Coventry University

320EKM Project

Fracture Control of Spacecraft Components

Module Leader:
Nerea Etura Luque

Supervisor:
Kashif Khan

Author:
Guillermo Joaquin Dominguez Calabuig
Email:
domingu3@coventry.ac.uk
guilledcalabuig@gmail.com

June 13, 2017

Declaration of Originality

This project is all my own work and has not been copied in part or in whole from any other source except where duly acknowledged. As such, all use of previously published work (from books, journals, magazines, internet, etc) has been acknowledged within the main report to an item in the References or Bibliography lists.

I also agree that an electronic copy of this project may be stored and used for the purposes of plagiarism prevention and detection.

Copyright Acknowledegment

Signed: Guillermo Joaquín Domínguez Calabuig Date: 05.05.2017

Office Stamp

Abstract

Fracture critical parts are indispensable for mission success and so it is important to perform as many analysis as possible to understand their fracture mechanisms and consider their criticality in order to optimise their performance and life as much as possible. The project is focused in analysing spacecraft components which are probably categorised as fracture critical parts as they suffer fatigue problems during the mission using industry standards based on NASA (NASA 2016) and ESA (ECSS 2009*a*) requirements for their space missions. Moreover, only metallic components are identified and analysed in order to allow the use of linear elastic fracture mechanics for analysis. A literature review summarising the fracture control procedures and explaining some basics of fracture mechanics, fatigue analysis and the extended finite element modeling is previously performed. Additionally, spacecraft experienced environments and materials used are reviewed. Pressurised structures of manned modules, propellant tanks, and planetary rover's wheels are identified among other components as possible fracture critical elements due to the cyclic loads experienced throughout all their service lives. The materials of the previous modules are identified as different aluminium alloys, such as an aluminium copper alloy (Al 2219), aluminium lithium alloy (Al 2195) and aluminium silicon magnesium alloy (Al 6061) respectively. These materials are analysed using a fracture mechanics approach and an extended finite element method (XFEM) implemented in the Abaqus software, simulating a static overloading case where an existing crack grows to failure under forces applied as boundary displacements. The aluminium-lithium alloy superior fracture capabilities over the others are identified in the analysis, but as mentioned in the conclusion (section 5), stress corrosion cracking and other environmental effects should be taken into account when deciding which materials are employed for each function of a space mission.

Contents

List of Figures

List of Tables

Abbreviations

AR Acceptance review

CDR Critical design review

ECSS European Cooperation for Space Standardization

EPM Elastic-plastic fracture mechanics

ESA European Space Agency

FCI Fracture critical item

FCR Fracture control requirements

FCSR Fracture control summary report

FEM Finite element modeling

FLLI Fracture limited life item

GEO Geostationary Earth orbit

ISS International space station

LEFM Linear-elastic fracture mechanics

LEO Low Earth orbit

NASA National Aeronautics and Space Administration

NDE Nondestructive evaluation

NDT Nondestructive test

PDR Preliminary design review

PFCI Potential fracture critical item

QR Qualification review

RLV Reusable launch vehicle

SENT Single edge notched tension

SRR System requirements review

TRL Technology readiness level

XFEM Extended finite element modeling

Nomenclature

ΔK Stress intensity range

ΔK_{th} Threshold stress intensity range

ν Poisson's ratio

σ Stress

σ_c Critical stress level

σ_Y Yield stress

σ_{Ult} Ultimate stress

a Crack size

a_c Critical crack size

a_j Degree of freedom for crack line function

b_k Degree of freedom for crack tip function

D Predicted cumulative fatigue-damague ratio

E Elasticity modulus

$F_\alpha(x)$ Asymptotic crack-tip function

G Energy release rate

$H(x)$ Heaviside function

K Stress intensity factor

K_I Stress intensity factor for Mode I

K_c Critical stress intensity factor for Mode I

K_{Ic} Plane strain fracture toughness for Mode I

K_{III} Stress intensity factor for Mode III

K_{II} Stress intensity factor for Mode II

K_{Iscc} Stress-corrosion cracking threshold for Mode I

N Load cycle

N Shape function

n Number of nodes

p Work done in plastic deformation near the tip

R Ratio between minimum and maximum stress

T Surface energy per unit area of the cracked surface

t^* Crack propagation time $t_{Fail} - t_0$

t_0 Time of crack growth initiation

t_{Fail} Time at plate failure

u Displacement

W Plate width

Acknowledgements

First of all, I would like to thank Mr Khan for supervising my end of bachelor project. Thanks to him I have introduced myself into a field of knowledge with vast applications which is fracture mechanics and fatigue, and learned how to use Abaqus thanks to his tutorials. Moreover, I would also like to thank Mr Ghaleeh for assisting my interim review presentation and for his feedback. I also have to thank my lecturers at the Polytechnic University of Valencia (UPV), Mr Busquets Mataix and Mr Giner Maravilla, for introducing and fomenting my interest of aerospace materials and mechanics and finite element modelling respectively. All their advices and teaching will surely help me in my future career and master focused on space engineering. Last but not least, I also want to show my gratitude to Nerea Etura Luque for allowing me to enroll to her module and her guidance and advices throughout the project.

Chapter 1. Introduction

Fracture critical parts are indispensable for mission success and so it is important to perform as many analysis as possible to understand their fracture mechanisms and consider their criticality in order to optimise their performance and life as much as possible. The project will be focused on fracture control procedures and requirements (FCR). Fracture control, as mentioned in section 1.2, is a very important subject of every product assurance scheme to maintain the required safety and operational levels. As usually small flaws are introduced during processing of materials and during fabrication of pieces and components (Sarafin 1995, p. 57), it is necessary to verify the structural life for cyclic loading for these components to establish some limits in the detectability of flaws, using fatigue and fracture analysis, proof tests and non-destructive tests and evaluations (NDT and NDE). Metallography needs to be used to study the sizes and forms of the cracks detected. It is normally performed on structural items such as pressure vessels, composite structures, joints and other load bearing components.

One of the main objectives of this project is to simulate some fracture critical spacecraft hardware material and analyse their behaviour in order to gather more data about them. The project will also be centred in summarising the fracture control requirements, plans and processes in order to achieve a complete understanding of the subject as it is fundamental in every space mission. An emphasis will be mode throughout the project in metallic components, as composites require further investigation and have more rigorous requirements.

The software employed to simulate is called Abaqus. It is a finite element analysis programme. Although NASA standards (NASA 2016) recommend to use NASGRO software (developed initially for the shuttle project (Wayne et al. 2011, 284), the principal sponsor of the development of fracture mechanics as a tool in fracture control) and ESA's (ECSS 2009*a*) recommend ESACRACK (which has some functions based on NASGRO modules), it is convenient to use this programme to familiarise with its interface as it is widely used in industry for various fields.

1.1 Space Industry Development

The space industry´s main role in today's society is to contribute to attain a smart, sustainable and inclusive growth. It does so by contributing to scientific progress, which drives innovation by supplying other sectors with the knowledge acquired, and by targeting major issues as climate change, limited resources and health. Only by data from ESA-led missions, 1870 referred papers have been published in 2015, 11% more than in 2014 (ESA & Fletcher 2016).

Throughout the last decade we have witnessed a steep increase in space commercialization, as many multinationals have expanded their business model to the space sector, especially in the communication and navigation sector. It seems that the exponential growth in technological advances worldwide will continue to encourage private space businesses into looking for new applications and opportunities. An overview of satellite industry revenue is illustrated in Figure 1.

