage of the propellant on board has been used during the deployment stage. Given that significantly less fuel is on-board the explosion risk due to overpressure is typically lower after deployment. The main exception is ion thrusters that typically do not begin usage until the mission phase, although they can be used during the deployment stage. During mission phase the spacecraft is thermally controlled.

3.1.8.3 Disposal Maneuvers

Disposal maneuvers generally have the final long duration firing of the thrusters on board. The propellant on board will continue to decrease relatively rapidly during this phase and will be less than either the mission or deployment phase. During disposal maneuvers phase the spacecraft is thermally controlled.

3.1.8.4 Passivation

During the passivation phase the vehicle is prepared for EOL. The remaining fuel if any is typically vented in a blow-down operation. The thrusters are fired until no thrust is observed. This will reduce the propellant to the unusable amounts, the amounts stuck to walls of the propellant tank or in non-accessible transfer lines. Typically, after passivation, Eqs. (1)–(4) are the only evaluation needed to determine probability of explosion. The exception in hydrazine tanks with residual hydrazine because of hydrazine decomposition. No thrusters will be able to run after this stage. The only risk in passivation stage is that the release of residual propellant could lower the disposal orbit if not done properly, but this risk is manageable.

3.1.8.5 End-of-Life

Temperature excursions during EOL are larger than during any other phase because there is no active thermal control, and no attitude control, but the average temperature is lower because the heaters are not active. During a nominal mission the spacecraft enters EOL with only the unusable amount of propellant in the tanks. Equations (1)–(4) can be used to determine the risk of overpressure. The propellant tanks may undergo freeze-thaw cycles during EOL because they are not thermally controlled.

3.2 Reaction Wheel Assemblies

The reaction wheel assemblies (RWAs) spin at a high rate of speed and are used for attitude control during the mission phase. The most likely failure event for an RWA is one in which the ball bearings fail, and the reaction wheel simply freezes up, but doesn't release from its mount. In one case, a CMG (Control Moment Gyro) rotor dislodged during a high-speed ground test. No such events have occurred on orbit [52]. Even if an on-orbit failure of this nature did occur it is not likely a large amount of explosion–like debris would be generated from such an event. The probability of explosion in reaction wheel assemblies does not vary with mission life; therefore, reliability data is not relative to risk of explosion.

3.2.1 Deployment

The RWAs are not activated during deployment. Attitude control is maintained with the REAs.

3.2.2 Mission

The RWAs are operating normally all through the mission. Occasional failures have been observed in which a reaction wheel seizes up and stops spinning. No debris-causing events have been observed on orbit.

3.2.3 Disposal Maneuvers

The RWAs are operating normally all through the mission. Occasional failures have been observed in which a reaction wheel seizes up and stops spinning. No debris-causing events have been observed on orbit.

3.2.4 Passivation

In order to limit the risk to other space subsystems from accidental break-ups after the completion of mission operations, all on-board sources of stored energy of a space subsystem such as flywheels and momentum wheels, should be depleted or safed when they are no longer required for mission operations or post-mission disposal. Depletion should occur as soon as this operation does not pose an unacceptable risk to the payload. Mitigation measures should be carefully designed not to create other risks. The evaluation should consider if these subsystems are safed during the passivation phase. During passivation when the spacecraft powers down the RWAs are typically designed to come to rest quickly. Once at rest there is no stored energy in the subsystem.

3.2.5 End-of-Life

The RWAs are typically at rest during EOL with no possible failure modes since typically spacecraft are not powered at EOL.

3.3 Pyro Firing Devices

Pyrovalves are stored energetic devices on the spacecraft that contain a pyrotechnic device for single use opening and closing of valves in propellant lines. Pyrovalves are designed to contain the explosion of the charges and for this reason there is very little danger of structural damage to the spacecraft. Since they are designed to have contained explosions within the propellant management system the likelihood that a pyrovalve would have a debris-generating explosion is very small. Ignition transients in a pyrovalve can lead to propellant leaks, and in one case did lead to mixing of oxidizer and fuel. The valve design which led to that failure is no longer in use [53]. Pyrovalve failure modes can result in external leaks of propulsion system fluids, but the standard design cannot credibly lead to a debris forming explosion. This kind of explosion has not been observed in over 60 years of space activity [54], so the probability of a pyrovalve-initiated debris-causing explosion is certainly less than that for a propulsion system explosion, 1×10^{-6}, and probably much smaller. The most likely failure mode of a pyrovalve is accidental leakage. Accidental leakage is not a debris forming event. The probability of explosion in pyrovalves does not vary with mission life; therefore, reliability data is not relevant to the risk of explosion. If a new pyrovalve is placed into service that does not have flight heritage, this new design will need to be re-evaluated for explosion risk potential.