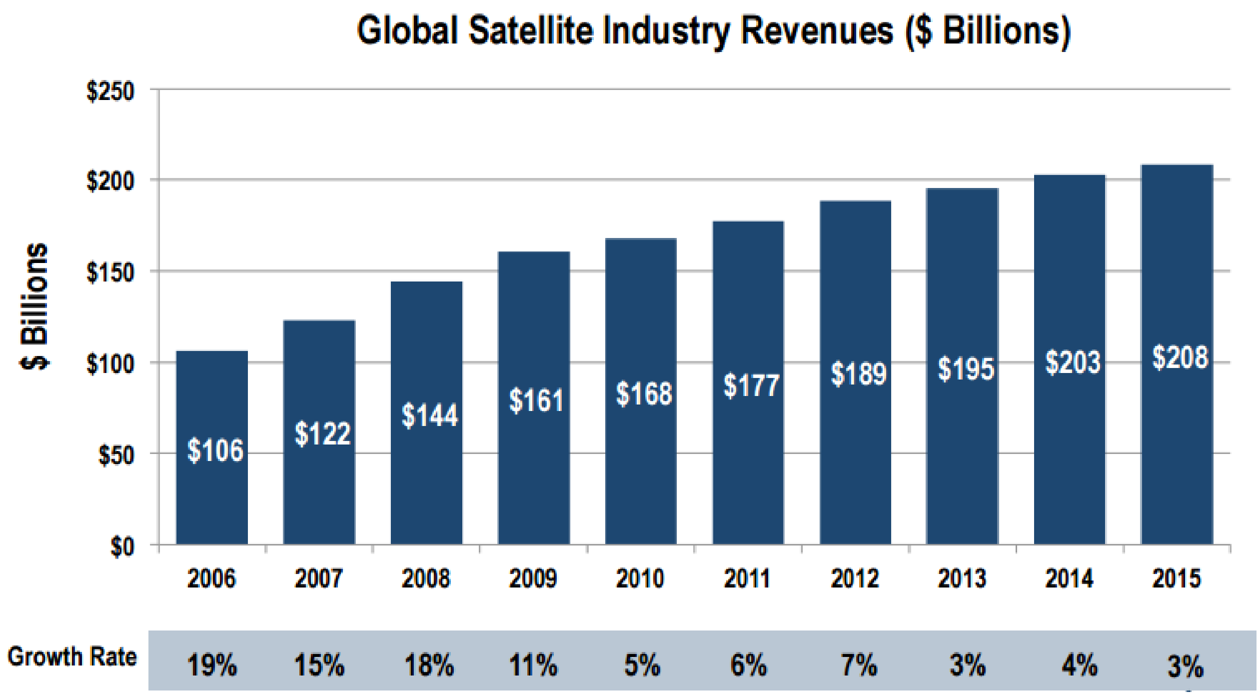


Figure 1: Satellite industry revenue(SIA & The Tauri Group 2016)

In order to improve space accessibility and exploration by inspiring agencies and private companies to take a step forward, it is important to study the field as much as possible. Therefore fields such as fracture mechanics and materials sciences need more investigation as material selection and engineer design determines the efficiency and productivity of every space mission. Nowadays satellites are incorporating standardised subsystems in order to increase the performance by saving in weight and cost while improving reliability.

It then seems plausible for the designer to make a big emphasis on the mass of the system, trying to save weight and costs as much as possible. This is because a lighter payload will increase the capabilities of the launch vehicle which is in charge of injecting the spacecraft from ground into orbit. Launch costs vary from approximately $5000 per kilogram to LEO (Low Earth orbit) to $30000 per kg to GEO (Geostationary Earth orbit) (CANNAE n.d.). As a comparison, Space X Falcon 9 can launch for $4109 per kilogram to LEO (*Upgraded Spacex Falcon 9.1.1 will launch 25% more than old Falcon 9 and bring price down to $4109 per kilogram to LEO* 2013). It can be seen that just a kilogram reduction brings enormous economic advantages. As money is saved in mass, it can partly be dedicated to reliability of the mission by improving test programmes which will be in charge of increasing security and the mission capabilities and life. A drawback is that structures need to be optimised by employing extremely thin and light parts subjected to hight levels of stress, creating fatigue problems if they are subjected to sufficient load cycles.

Moreover, new reusable launch vehicles (RLV) are starting to be developed settling new requirements in order to achieve a high level of security as their components are subjected to bigger fatigue load cycles. It has always been considered that the maintenance schemes for reusable spacecraft like aircraft has always been superior to other travel modes. One component analysed in this project will be based on the aluminium-lithium propellant tanks employed by Space X new RLV, Falcon 9 (Space X 2015). The purpose is to gather as much data as possible from a structural life analysis in order to contribute to fatigue/fracture data availability.

1.2 Product Assurance and Fracture Control

Product assurance management provides technical management leadership in a number of disciplines (Dunn 2016, p. 55). It needs to make sure that each step in the development of a spacecraft produces high-quality components for the amount of money spent. This is applied to the different models of a spacecraft; thermal, structural, engineering, qualification and flight model. The qualification model (or protoflight model), which is a fully functional model of the spacecraft which can be even used as a flight spare in case of launch failure, is subjected to various environmental ground tests such as acoustic and vibration tests, but in fracture control we can no longer rely on them for structural life (we always have to assume initial crack which probably is not there) (Sarafin 1995, p. 387). The general range of the disciplines are the following:

- **Quality Assurance and Software Assurance**, in charge of standardising procedures in order to improve quality of the final product. It is concerned with design, calibrations, workmanship standards, heat treatment control, inspection (or quality control) and testing. Many documentation for this project has been obtained from the ECSS and NASA technical standards (Dunn 2016, p. 55).

- **Reliability and Safety Engineering and Assessment**, where activities are performed to study individual components under environmental testing or real life to predict and quantify their lifetimes, improving safety. In order to increase safety in a space vehicle, designers implement different safety margins and redundancy between load paths and systems. Fracture control procedures are also included here.

- **Materials and processes**, where a process is established to assure sufficient material traceability and documentation of every process followed. It covers topics such as selection criteria and rules.

- **Component part selection procurement**, where a similar process as for the materials and processes discipline is performed but with electric, electronic and electromechanical components verification, traceability and validation.

In contrast to typical quality control employed in other demanding industries, it is very important for this policy to include various materials experts and laboratories to give quick functional support to the different spacecraft projects.

Chapter 2. Literature Review

As mentioned throughout the introduction, fracture control is of paramount importance in the space industry. It is based on the fundamental assumption in structural mechanics that all components have initial small crack defects introduced during fabrication or in service. If these crack grow to an unacceptable level, it can reduce service life and even cause a catastrophic loss. Fracture control methodology aims to prevent these adverse effects, while fracture control planning developed by different space associations are focused on standardising the techniques employed to assure overall safety and quality. However, implementing fracture control is expensive and can sometimes introduce new hazards if it is not done correctly as it complicates the design, like the Tethered satellite in 1992 which failed to deploy in its Shuttle mission (Sarafin 1995, 387), so it is crucial to implement it correctly making sure of every reason for it. The requirements to be imposed are specified in different standards such as the NASA's (NASA 2016) and ECSS's (ECSS 2009*a*), an initiative established to develop standards for use in all European space activities.

It should be understand that safety is not just the final outcome of implementing an efficient fracture control programme, as it can also be used to justify new technological advancements in order to improve performance and therefore system efficiency as investigations and analysis are performed in the background. A perfect example can be seen with the overwrapped pressure vessels developed in the shuttle programme, where thanks to the research in composites fracture mechanics, the orbiter was able to save 546 more kilograms (Wayne et al. 2011, p. 280). Such applications consolidated NASA as the industry leader in the development and application of fracture mechanics technology and fracture control methodology.

Moreover, this standards are only meant to give a set of requirements, but is not focused on how these requirements should be met, leaving the industry and contractors free choice on processes and activities. This is coherent with the space industry as a high specialisation is required and new techniques, processes and machinery are constantly introduced.

NASA standards uses different terms to classify components depending on their criticality. Exempt parts are those which are not subjected to crack growth, non-fracture critical parts are those which can have crack growth but aren't considered a thread (which will follow conventional aerospace industry verifications and quality assurance procedures as mentioned in 1.2), and fracture critical parts are the rest, which need to have their damage tolerance verified and validated by testing or analysis. These analysis take into account components with flaws in the worst location and subjected to the most unfavourable loads, with the requisite of proving that the assumed crack based on previous NDE would not cause a failure in four service lifetimes.

On the other hand, ECSS standards also satisfy NASA's fracture control requirements, but nomenclature may differ. For example, fracture critical parts are called fracture critical items (FCI).

In the following section 2.2, the assessments used to classify each component, along with the organizations involved, requirements, analysis to be performed and other fracture control programme aspects will be explained thoroughly.

2.1 Principles

The requirements imposed in each standards are based on a series of assumptions and prerequisites. Naming these is very necessary to know the background of each assumption and to understand the structure of this project. They are especially useful when alternative approaches are inspected and validated for a required safety and reliability level (ECSS 2009*a*).

- Structural elements have crack-like defects located in the worst possible area and orientation. If NDE does not locate such defect, it does not mean that the assumption is incorrect, but that an upper-bound is introduced on the initial crack size. If suffering from a sufficient number of cycles of an enough amplitude, materials show a tendency to propagate the crack even in an adequate environment.

- The propagation of the initial or load induced crack under a cyclic or continuous stress depends on different aspects such as the material's behaviour, initial size and geometry of crack and item, environment, the amplitude and number of cycles, the time spent under the sustained load and the temperature. Therefore a summary of the materials employed in the space industry 2.3.1 and the environment affecting every space project will be made 2.3.2.

- Linear elastic fracture mechanics (LEFM) is an analytical tool for prediction of crack propagation and critical size which is adequate for metallic materials. For this reason, and in order to explain the theory behind the simulations, an introduction of this field will be made in the methodology section 3, along with fatigue analysis.

- This engineering disciplines however are considered inadequate for the analysis of non-metallic materials such as composites, bonded and sandwich structures. For this reason the fracture control is based on safe life assessments. Such requirements will be mentioned but are beyond the scope of this project.

- Uncertainties in measured material properties and fracture mechanics analysis are considered. For this reason scatter factors and load enhancement factors are implemented.

2.2 Fracture Control Programmes

2.2.1 Programme

2.2.1.1 Fracture Control Plan

The supplier or subcontractor implements a fracture control plan specifying the fracture controls that are established to diminish the risk of catastrophic failure caused by flaws through the service life and it is approved by the responsible authority. It addresses all parts in the program-specific hardware, meeting the requirements specified in the standards such as item classification, responsibilities, approaches, and activities. Approaches such as fault screening, traceability and material selection of fracture critical parts should be detailed.

Both documents take into account the design characteristics of space projects, specifying that the fracture control plan should be updated to keep it current with the programme fracture control approaches.

The ECSS standards gives more details about the design process stating what should be produced in each review. These are:

- **SRR.** For the system requirements review, where compatibility between systems is reviewed, a preliminary hazard analysis, fracture control screening and a written statement stating the applicability of fracture control.
- **PDR.** The preliminary design review is performed after evaluation of thermal and engineering models, with the objective to approve the preliminary design including materials and processes. More details should be stated and the fracture control plan should be submitted for approval by the responsible agency. Also, a list of the potential fracture critical items should be made
- **CDR.** In the critical design review a final design is established and flight hardware manufacturing can start. A final control plan should be approved, verifying requirements, describing results of analysis and tests and listing the items in a more detailed way.
- **AR or QR.** The acceptance or qualification review checks that all qualification activities on subsystems are complete. It requires a fracture control summary report showing completion of every verification activity. Also, tests, evaluations and analysis reports should be performed and every item should be classified and listed.

2.2.1.2 Responsibilities

In the ECSS document, the fracture control or safety authority responsible for the implementation of the fracture control programme is referred as the customer. NASA on the other hand, refers to the responsible fracture control board RFCB as the designated entity that will ensure compliance with the technical requirements. The term will be called the corresponding authority throughout this project for means of clarification.

2.2.1.3 Applicability

Both documents explicitly specify that human-rated space flight projects should impose the whole requirements, However, the ECSS standards develops also a subsection with a reduced fracture control programme for unmanned single missions.

2.2.2 Parts Classification and Requirements

As mentioned earlier, NASA and ECSS standards use different terms and classifications for each component. This means that the approach followed by each standard is different, but requirements and activities for components in need of fracture control are finally the same. Regarding composites, both standards mention that they have to satisfy not only the hardware requirements, but additional requirements. There can be omissions or deviations from the standard if approved by the corresponding authority. While NASA specifies non-structural parts with no credible failure mode caused by flaws, with no credible potential for causing a catastrophic hazard an others approved by the RFCB as Exempt Parts, ECSS specifies by means of screening for structural elements and hazard analysis which ones are potential fracture critical items (PFCI), not mentioning throughout the standards items which don't fall in this category.

ECSS standard then uses a damage tolerance analysis categorising items as safe-life, fail-safe, contained and low-risk and a set of analysis depending on the specific component to classify them into FCIs (with a subsection not mentioned in NASA standards for fracture limited life items and not fracture critical items which remain as PFCIs. Throughout these analysis, it is required to analyse the analytical life to see if they are four times bigger than their service life. NASA uses a similar approach to categorise Non-Fracture Critical Parts and Fracture Critical Parts introducing a section for each where it specifies the different approaches to identify and evaluate each item. The following section will explain the different approaches. The logic diagrams employed by each standards to classify their components are shown in Figure 2 by ECSS and Figure 3 by NASA.