3.3.1 Deployment

In deployment, pyrovalves may be used to open the flow of propellants to the thrusters. The pyrovalve in its unfired state provides protection from the unintended release of hazardous chemicals on the launch pad. On bipropellant subsystems, after the deployment is finished another pyrovalve is typically fired to isolate the thruster from the propellant subsystem. Pyrovalves may also be fired to separate the regulated pressurant subsystem from the propellant tanks. After these pyrovalves fire, the propellant tanks run off the pressurized helium that is already in the tank ullage. The ullage is most of the tank volume at this point.

The failure mode for pyrovalves is that they may not fire. To protect against this, current pyrovalves have redundant actuators. If the pyrovalve fails to open the propellant lines, the spacecraft will be unable to deploy. If the pyrovalve fails to isolate the thruster at the end of deployment, there are possible propellant leak paths that could leave the spacecraft with insufficient propellant to complete its mission. The primary reason that Normally Open pyrovalves are used upstream of the propellant tanks is to ensure fuel and oxidizer vapors don't mix over the course of the mission. The mixing of the propellant vapors could lead to a sudden explosion whereas helium leaking into the propellant tanks would be a slow process. If the pyrovalves fail to isolate the regulated pressurant system, the propellant tanks could be over pressurized. In one known case, a helium regulator failed in end of life, opening the helium to the propellant tanks. The helium is stored in the pressurant tanks at much higher pressure than the pressure for which the propellant tanks are rated. This caused the propellant tanks to fail, releasing debris. The satellite in question had no pyrovalves to isolate the regulated pressurant system. There are no known cases of pressurant tanks equipped with pyrovalves overpressurizing the propellant tanks. However, the most likely failure mode of a pyrovalve is accidental leakage.

3.3.2 Mission

During the mission the pyrovalves have typically already fired.

3.3.3 Disposal Maneuvers

During disposal maneuvers the pyrovalves have typically already fired.

3.3.4 Passivation

No pyrovalves are typically active during the passivation phase. Unused pyrotechnic charges which are designed to activate a subsystem, but which are not capable of causing vehicle fragmentation, need not be fired during passivation [55]. This is because if these charges were accidentally fired in the future EOL, the result would not be the generation of orbital debris.

3.3.5 End-of-Life

No pyrovalves are active during the EOL phase. If an unused charge were to accidentally fire during EOL, the result would not be the generation of orbital debris [55].

3.4 Electric Power Subsystem

The Electrical Power Subsystem (EPS) contains one potential source of stored energy, the batteries. There are several different types of batteries that could potentially be used on a spacecraft. Previously spacecraft use nickel hydrogen (NiH_2) batteries for the majority of Class A missions. However, battery technology is undergoing rapid advancement due to a variety of energy storage needs. These newer batteries are now finding their way into spacecraft. Rechargeable lithium ion (Li-ion) batteries have to be considered given their increasing use in future vehicles. Single use, or primary, chemical batteries (such as silver-zinc, primary lithium metal, zinc-carbon, alkaline, and others) may also be used for specific missions. Due to the increase in capability of rechargeable batteries coupled with the use of solar cells to power vehicles, primary batteries are no longer power spacecraft during eclipse, but are occasionally used for specialty purposes.

Per AFI 91-202 [11]: Section 10.8.5.6.3 states that batteries at EOL shall undergo de-activation. If this is impractical, the batteries shall be left with a permanent electrical drain to prevent recharging. If possible, pressurized batteries shall undergo depressurization at end of life. Structurally, a two-fault tolerant battery casing design is preferred. If AFI 91-202 is followed by the standard operating procedure of the battery, then the risk evaluation of the battery is a simple process. All deviations from preferred operational procedure need to be evaluated.

These Air Force guidelines are similar to the design guidelines for ESA [55]: The discharge of batteries (safe complete discharge to 0 V or to a safe state of charge to ensure no break-up risk) and their subsequent disconnection from charging circuits is a preferred option, as passivation of the solar array can be sufficient to lead to rapid complete battery discharge (initially via the power bus loads and, then, also via the leakage current of control electronics connected to the battery). The batteries can also be left with a permanent electrical drain to prevent recharging; the passivation device should be robust enough to cope with the long lifetime required and the harsh environment at EOL (e.g. loss of temperature control, radiations) to avoid losing battery passivation over time; if it is not possible to eliminate all energy or disconnect the batteries, the batteries can be self-protected provided that the absence of break-up risk can be guaranteed (to be supported by risk assessment, analysis or tests under the environmental conditions after passivation); small batteries (e.g. for cubesats) can be protected in containers to limit debris propagation in case of explosion (to be demonstrated by analysis based on the maximum stored energy and the enclosing structure).

Li-ion battery use has further guidelines recommended by NASA and SMC [56][57]. These guidelines state that lithium-based cells/batteries have a high specific energy and hazard potential, thus they are required to be at least two-fault tolerant to any catastrophic failure unless a more stringent requirement is dictated [56]. To satisfy AFI 91-202, battery cells are normally designed with burst disks that will fail and eject electrolyte and possibly some internal battery material if a high temperature exceedance occurs. In a standard battery design, the material is ejected into a secondary container which also has a vent valve to space. It is recommended that the cell pressure relief devices be demonstrated by test to show that the vent operates as intended and that the vent is adequate to prevent cell fragmentation from an overpressure condition [56][57].

Batteries that follow the guidelines and have been tested to have both adequate cell venting and sufficient secondary containment that prevents fragmentation are considered two fault tolerant, and the cause of the battery cell failure (i.e. over temperature, internal short, external short, or overcharge) is not relevant. If two-fault-tolerance is maintained in the battery design the possibility of a debris forming explosion event from the battery is $<1\times10^{-6}$ at all phases of operation. If two fault tolerance is not demonstrated by testing and maintained throughout operation, then the following sections need to be applied.

3.4.1 Over Temperature Induced Overpressure Due to Failures

There are several scenarios that can lead to an over temperature condition in a battery. Since batteries typically contain either a gas (hydrogen in Ni-H_2) or a volatile electrolyte solution, operation of batteries at temperatures beyond their recommended temperature range can cause rupture of the cell due to over pressure. For gas filled batteries the ideal gas law calculation [Eqs. (1)–(4)] can be made to determine the probability of an overpressure of the battery device due to changes in temperature. For organic electrolyte containing batteries a knowledge of the boiling point and vapor pressure of the electrolyte components are needed. There are several scenarios that may lead to over temperature situations: overcharging of the battery, internal and external shorts, failure of battery heaters in the on position, and variance in temperature at EOL.

At EOL is also possible that temperatures below the freezing point of the battery electrolyte will be achieved. Bursting of the cell due to freezing is not an explosive event.

Battery cells are often designed to rupture at a vent before an overpressure burst occurs. A vented secondary container is used to contain debris from a burst cell and allow over pressures to vent to space.

3.4.1.1 3.4.1.1 Operating Voltage Thresholds Exceeded (high or low)

During operation the battery is controlled by the battery charge control (BCC) circuit. An over temperature cutoff switch should be provided to prevent the battery temperature from exceeding maximum temperature range as a result of overcharging via the BCC circuit. During EOL the spacecraft is non-operational and thus has no opportunity for operational voltage exceedance on the batteries to occur due to charging provided the guidelines are followed.

3.4.1.2 Short Circuits

A short circuit can potentially result in a catastrophic scenario of runaway over temperature depending on the resistance of the short, battery state-of-charge, and how the battery cells are configured within the battery. In the case of an external short-circuit, the battery's positive and negative tabs are connected directly with a minimal electric load that is external of the cell case. Under normal battery operation, the battery's positive and negative electrodes are divided by a separator, usually a thin polymer or ceramic material that prevents the uncontrolled transport of electrons from the negative electrode to the positive electrode. In the case of an internal short-circuit, which could be a result of a debris penetration, a crash accident, or foreign object debris (FOD) within the device, the separator is ruptured in a localized area. The battery produces an electric current from electrochemical reactions that is discharged through the shorted area. This localized energy release causes heating of the cell components which can lead to melting and progressive failure of the separator, or even thermal runaway. To help prevent internal short circuits due to foreign objects inside the battery cell IEEE standards (5.4.2.3 detection of damaged cores) recommend X-raying to detect nonconforming cell cores [58]. Not every short will lead to catastrophic (explosive) failure of the battery, but if the resistance of the short path is low enough that it creates a temperature rise then there is a risk that the cell will spontaneously go into thermal runaway. Thermal runaway occurs when the internal battery temperature is high enough for the cathode material to decompose into highly oxidizing components that ignite the fuels in the cell, such as organics, polymers, and aluminum. Above a certain threshold temperature on most batteries cells a thermal runaway can occur in any cell that hasn't been discharged. Furthermore, if multiple cells are configured in a parallel string, an external short will draw energy from all the parallel cells which will significantly increase the amount of heat generated at the short site. The possibility for a short circuit to lead to thermal runaway becomes less at lower states-of-charge. Cells at a higher state-of-charge contain more stored energy and thus can generate more heat during a short circuit.

Generally external shorts are mitigated by a robust design as well as multiple layers of isolation. Internal shorts from foreign object debris (FOD) or manufacturing defects are now the most common causes of thermal runaway in commercial rechargeable batteries such as lithium ion cell applications. This can be viewed as a single point failure for the cells. Space systems are reliant on a good manufacturing and critically robust screening processes to prevent this type of catastrophic failure. However, at this time space prime contractors view the potential of internal shorts as not credible despite catastrophic incidents frequently occurring within other military and commercial non-space battery applications.