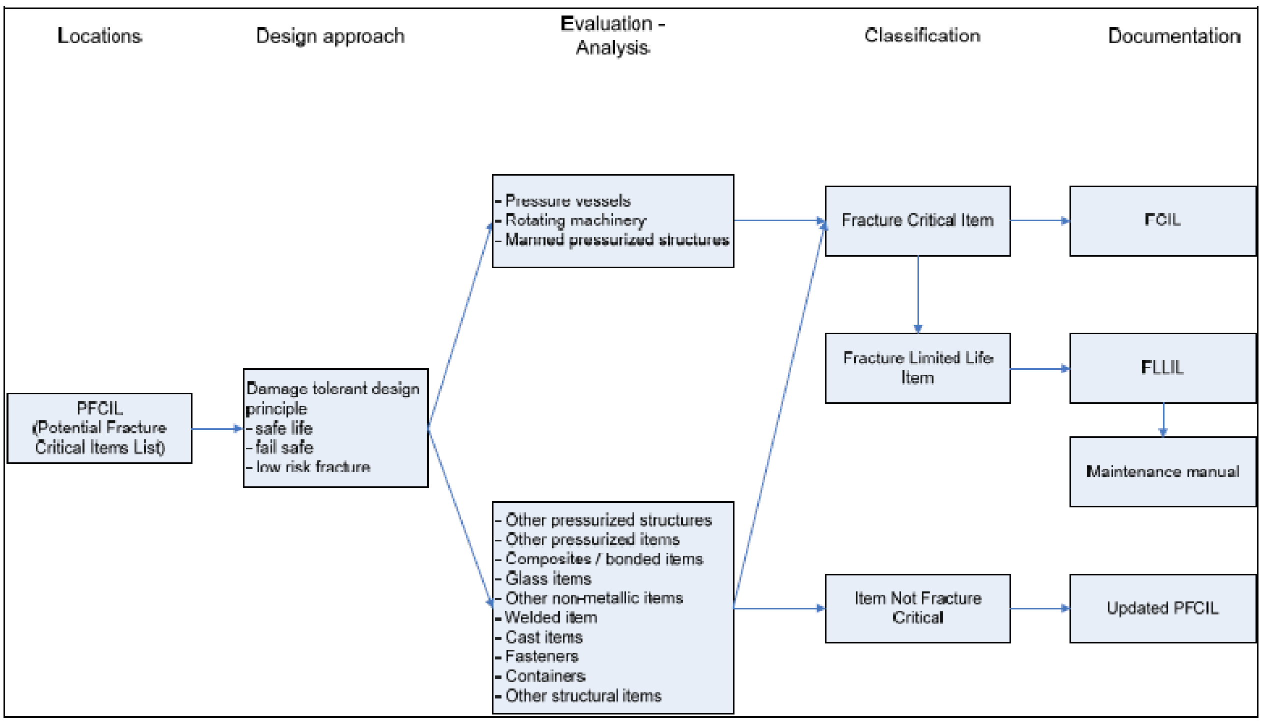


Figure 2: Fracture Control Classification (ECSS 2009*a*)

2.2.2.1 Exempt parts or non PFCI

It has been mentioned before which parts are considered exempt by NASA standards, which are essentially the one which are not considered PFCI by ECSS (non-structural parts). NASA requires identification and to show that this items meet the exempt classification in the fracture control summary report (FCSR) in accordance with the requirements. They are considered to comply with FCR and they just have to follow conventional aerospace verification and qualification assurance procedures.

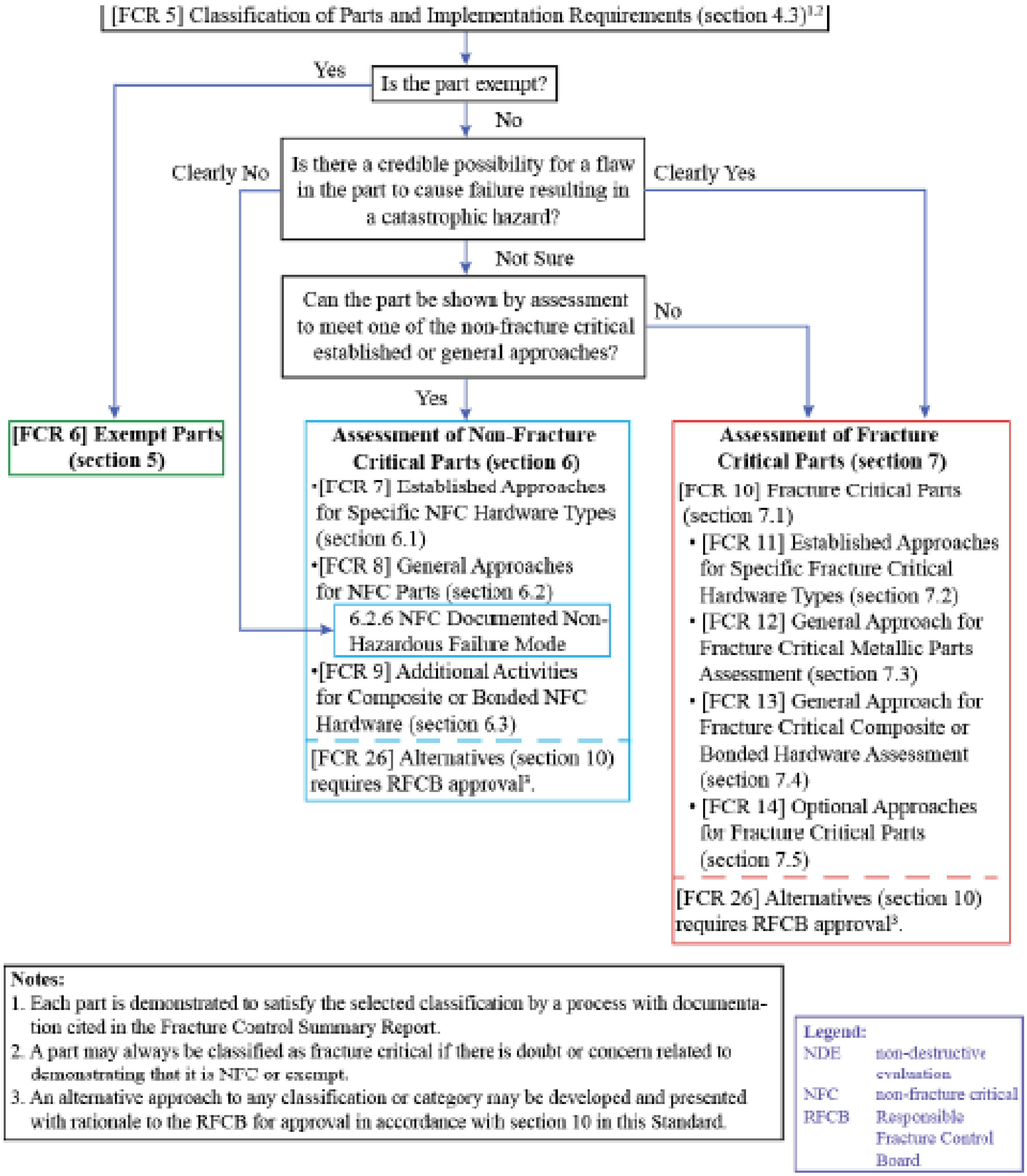


Figure 3: Fracture Control Classification (NASA 2016)

Some examples are flexible insulation blankets, enclosed electrical circuit components/boards, wire bundles and certain batteries.

2.2.2.2 Design principles

Safe-life To classify items as safe life, there are different procedures for metallic items and for composite, bonded and sandwich items. Safe life composites can only be classified as fracture critical or redesigned, while metallic safe life items can be classified as non-fracture critical (Remaining as PFCI), fracture-critical items or fracture limited-life item.

For metallic items, this is derived by seeing if with initial standard defect sizes, the analytical life is four times bigger than the service life. If this is not the case, an improved inspection method which could detect lower initial crack sizes has to be considered. If then with this new method the analytical life is four times bigger, the item can be classified as fracture critical. If not, a reduced safe life has to be considered, which for a RLV such as the shuttle, it still needs to take into account 2 flights. By satisfying this reduced safe life the item can be considered fracture limited-life item (FLLI) (if system programmatics agree) which is a subsection of fracture critical item. NASA standard does not take into account this case. It should be noted that this implies that a maintenance programme should be performed.