3.4.2 Nickel Hydrogen Battery

Nickel hydrogen batteries contain pressurized hydrogen. An ideal gas law calculation [Eqs. (1)–(4)] can be made to determine the probability of an over pressure of the battery device due solely to changes in temperature. For typical on-orbit batteries this calculation is already determined. The Maximum allowable working pressure (MAWP) is 950 psi. The batteries have a designed burst pressure twice the MAWP, ≥2000 psi, and a proof pressure 1.5x MAWP, ≥1500 psi. The EOL safing pressure is estimated to be 200 psi. For there to be an explosion risk temperature would need to exceed 1000K. There is no credible heat source capable of heating the batteries to their burst pressure. Also based on how the battery operates as battery temperature increases, the self-discharge rate also increases. This process reduces the pressure as it consumes the hydrogen. The self-discharge rate is much faster than a pressure increase due to rising temperature. The batteries do not change the overall risk assessment for explosion at normal operation or EOL. Without a large heat source, the probability of explosion is less than 1 in 1,000,000. If available, the contractor analysis may also calculate the integrated probability of an explosion in the batteries over all mission phases for the typical battery design.

For the typical battery flown on current spacecraft, the design of the battery cell is a welded Inconel 718 cylinder designed in accordance with pressure vessel requirements of MIL-STD-1522A [59] and Eastern and Western Range (EWR) 127-1 [60] section 3.12.4. The battery cell has a design safety factor of 2:1 for EOL battery pressure of 1000 psig. Typically, the battery cell pressure is between 450-550 psig. This means that the battery cell safety factor for ground processing ranges between 3 or 4:1. The battery cell is designated as hazardous leak before burst (LBB). A typical requirement is that no single part shall cause loss of the battery. A battery over-temperature cutoff switch has been provided to prevent the battery temperature from exceeding 95°F (35°C) as a result of overcharging via the Battery Charge Control circuit. Also required is that no rupture of a battery cell shall occur due to overcharging at specified charge rates and that the design of the cell bypass shall not present a flammability hazard as a result of internal contact shorting (e.g. solder ball). If new battery designs are flown a reevaluation of the explosion risk will need to be conducted.

3.4.2.1 Deployment

There are no known on-orbit failures of a US government spacecraft due to a NiH_2 cell or battery explosion. Some Soviet spacecraft failures have been attributed to battery explosion [54]. Aerospace has conducted abuse testing [61] of NiH_2 cells in an attempt to cause a cell to burst or to explode and have not been successful. Methods attempted included short circuiting, overcharging a cell, and rapid charge or discharge, among others. The chance of a temperature excursion leading to explosion is less than one in a million based on the thermal model of the battery subsystem (section 3.4.2).

3.4.2.2 Mission

The risk of explosion from the batteries does not vary from deployment to the mission phase. The batteries are active during the mission phase, especially during times of non-illumination. The temperature is controlled in this phase, so the chance of a temperature excursion leading to explosion is less than one in a million based on the thermal model of the battery subsystem (section 3.4.2).

3.4.2.3 Disposal Maneuvers

The risk of explosion from the batteries does not vary from deployment to the disposal maneuvers phase. The batteries are active during the disposal maneuvers phase. The temperature is still controlled in this phase, so the chance of a temperature excursion leading to explosion is less than one in a million based on the thermal model of the battery subsystem (section 3.4.2).

3.4.2.4 Passivation

“The subject of spacecraft battery disconnection from charging circuits can sometimes lead to extensive debates between the operational community and those responsible for environmental safety. Leak-before-burst designs, while beneficial, do not eliminate the possibility of an explosion. Inadvertent disconnection of the battery from the charging line can easily be avoided with separate, multiple separation subsystems. If planned early in the design phase, such circuits are low cost and low weight. Some exceptions to the rules have also arisen. Many modern spacecraft batteries are highly pressurized and cannot easily be depressurized at the end of mission. If properly disconnected from charging circuits, these batteries are normally safe despite their internal pressures. The NiH_2 batteries are typically placed in “let down mode” during passivation. This will drain stored energy in the battery and prevent recharging. If the batteries on the spacecraft are of a new design, the passivation procedure of the batteries will have to be re-evaluated.

3.4.2.5 End-of-Life

During EOL batteries are discharged and not operational.

3.4.3 Lithium Ion Battery

A lithium-ion battery is a rechargeable battery in-which lithium ions move from the negative electrode to the positive electrode during discharge and back when charging. Cathode and anode materials are chosen based on a range of considerations such as energy density, power density, cycle life, storage life, and cost [60]. The Aerospace Corporation Lithium-Ion Battery Standard for Spacecraft Applications has been developed [63]–[65]. The electrolyte used is a high vapor pressure organic electrolyte and a lithium salt. The Li-ion batteries enter EOL with the residual unused charge remaining. This charge might be as high as 80% of the initial charge on the subsystem batteries. These batteries will self-discharge very slowly during the beginning of EOL. There is often a small current load on them due to active electronic components on the ordnance bus that would help dissipate the remaining charge. The discharge would occur within two years of entering EOL.

Since lithium-based cells/batteries have a high specific energy and hazard potential, they are required to be at least two-fault tolerant to any catastrophic failure unless a more stringent requirement is dictated. Li-ion cells are normally designed with burst disks that will fail and eject electrolyte and possibly some internal battery material if a high temperature exceedance occurs. Overpressure protection for cells is required for safe handling of the battery prior to installation for flight; examples include burst disks and heat-sealed pouches. Even when the electrolyte leakage from a cell is due to an energetic rupture of the burst disk, such as can occur during a thermal runaway event, catastrophic failure of the whole battery housing is not expected because the housing also contains a relief burst disk designed to accommodate all the cells within the battery venting at the same time at a 3:1 burst pressure safety factor. It is recommended that the cell pressure relief devices be demonstrated by test to show that the cell vent reliably operates as intended and that it is adequate to prevent cell fragmentation. Any expelled electrolyte from the cells is expected to condense on the inside of the battery housing and later escape from the structure as a gas. In addition, it is recommended that the battery housing and vent performance should be adequate to prevent a debris forming event, as demonstrated by testing of a worst-case thermal runaway condition. As long as two-fault-tolerance is demonstrated and maintained in the battery the possibility of an explosion event from the battery is $<1\times10^{-6}$. Any design flown without meeting these guidelines will need risk evaluation based on overpressure due to temperature or voltage exceedance and failure due to shorting.

The freezing point of the electrolyte is dependent on the organic carbonate used. The salt inside the electrolyte could lower the freezing point further. Without specific information on the freezing point of the specific electrolyte and its volume change during crystallization, it must then be assumed freeze-thaw cycling is possible in Li ion batteries during EOL. The lower temperature limit of 248 K, the low qualification temperature, is assumed to be due to freezing concerns and 248 K is within one standard deviation of expected temperatures. The rupture of the battery cell would result in electrolyte leaking into the battery casing but does not represent a debris forming explosion event.

3.4.3.1 Deployment

Provided guidelines are followed, the chance of a temperature excursion leading to explosion is less than one in a million based on the thermal model of the battery subsystem.

3.4.3.2 Mission

The risk of explosion from the batteries does not vary from deployment to the mission phase. The batteries are active during the mission phase, especially during times of non-illumination. The temperature is controlled in this phase, so the chance of a temperature excursion leading to explosion is less than one in a million based on the thermal model of the battery subsystem.

3.4.3.3 Disposal Maneuvers

The risk of explosion from the batteries does not vary from deployment to the disposal maneuvers phase. The batteries are active during the disposal maneuvers phase. The temperature is still controlled in this phase, so the chance of a temperature excursion leading to explosion is less than one in a million based on the thermal model of the battery subsystem (section 3.4.3).

3.4.3.4 Passivation

The lithium ion batteries are typically placed in "let down mode" during passivation. This will drain stored energy in the battery and prevent recharging. If the batteries on the spacecraft are of a new design, the passivation procedure of the batteries will have to be re-evaluated.

3.4.3.5 End-of-Life

During EOL batteries are discharged and not operational.

3.4.4 Other Batteries

There are several electrochemical cells that potentially could be used for satellites. These batteries are typically one-time use items, however the one-time use can be a continual drain over the life of the vehicle. Chemical batteries that contain significant amounts of energy should still be compliant with the recommendations of NiH_2 or lithium ion batteries. If two fault tolerance is maintained, then no elevated risk is expected from use of these batteries. However, each novel battery design should be evaluated for adherence to guidelines and where novel batteries fall short of guidelines risk assessment for explosion potential should be done.

3.4.5 Small (Cube) Satellites

There is a desire to fly small satellites with batteries that don't adhere to the recommended guidelines (section 3.4). They typically fall short of passivating the battery at end of life, and or having a secondary container to ensure no fragmentation is possible (two fault tolerance). Small satellites, picosats and CubeSats typically have a small number of commercial Li-ion cells, enclosed or partially in a battery box. NASA guidelines [56] for unmanned spacecraft called for Li-ion cells to have;

- Electrical Safety
  - Individual cells should be capable of surviving a short circuit current with a vent opening to release products.
  - Current and temperature monitoring should be utilized to preclude the inadvertent venting of cells.

- Flight Battery cases should be designed to an ultimate safety factor of 3:1 with respect to the worst case pressure buildup for normal operations.

- Voltage Limits
  - No cell should be allowed to discharge below the minimum voltage limits recommended by the manufacturer during discharge or charge above the maximum voltage limits recommended by the manufacturer during charge.

The stricter guideline for two fault tolerance is reserved only for crewed spacecraft in this NASA guideline. Commercial Li-ion batteries have a vent as a safety feature that is universal. Also they typically have a resettable PTC, which disables the cell if the current discharge is too high, and a current interrupt device (CID) that is designed to trigger at pressures before the vent opens, disabling the cell. Both of these features act to reduce risk, but neither guarantees protection from explosions and they don't count toward the two fault tolerance criteria.