Fail-safe For a material to be classified as fail-safe, sufficient redundancy needs to be provided. It should be checked that the failure of the part does not generate pieces or debris that could create a catastrophic or critical hazard. There are some minimum masses and momentums specified, and in NASA standards another classification is used for this items called the low-release mass requirements.

For Metallic parts the remaining fatigue life of the structure should be evaluated using Miner's rule (Equation [3.6]) explained in section 3.1.2, mean fatigue life material characteristics, and a life factor of four as scatter ratio.

If it is then demonstrated that the remaining structure analytical life is bigger than four times the service life, the component can be classified as not fracture critical, if not, a reduced life can be considered to see if it can be classified as a fracture limited-life item. If not, it should be evaluated as a fail-safe item or redesigned.

Contained items Items that are safely confined in an enclosed container should they become loose due to a failure caused by flaw can be considered non fracture critical. An assessment should be performed to verify if this loose item does not penetrate or fracture the enclosure with some safety factors considered. The composites, bonded or sandwich structures the container should not be fracture critical already in order to have a single-point of failure. The container is therefore considered a fracture critical part.

Low-risk fracture items This classification is intended for components that are extremely unlikely to contain or develop critical flaws because of low likelihood of flaws to be created by manufacturing, environmental effects, or service life events. NASA (2016) identifies different criteria for metallic parts to be classified as low-risk, depending on its ductility properties, material employed, von Mises stresses and others. ECSS (2009*a*) refers to table 5-1 of (ECSS 2009*b*) to select materials which are not sensitive to stress corrosion cracking.

It should be noted that metallic welds and castings cannot be qualified as low risk parts as they are manufacturing processes that may contain critical flaws unless inspection data suggests otherwise.

2.2.2.3 Non-fracture critical parts or items

In NASA technical standard, some parts may be classified by the hardware developer as non-fracture critical. The corresponding authority then needs to approve the FCP to ensure that it include the specified elements and approaches. Parts classified as non-fracture critical which are proved to be in accordance with the requirements through the documentation of the FCSR comply with FCR and they just have to follow conventional aerospace verification and qualification assurance procedures.

In section 6 of NASA (2016), other design principles not mentioned 2.2.2.2 are considered such as non-hazardous leak before burst (NHLBB) pressurised components and non-hazardous failure mode. Moreover a list of components which fall in this category are categorised. NHLBB parts are those which the failure mode produces a leakage which is not hazardous for the rest of the spacecraft and it does so before bursting. For metallic parts, an acceptable approach is to show by analysis that a worst-case surface crack will grow into a through the thickness crack without causing unstable propagation.

On the other hand, non-hazardous failure modes are different from exempt parts in the sense that the failure modes identified present no credible catastrophic hazards.

ECSS standards is more focused in categorising fracture critical items, with non-fracture critical items coming along while the analysis proceeds.

Some examples of possible non-fracture critical parts are the following (If they meet some requirements stated in section 6.1 of NASA (2016) and are PFCI not categorised as FCI in (ECSS 2009*a*) (Section 8, Special Requirements) :

- Metallic Fasteners, Rivets, Shear Pins, Locking Devices. If they are low-released mass, contained, fail-safe (Rivets and Fasteners) or low risk (Fasteners). The best approach for fasteners is to try to make them fail-safe as they have small areas and are usually under high stresses and it's very difficult to pass safe-life analysis (Sarafin 1995, 394).

- Shatterable Components and Structures. Differentiating between internal and external with different requirements.

- Rotating Hardware. In (ECSS 2009*a*) rotation machinery is considered with some differences.

- Sealed Containers. It has requirements such as not containing hazardous material and being a leakage before burst component.

- Tools, Mechanisms and Tethers.

- Batteries. They should meet the NHLBB and sealed container definition mentioned earlier.

2.2.2.4 Fracture critical parts/items

Parts classified as fracture critical require risk mitigation activities, providing assurance that flaw sensitivity is understood considering flaw screening, qualification and acceptance testing and material parameters. A part is classified as fracture critical if there is no doubt or concern that it is not fracture critical. They receive more attention than other items, consisting in a set of activities to reduce the risk of failure due to a crack, damage tolerance assessments to show life requirements are met, screening, traceability, material requirements and documentation of the assessment and hardware implementation process.

They can be classified as:

- non-metallic PFCI unless fail safe, low-risk fracture or contained.
- Metallic PFCI which require better NDE than standard NDE
- Pressure vessels. It requires that it contains 19307 (19210 for ECSS) J of stored energy or greater based on adiabatic expansion of a perfect gas, stores a gas that will experience and maximum design pressure greater than 690 kPa or a liquid or gas in excess of 103 kPa that would create a catastrophic hazard if released. They are usually made of titanium or aluminium alloys, for example Ti6Al4V is compatible with hydrazine and aluminium with liquid oxygen (Dunn 2016, p. 31).
- Pressurised structures. It requires that is a pressure shell of a manned module, a manned pressurised structure designed with the criterion of leak before burst and safe life to leakage or similar conditions as the pressure vessel.
- PFCI which require a maintenance periodic inspection in order to achieve the required life (FLLIs as mentioned earlier).
- Rotating machinery with a minimum kinetic energy and angular momentum.

2.2.3 Analysis and Fracture Mechanics

All fracture critical parts need activities performed to understand the sensitivity to cracks and damages. These are explained through sections 7 in (ECSS 2009*a*) and 7.3 and 7.4 in (NASA 2016). The approaches for metallic and composites or bounded parts is different as linear elastic theory and others is not adequate for composites as mentioned in 2.1. For that reason composites have a much complex approach which will not be addressed as the main scope of this project is metallic components.

In order to enable crack growth prediction and critical-size calculation the following data is necessary:

- Critical failure mode identified
- Service-life profile
- Stress distribution and load spectra.
- Material properties
- Critical initial crack size in the worst orientation and location based on screening method implemented

- Stress intensity factor solutions

There are two different approaches, demonstrating a margin on the required lifetime and crack size based on initial crack sizes and as alternative the critical initial defect (CID) size can be calculated iteratively with the condition that it can survive four times the required service life.

In order to perform the crack growth calculations, it is necessary to follow the steps mentioned in the technical standards, for example in section 7.2.8 of **??**

For the critical crack-size a_c calculations, LEFM should be used. In some cases this theory won't be valid, as when the material has a non-linear plastic behaviour, being better to used EPFM methods.

2.2.4 Quality assurance and inspections

Each technical standard specifies additional requirements for the material selection process of fracture critical parts. They require that materials are selected and controlled from trust sources inside the same organization. It is necessary to include all materials usage agreement in the FCSR. Moreover traceability of potential fracture critical parts (PFCIs) is required to make sure that materials used in the manufacture has the same properties used in the analysis and verification tests and so that structural hardware is manufactured and inspected in conformance with the requirements for the fracture control programme. Additional verification using fatigue analysis and testing is required for safe life and fail save items with defects with sizes larger than the acceptance criteria used in the manufacturing.