Commercial batteries typically have some destructive testing for things related to thermal runaway and most have a UL certification [66]. To obtain UL certification batteries (both lithium ion and traditional nickel metal hydride) go through destructive testing. This destructive testing includes electrical, mechanical, and environmental testing. These certifications are conducted for terrestrial environments. This destructive testing generally requires that the samples shall not explode or catch fire during the test. However, some testing adds the additional requirement that the sample shall not vent or leak. Thus, an UL certification doesn't guarantee that the core of the battery won't vent or leak under all destructive scenarios. UL Standard 1642 does include a test which subjects the test battery to heat above a burner within a small, screened-in enclosure "until it explodes or the cell or battery has ignited and burned out" [66]. In order to comply with UL 1642, "no part of an exploding cell or battery shall penetrate the wire screen such that some or all of the cell or battery protrudes through the screen." This UL certification is for individual batteries, unless the manufacturer indicates that it is intended for use in series or parallel. For series or parallel use, additional tests on five sets of batteries are to be conducted using the maximum number of batteries to be covered for each configuration. Unless the exact satellite configuration of the batteries is tested the UL certification will not necessarily indicate low risk to debris forming events. A series of batteries may require additional testing to UL certify it. Lastly a UL certification doesn't indicate the final product is also UL certified. The UL documents conclude that the final acceptability of these batteries is dependent on their use in a completed product that complies with the requirements applicable to such product. Small satellites that wish to avoid one of more of the guidelines in section 3.4 may do so at the possibility of increased risk.

Due to the lack of failure modes for individual Li-ion cells to explosively fail, as well as the possible enclosure of a battery box, the probability of a small sat to have a battery failure resulting in fragmentation of the vehicle can be said to be less than 1 in 1000. The reliability of these small satellite batteries is often greater than 1 in 1000 thus, if every cell failure lead to debris formation, they'd still be compliant. The probability of a thermal runaway event in a larger vehicle battery causing fragmentation must be tested for each battery design to determine the likelihood of a propagating thermal runaway leading to a debris forming event [67].

3.5 Thermal Control Subsystem

The only part of the thermal control subsystem with stored energy is the heat pipes. These have never been identified as the likely source of a satellite break-up [54]. The heat pipes are pressure devices filled with liquid ammonia (NH_3). The main explosion risk for heat pipes is a temperature excursion that increases the pressure beyond the capability of the heat pipe. The design burst pressure of the heat pipes is twice the MAWP. Typically, the heat pipes are tested to a qualification pressure (typically 2250 psi). The contractor or customer should supply the pressure rating of the heat pipe subsystem used on the vehicle for evaluation. The heat pipe pressure analysis is similar to the propellant tanks that contain liquid propellant. The pressure is controlled by the vapor pressure of the ammonia in the heat pipe. The vapor pressure of the ammonia can be predicted at a given temperature by the following equation [68]:

$$P(kPa) = 10^{\left(6.67956 - \frac{1002.711}{T - 25.215}\right)} \tag{29}$$

A typical flight qualified heat pipe is predicted to reach the burst pressure (2250 psi) at 428 K. This is greater than the liquid ammonia critical temperature (405 K) and pressure (1645 psi) [68] above which ammonia would change from a liquid to a super critical fluid. The 428 K temperature predicted to reach the burst pressure is more than five standard deviations from a typical spacecraft temperature (~291 K, with σ = 2 K for operations) (272 K, with σ = 11 K for EOL) during any phase of the spacecraft lifetime including EOL. If the heat pipes are not typical or have a redesign or use a medium different than liquid ammonia, further analysis will be needed to determine the explosion risk.

Another chemical used in cryogenic heat pipes is ethane. The vapor pressure of the Ethane can be predicted at a given temperature by the following equation [68]:

$$P(bar) = 10^{\left(3.93835 - \frac{659.739}{T - 16.719}\right)} \tag{30}$$

The ethane critical temperature is (305.3 K) and pressure (49 bar) [66]. Ethane operates at roughly -113 C to -23 °C and is non-operational to 50 °C. Ammonia can operate at -40 °C to 60 °C and is non-operating up to 127 °C. At high temperatures, water heat pipes could potentially be used.

Table 4 is appropriate for evaluation of heat pipes.

Occasionally cryocoolers which are gas filled maybe used for thermal management. These cryocoolers can be treated as any other pressurant system.

3.5.1 Deployment

The pressure in the heat pipes is controlled by the vapor pressure of the liquid ammonia [Eq. (29)]. During the operational lifetime of the spacecraft the heat pipes are designed to actively dissipate any excess heat severely reducing the temperature excursions experienced by the spacecraft. The typical maximum expected temperature deviations are ± 11 °C. A thermal model of the spacecraft provided by the contractor or customer should be able to identify whether the spacecraft is expected to experience typical temperature excursions. As stated above, the temperature at which the vapor pressure of the liquid ammonia is high enough to burst the heat pipes is typically much more than five temperature standard deviations above the operating temperature. The probability of an explosion in a heat pipe during deployment is less than one in a million.