2.2.4.1 Inspections and flaw screening

All fracture critical items should be inspected in order to consider acceptable their release to service. There are numerous techniques. NASA standards forwards to another technical standard (NASA-STD-5009, NASA (2008)) with information on NDE while ECSS describes the inspections to perform in this standard. For non-metallic materials there are no NDE standards available due to the diversity of elements used, so the approach should be reproduced in the FCP in order to demonstrate its validity. For metallic components in the standard level of NDE, one of the following industrial technique can be used:

- Fluorescent penetrant
- X-ray
- Magnetic particle
- Ultrasonic
- Eddy current

The initial crack sizes and geometries are defined in Table 1 for the previous techniques mentioned. When special improved NDI techniques have to be used (as described in 2.2.2.2), the validity and confidence should be demonstrated.

Table 1: Initial crack size summary, standard NDE (ECSS 2009a)

NDI method	Crack location	Part thickness t [mm]	Crack configuration number (see NOTE 1)	Crack type	Crack depth a [mm]	Crack length c [mm]
Eddy current NDI	Open surface	$t \leq 1{,}27$ $t > 1{,}27$	4 1, 3, 8	through surface	t 0,51 1,27	1,27 2,54 1,27
	Edge or hole	$t \leq 1{,}91$ $t > 1{,}91$	5, 9 2, 7	through corner	t 1,91	2,54 1,91
	Cylinder	N/A	10	surface	see NOTE 2	1,27
Penetrant NDI Sensitivity Level ≥3	Open surface	$t \leq 1{,}27$ $1{,}27 \leq t \leq 1{,}91$ $t > 1{,}91$	4 4 1, 3, 8	through through surface	t t 0,81 1,91	2,54 3,82 - t 4,05 1,91
	Edge or hole	$t \leq 2{,}50$ $t > 2{,}50$	5, 9 2, 7	through corner	t 2,54	2,54 2,54
	Cylinder	N/A	10	surface	see NOTE 2	1,91
Penetrant NDI of titanium alloys, welds and Sensitivity Level < 3 for all other materials	Open surface	$t \leq 3{,}0$ $t > 3{,}0$	4 1, 3, 8	through surface	t 3,00 1,50	3,00 3,00 7,50
	Edge or hole	$t \leq 3{,}0$ $t > 3{,}0$	5, 9 2, 7	through surface	t 3,00	3,00 3,00
	Cylinder	N/A	10	surface	see NOTE 2	3,00
Magnetic Particle NDI	Open surface	$t \leq 1{,}91$ $t > 1{,}91$	4 1, 3, 8	through surface	t 0,97 1,91	3,18 4,78 3,18
	Edge or hole	$t \leq 1{,}91$ $t > 1{,}91$	5, 9 2, 7	through corner	t 1,91	6,35 6,35
	Cylinder	N/A	10	surface	see NOTE 2	3,18
Radiographic NDI	Open surface	$0{,}63 \leq t \leq 2{,}72$ $t > 2{,}72$	1, 2, 3, 7, 8	surface	$0{,}7 \times t$ $0{,}7 \times t$	1,91 $0{,}7 \times t$
Ultrasonic NDI	Open surface	$t \geq 2{,}54$	1, 2, 3, 7, 8	surface	0,76 1,65	3,81 1,65

NOTE 1 The crack configuration numbers refer to the crack configurations shown in Figure 10-1, Figure 10-2 and Figure 10-3.

NOTE 2 For cylindrically shaped items (see Figure 10-3) the crack depth a can be derived from the crack length c of this table for a/c = 1,0 with the following formula:

$$a = r\left(1 + \tan\frac{c}{r} - \sec\frac{c}{r}\right)$$

Exception: fastener thread and fillets, see clause 8.8.

2.2.4.2 Material selection

Each technical standard specifies additional requirements for the material selection process of fracture critical parts. They require that materials are selected and controlled from trust sources inside the same organization. It is necessary to include all materials usage agreement in the FCSR.

2.2.4.3 Traceability

ECSS standards specify that all PFCI should be traceable while NASA standards requires it only for fracture critical parts and NFC composites or bonded parts. This is to make sure that materials used in the manufacture has the same properties used in the analysis and verification tests and so that structural hardware is manufactured and inspected in conformance with the requirements for the fracture control programme.

2.2.4.4 Detected defects

Safe life and fail safe items with defects with sizes larger than the acceptance criteria used in the manufacturing, 50% of the maximum allowed size detected by a NDE or 50% if the standard size of NDE for metallic materials have to be processed through additional verification requirements. These correspond to verifying if the defect is a crack-like flaw, and analysing or fatigue tested as appropriate for metallic or composite materials.

2.2.5 Reduced Fracture Control Programme

A reduced fracture control programme (RFCP) can be used in **??** when unmanned, single-mission, spacecraft and payloads, and ground segment equipment. In this case, PFCI are reduced and requirements for proof-testing of components such as rotating machinery, glass and non-metallic items other than composites, bonded and sandwich items are reduced.

2.2.6 Documentation

The documentation needed for the fracture control programme consists in the fracture control plan, the fracture control summary report containing the information to show fracture control compliance for all parts to the requirements in the FCP and Engineering Drawing or lists identifying the different fracture critical parts as it is essential to prove that the appropriate NDE, special handling, grain directions, serialization and traceability needs are implemented.

2.3 Other Considerations

2.3.1 Materials employed in Spacecraft and Launchers

Space industry requires extraordinary demands on properties, cost and efficiency of materials as it is highly related with safety and reliability considerations. The vast majority of materials used are carefully chosen from commercial alloys, polymers and ceramics that can be processed using well established techniques. This is mainly because contractors prefer to stick to trusted technology with a high level of readiness. Moreover, in order to validate new materials or processes many effort is necessary to produce quality documents, process controls and reliable data. Also, the quantities required for the spacecraft industry are of many orders of magnitude lower than the preferred by commercial materials producers which could mean inviable prices.

Primary and secondary structural elements are made normally from light alloys based on aluminium, magnesium, titanium and to a limited extent, beryllium. Nickel based super alloys are widely used for their high temperature applications and oxidation resistance. A review of different materials employed in the sector will be made in the following sections.

2.3.1.1 Metallic materials

Aluminium

Rocket structural materials are normally based on the Duralumin series of aluminium. This trade name is nowadays obsolete, and it usually refers to heat treated aluminium-copper alloys designated as the 2000 series. For example, Ariane IV has used aluminium alloys as the AA2024 (widely used in aircraft fuselage construction), AA7075 and AA7020 (Dunn 2016). Unfortunately, stress corrosion cracking (SCC), explained in 2.3.3, is of an important concern in these alloys. The attractiveness is that it is a very light weight metal, of relatively low-cost, that can be heat treated to high strength levels. Also, it is very easily fabricated and operates well from cryogenic temperatures to moderate temperatures.

The aluminium-copper series (2XXX) are used in damage tolerance application, while aluminium-zinc (7XXX) are used where higher strength is required (Campbell 2006). 2XXX alloys have slightly higher temperature capabilities. Impurities such as iron and silicon are a concerned, but improvements in processing have reduced these improving toughness and better resistance to fatigue crack imitation and growth. Also, the improved aging heat treatments for 7XXX alloys have improved fracture toughness and greatly reduced SCC susceptibility with a minimum impact on strength.