3.5.2 Mission

During the mission phase the heat pipes are active and the spacecraft is thermally controlled. There is no difference between the deployment phase and the mission phase for the heat pipes operation, thus no change in the explosion risk.

3.5.3 Disposal Maneuvers

During the disposal maneuvers phase the heat pipes are once again active and the spacecraft is thermally controlled. In the disposal maneuvers phase the risk of explosion is similar to the mission and deployment phase.

3.5.4 Passivation

Heat pipes by design contain internal fluids at high pressure and, again, are not easily depressurized. On the other hand, these ruggedly-built subsystems have never been identified as the likely source of a spacecraft break-up. Therefore, depressurization of heat pipes is currently not considered a requirement to comply with passivation guidelines. In the passivation phase the risk of explosion is similar to the mission phase.

3.5.5 End-of-Life

At EOL the temperature of the spacecraft is no longer actively controlled, although the heat pipes will still transport heat even if the electrical subsystem no longer actively uses on-board heaters. The heat pipes do not require electrical subsystems to be active. At EOL the temperature excursions are once again typically larger than during active life. Similar to the liquid propellant tanks a worst-case attitude for solar heating purposes may be assumed. Table 6 is suitable for analysis of the heat pipes in this stage. If the temperature needed to reach proof pressure of the heat pipes at EOL, calculated by using Eq. (29), is more than five standard deviations from the predicted EOL temperature, the probability is less than 1 in a million.

4. Conclusions

The evaluation report is required to provide the integrated probability of explosion which is often shown by summing the independent explosion probabilities. The language of this section is typically "All the failure modes leading to debris-generating explosions identified in this report have a total probability of X." As part of this summary, a table similar to that shown in example Table 12, will be produced that breaks down the report evaluation by subsystem. The spacecraft may have more than one type of subsystem, e.g. thruster types, and the table may reflect that. The table should list failure modes considered, cause of failure, the mitigating factors on the spacecraft (if any), the mission phase evaluated, and the probability determined by the current evaluation.

Table 12. Summary for All Spacecraft Components/Subsystems That Contain Stored Energy and May Lead to an Accidental On-Orbit Explosion During Any Operational Phase

Spacecraft Subsystems/ Critical Item	Failure Modes	Causes	Mitigating Factors	Mission Phase(s)	Probability of Accidental Explosion	Severity	Probability	Risk
Propulsion								
Pressurant	Over pressure	Temperature increase		Deployment	x	x	x	x
Pressurant	Over pressure	Temperature increase		Mission	x	x	x	x
Pressurant	Over pressure	Temperature increase		Disposal maneuvers	x	x	x	x
Pressurant	Over pressure	Temperature increase		Passivation	x	x	x	x
Pressurant	Over pressure	Temperature increase		End-of-Life	x	x	x	x
Pressurant	Over pressure	Regulator failure		Deployment	x	x	x	x
Pressurant	Over pressure	Regulator failure		Mission	x	x	x	x
Pressurant	Over pressure	Regulator failure		Disposal maneuvers	x	x	x	x
Pressurant	Over pressure	Regulator failure		Passivation	x	x	x	x
Pressurant	Over pressure	Regulator failure		End-of-Life	x	x	x	x
Electric thruster gas	Over pressure	Temperature increase		Deployment	x	x	x	x
Electric thruster gas	Over pressure	Temperature increase		Mission	x	x	x	x
Electric thruster gas	Over pressure	Temperature increase		Disposal maneuvers	x	x	x	x
Electric thruster gas	Over pressure	Temperature increase		Passivation	x	x	x	x
Electric thruster gas	Over pressure	Temperature increase		End-of-Life	x	x	x	x

Spacecraft Subsystems/ Critical Item	Failure Modes	Causes	Mitigating Factors	Mission Phase(s)	Probability of Accidental Explosion	Severity	Probability	Risk
Propellant	Over pressure	Temperature increase		Deployment	x	x	x	x
Propellant	Over pressure	Temperature increase		Mission	x	x	x	x
Propellant	Over pressure	Temperature increase		Disposal maneuvers	x	x	x	x
Propellant	Over pressure	Temperature increase		Passivation	x	x	x	x
Propellant	Over pressure	Temperature increase		End-of-Life	x	x	x	x
Oxidizer	Over pressure	Temperature increase		Deployment	x	x	x	x
Oxidizer	Over pressure	Temperature increase		Mission	x	x	x	x
Oxidizer	Over pressure	Temperature increase		Disposal maneuvers	x	x	x	x
Oxidizer	Over pressure	Temperature increase		Passivation	x	x	x	x
Oxidizer	Over pressure	Temperature increase		End-of-Life	x	x	x	x
Propellant	Autoignition	Temperature increase		Deployment	x	x	x	x
Propellant	Autoignition	Temperature increase		Mission	x	x	x	x
Propellant	Autoignition	Temperature increase		Disposal maneuvers	x	x	x	x
Propellant	Autoignition	Temperature increase		Passivation	x	x	x	x
Propellant	Autoignition	Temperature increase		End-of-Life	x	x	x	x
Hydrazine	Decomposition	Temperature increase		Deployment	x	x	x	x