Other type of alloys are also used as the aluminium-silicon-magnesium series (6XXX) (Campbell 2006), with a high SCC resistance and aluminium-lithium of the 8XXX series (although, due to its little lithium content it is usually categorised in other series as the 2XXX). The later one is very attractive as addition of lithium increases the modulus and reduces density. The drawbacks in the current development of this material are excessive anisotropy, lower desired properties, delamination and low SCC threshold. This drawbacks are trying to be circumvented in the new generation of alloys, making components made of this material an interesting candidate for fracture analysis. For example, Aluminium-lithium alloy 2195 was used for the fuel tank on the Space Shuttle (Campbell 2006, p. 31). .

Titanium

Titanium can be used to save weight by replacing steel alloys and super alloys where temperatures permit it. It is also used instead of aluminium when temperatures exceed aluminium's capabilities or where fatigue and corrosion is a problem. It has a high resistance to fatigue, high temperature capabilities and good resistance to corrosion (Campbell 2006). The alpha-beta alloy Ti-6Al-4V alloy is widely used although new stronger ones are starting to replace it. One of its drawbacks is its limited weldability due to its two-phase microstructure but have the best balance of mechanical properties (Campbell 2006, 188). It is also very amenable to superplastic forming and can be combined with diffusion bonding to produce complex structures.

In the fracture control standards explained in 2.2, titanium alloy fasteners cannot be used in safe life applications because of its generic environmental assisted and sustained load cracking. They normally require assessments that need to be approved by the corresponding authority.

Super alloys

Super alloys are widely used in this space industry, due to their high strength, good corrosion resistance and good fatigue and creep resistance. They are especially necessary for high temperature applications as in the propulsion system. As a general, they include nickel, iron-nickel and cobalt based alloys. Nickel alloys normally contains additions of molybdenum and niobium to form a solid solution hardening and formation of Ni_3Nb, a hardening precipitate (Dunn 2016). This is known as Inconel 718, which was used for the propulsion system of the Space Shuttle (Wayne et al. 2011).

Beryllium and Magnesium

Both of this alloys are extremely lightweight materials although they have serious drawbacks which have to be considered. Magnesium is normally not that strong as aluminium, however they are much lighter, being used normally for structural parts. The biggest negative apsect is there poor corrosion resistance and sublimation problems after long vacuum exposure so it requires adequate plating and chemical conversion coating (Dunn 2016). Beryllium on the other side has very good mechanical properties but its manufacturing is very expensive. Also its powder and dust are toxic, being of a big concern for the fabrication, ground handling and manned space missions. Usually they are considered FCIs as beryllium is a brittle material (Sarafin 1995, 394).

2.3.1.2 Non-metallic materials

Composites and metal and ceramic matrix composites are used extensively in the space industry. Spacecraft uses plastic for components, elastomers for propellant diaphragms, ceramic for optical mirrors and composites for an exhaustive list of applications. Composites are light weight and can be optimised for strength and stiffness, improved fatigue life, corrosion resistance and reduced assembly costs due to fewer detailed parts and integrally co-cured structures. High strength fibre composites, especially carbon fibres offer a significant advantage over other aerospace metallic alloys. The number one deterrent is their costs due to its difficult manufacturing and design. In fracture control planning, significant additional requirements are imposed to components made of this materials. They primarily suffer from outgassing, a very complex phenomenon similar to that of sublimation of metals and alloys in space.

2.3.2 The effect of the Environment

Spacecrafts operate in a complex environment outside Earth's protective shield, the atmosphere, which provides familiarity in design. They go through different phases during their whole mission. Although they spend the majority of their time in space, the manufacturing process, ground testing, transportation, launch and possibly an atmospheric re-entry and operation in a complete different environment of another planet should also be noted as the complete success depends on their capacity to withstand all these environments. It is important to consider all loading events for safe life analysis, in order for crack-growth analysis (not so for fatigue analysis).

Engineers from every field have to work together to collaborate in order to prevent or solve problems related with this failure modes. Problems associated with launcher and spacecraft integration need an interdisciplinary approach where materials engineers play a vital role. It should be remembered the Space Shuttle Challenger accident in 1986 caused by not considering during the design the environmental effects of an elastomeric seal (Dunn 2016, p. 61).

Ground environment

It is, however, the ground environment one of the biggest hazards for a space mission (Griffin & French 2004). Its associated failures are far more common than flight failures (which in fact are normally traced back to ground environments) (Sarafin 1995, p. 57). The main concerns of this environment are particle contamination, which can reduce degrade materials and mechanisms, chemical contamination, which includes corrosion (explained in section 2.3.3) and water absorption and is caused by the atmosphere, processing residue and non-compatibilities of some material and, finally, electrostatic charging which can potentially destroy some electronic components. Ground testing involves high number of loading events contribute to fatigue and fracture damage (Sarafin 1995, p. 395). Also, ground transportation should be taken into account.

Space environment

The space environment is also of big concern. It englobes vacuum effects, thermal radiation, charged-particle radiation, atomic and molecular particles, micrometeoroids and debris, magnetic fields and gravitational fields. The first two and their effect in crack-propagation are explained in the following paragraphs as they are of big concern for fracture control.

Vacuum has a big effect on material properties and structures. As spacecraft structures are manufactured in Earth's ambient pressure, special care has to be taken with sealed structures, venting close spaces or designing the structures to withstand the extra pressure. Outgassing and desorption are also of big concern for polymer based materials. Outgas is caused by the release of organic constituents or previously absorbed gases, which can further condensate in critical parts. Desorption is caused by previous water absorption, causing the structure to contract and further contaminate other critical surfaces.

Moreover, impacts from micrometeorites and debris has to be considered as the material experiences a condition of dynamic rapid loading (Liu 2005), where the fracture toughness can vary.

Due to the different types of thermal radiation (solar, albedo, planetary, and spacecraft components), thermal effects are also of a concern for the designers. Satellite orbits therefore cause different thermal environments in the spacecraft producing continuous cyclic loads (as materials contract when temperature decreases and expand when it increases) which can cause structural failure or reduce structural life due to fatigue damage. Moreover these different temperatures affect material properties such as ductility and strength. This effects can be creep fatigue, thermal fatigue and thermo-mechanical fatigue (when mechanical cycles are also involved) (Liu 2005, p. 150).

2.3.3 Corrosion effects

It is important to note that although spacecraft spend the majority of their service lives in a dry vacuum environment (space), corrosion is quite common in some materials, especially in aluminium and magnesium environments. As mentioned earlier 2.3.2, it is produced due to the ground environment exposure to manufacturing debris and water vapour. It involves fretting caused by the breakdown of a protective oxide layer causing cracks to appear sooner, galvanic attack due to electron movements from one metal to another, hydrogen embrittlement and stress-corrosion cracking (Sarafin 1995, p. 59). They are of important consideration for the fracture control analysis.

Hydrogen embrittlement is caused by atomic hydrogen diffusing into a metallic material making it susceptible to a brittle fracture. Titanium and steel alloys are of big concern, specially the former one as it shows a strong micro-structural dependence (Liu 2005, p. 114). It is of special importance for the propulsion system components as hydrogen is normally used as a rocket fuel. The Space Shuttle Main Engine criteria for selecting fracture critical parts included components made of Inconel 718 (A nickel based super alloy explained in 2.3.1) exposed to gaseous hydrogen (Wayne et al. 2011, p. 298), such as the combustion chamber (Jewett & Halchak 1991). It is a complex phenomenon with a high dependency on temperature which goes beyond the scope of this project.