Spacecraft Subsystems/ Critical Item	Failure Modes	Causes	Mitigating Factors	Mission Phase(s)	Probability of Accidental Explosion	Severity	Probability	Risk
Hydrazine	Decomposition	Temperature increase		Mission	x	x	x	x
Hydrazine	Decomposition	Temperature increase		Disposal maneuvers	x	x	x	x
Hydrazine	Decomposition	Temperature increase		Passivation	x	x	x	x
Hydrazine	Decomposition	Temperature increase		End-of-Life	x	x	x	x
Thermal					x	x	x	x
Ammonia	Over pressure	Temperature increase		Deployment	x	x	x	x
Ammonia	Over pressure	Temperature increase		Mission	x	x	x	x
Ammonia	Over pressure	Temperature increase		Disposal maneuvers	x	x	x	x
Ammonia	Over pressure	Temperature increase		Passivation	x	x	x	x
Ammonia	Over pressure	Temperature increase		End-of-Life	x	x	x	x
Ammonia	Autoignition	Temperature increase		Deployment	x	x	x	x
Ammonia	Autoignition	Temperature increase		Mission	x	x	x	x
Ammonia	Autoignition	Temperature increase		Disposal maneuvers	x	x	x	x
Ammonia	Autoignition	Temperature increase		Passivation	x	x	x	x
Ammonia	Autoignition	Temperature increase		End-of-Life	x	x	x	x
Pyrovalve					x	x	x	x
Pyrovalve	Accidental firing	Explosive activated		Deployment	x	x	x	x

Spacecraft Subsystems/ Critical Item	Failure Modes	Causes	Mitigating Factors	Mission Phase(s)	Probability of Accidental Explosion	Severity	Probability	Risk
Pyrovalve	Accidental firing	Explosive activated		Mission	x	x	x	x
Pyrovalve	Accidental firing	Explosive activated		Disposal maneuvers	x	x	x	x
Pyrovalve	Accidental firing	Explosive activated		Passivation	x	x	x	x
Pyrovalve	Accidental firing	Explosive activated		End-of-Life	x	x	x	x
Reaction wheel assembly					x	x	x	x
Reaction wheel	Unintentional release	Structural failure		Deployment	x	x	x	x
Reaction wheel	Unintentional release	Structural failure		Mission	x	x	x	x
Reaction wheel	Unintentional release	Structural failure		Disposal maneuvers-	x	x	x	x
Reaction wheel	Unintentional release	Structural failure		Passivation	x	x	x	x
Reaction wheel	Unintentional release	Structural failure		End-of-Life	x	x	x	x
Power					x	x	x	x
Batteries	Over pressure	Temperature increase		Deployment	x	x	x	x
Batteries	Over pressure	Temperature increase		Mission	x	x	x	x
Batteries	Over pressure	Temperature increase		Disposal maneuvers-	x	x	x	x
Batteries	Over pressure	Temperature increase		Passivation	x	x	x	x
Batteries	Over pressure	Temperature increase		End-of-Life	x	x	x	x
Batteries	Over pressure	Over charge		Deployment	x	x	x	x

Spacecraft Subsystems/ Critical Item	Failure Modes	Causes	Mitigating Factors	Mission Phase(s)	Probability of Accidental Explosion	Severity	Probability	Risk
Batteries	Over pressure	Over charge		Mission	x	x	x	x
Batteries	Over pressure	Over charge		Disposal maneuvers-	x	x	x	x
Batteries	Over pressure	Over charge		Passivation	x	x	x	x
Batteries	Over pressure	Over charge		End-of-Life				
Batteries	Over pressure	Internal short		Deployment	x	x	x	x
Batteries	Over pressure	Internal short		Mission	x	x	x	x
Batteries	Over pressure	Internal short		Disposal maneuvers-	x	x	x	x
Batteries	Over pressure	Internal short		Passivation	x	x	x	x
Batteries	Over pressure	Internal short		End-of-Life	x	x	x	x
Batteries	Over pressure	Propagation		Deployment	x	x	x	x
Batteries	Over pressure	Propagation		Mission	x	x	x	x
Batteries	Over pressure	Propagation		Disposal maneuvers-	x	x	x	x
Batteries	Over pressure	Propagation		Passivation	x	x	x	x
Batteries	Over pressure	Propagation		End-of-Life	x	x	x	x

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