On the other side, stress-corrosion cracking (SCC) is caused by the growth of a crack formed by an intergranular corrosion pit, while the material is under tensile stress leading to a brittle fracture. This tensile stress causing SCC can be just caused by residual tensile stresses, affecting fatigue life (Liu 2005, p. 97). Materials may need a high resistance to it as it is subjected to an atmospheric environment during the ground and launch phases. Table 2.3 by Dunn (2016) and table 2.4 by Liu (2005) show some of these materials and ECSS standards require materials from table 5-1 of ECSS-Q-ST-70-36 (ECSS 2009*b*) for fracture critical components.

2.4 Some fracture critical components

Table 2 shows some potential fracture critical components.

Component	Material	Reason
Falcon 9 and Space Shuttle Aluminium-lithium propellent tanks. (Dunn 2016, p. 36) and (Campbell 2006, p. 31)	Al-Li	Pressure vessel (Clause 8.2.2 of (ECSS 2009*a*)). Reusable so subjected to many launch and ground handling cycles (probably has fatigue problems).
Space Shuttle main engine combustion chamber. (Wayne et al. 2011)	Inconel 718	Due to potential hydrogen embrittlement and acoustic fatigue. Reusable so subjected to many launch and ground handling cycles (probably has fatigue problems).
Pressure shell of Node-3 (Tranquility) of the ISS. (European Space Agency n.d.*b*)	Al 2219 - T851	Pressurised structure, pressure shell of a manned module (Clause 8.2.3.1 a.1. of (ECSS 2009*a*))
Dome and its skirt of Cupola, installed in Node-3 of the ISS. (European Space Agency n.d.*a*)	Al 2219 - T851	Pressurised structure, pressure shell of a manned module (Clause 8.2.3.1 a.1. of (ECSS 2009*a*))
Shutters to protect windows of Cupola, installed in Node-3 of the ISS. (European Space Agency n.d.*a*)	Al 7075 - T7352 and Al 6061 - T6	Probably fracture critical as has to sustain micrometeorite and debris impacts (could cause a catastrophic event), specially for the leading windows and thermal cycles.
Solar cell arrays structure. (NASA 1971)	Al Honeycomb	Important structural item, if it fails, the power system is compromised. Probably fracture critical as has to sustain many thermal cycles.
Planetary rover's wheels. (Baker 2012) and (Baseda et al. n.d.)	Al 6061	Probably fracture critical as it has to sustain a great number of load cycles.

Table 2: List of fracture critical components

2.5 Specific analysed materials

The components from Table 2 analysed are the following.

2.5.1 Aluminium-lithium propellant tanks

Aluminium-lithium propellant tanks are starting to be employed in the space industry. Some advantages, aside from its excellent material properties, is that it can be friction stir welded and is demisable. It is lightweight and has lower density than typical titanium alloy. It's a very critical part as it has been historically associated to crack growth and leaks due to porosity formed because of welding aluminium (Wayne et al. 2011, p. 281).

For example, Falcon 9 uses this kind of propellant tank in its first stage, manufactured by means of friction stir welding (Dunn 2016, p. 36). Due to the importance of RLV kind of vehicles it is interesting to perform a fracture control analysis on this component.

Due to limited data of the Falcon 9 exact aluminium alloy, the aluminium-lithium alloy 2195, used for the fuel tank of the Space Shuttle (Campbell 2006, p. 31) will be analysed (another RLV vehicle). This alloy is considered 30% stronger and 5% less dense (NASA 2005) than the original 2219 alloy used, analysed with the second component. Its mechanical properties are shown in Table 3.

Alloy	E	σ_Y	σ_{Ult}	ν	Reference
Al 2195	72.8 GPa	580 MPa	602 MPa	0.33	Firrao & Doglione (2001)

Table 3: Al 2195 mechanical properties

2.5.2 Pressure shell of Node-3 of the ISS

It is interesting to analyse Al 2219 - T851 as it is used as a pressurised structure in various ISS modules as in the Node-3 (European Space Agency n.d.*b*) and its observatory module (European Space Agency n.d.*a*). Aluminium-copper alloys are widely used through the space industry. They are heat treatable to higher strengths by precipitation hardening and some are weldable (usually they are mechanically joined, but they can be joined also by friction stir welding). While they have moderate yield strengths, they possess very good resistance to fatigue crack growth and good fracture toughness (Campbell 2006). Its mechanical properties are shown in Table 4.

Alloy	E	σ_Y	σ_{Ult}	ν	Reference
Al 2219	73.1 GPa	352 MPa	455 MPa	0.33	CRP MECCANICA S.r.l. (n.d.)

Table 4: Al 2219 mechanical properties

2.5.3 Planteary rover's wheels

Another interesting to analysis to perform is of the alloy 6061-T6 a aluminium-silicon-magnesium alloy which is typically used for planetary rover's wheels (Baker (2012) and Baseda et al. (n.d.)) as it combines relative high strength and high resistance to corrosion (Matweb n.d.). Its mechanical properties are shown in Table 5. The properties are taken

Alloy	E	σ_Y	σ_{Ult}	ν	Reference
Al 6061	$68.9\,GPa$	$283\,MPa$	$324\,MPa$	0.33	Matweb (n.d.)

Table 5: Al 6061 mechanical properties

at $-28°C$ corresponding to low operation temperatures that the rover could experience on Mars (Baseda et al. n.d.).

Chapter 3. Methodology

To analyse the identified components, ABAQUS programme will be used to model crack propagation obtaining primary data. The analysis will be simplified by using 2D and 3D plates with the same materials of the components. Therefore, a comparison between both materials will be performed.

Throughout this section, the theory behind the fracture control programmes and Abaqus simulations based in structural life analysis and finite element modelling will be explained in 3.1. Procedures followed to obtain the results will be also developed in section 3.3.

3.1 Structural life analysis

The importance of structural life analysis has been explained in chapter 2. To explain the analysis performed in a fracture control programme, this section will explain the theoretical background in a similar way as Sarafin (1995). These fields are based on the effect of stress concentrations caused by cracks subjected to cyclic loadings, such as space mechanisms operating continually (structures loaded by high-frequency and random vibrations), reusable space structures and parts sensitive to on-orbit dynamic and thermal stresses.

It is important to first distinguish between fatigue damage and fatigue failure. Fatigue damage is the material gradual degradation after a number of cyclic loadings, causing small cracks to form due to damage caused near microscopic defects. On the other hand, fatigue failure is when the part ruptures due to crack growth caused by the fatigue damage after a number of cycles.

Current used approaches for assessing structural life divides the estimation of total life in fatigue crack initiation and propagation to final failure (Liu 2005, p. 138). Crack initiation uses fatigue technology to forecast crack initiation life and crack propagation life is predicted with fracture mechanics.

Fatigue analysis is used in fracture control programmes to analyse some components. It is based on empirical data and considers the life of the material until failure at a particular stress level without assuming any initial crack or how fast they grow. Fracture mechanics is more theoretical and assumes a known crack, which is used by fracture control programmes by assuming it in the worst possible location and orientation as previously explained in section 2.1 and predicting its crack growth. Empirical data is also used in combination with the theory to predict when the crack growth will become unstable causing a fatigue failure. It is therefore more conservative as it also depends on more factors such as the loading sequence and assumes the worst case scenario.

3.1.1 Fatigue loading

Fatigue loading can be of constant amplitude or of variable amplitude (spectrum loading). In fatigue testing, loads cycles are usually applied in the form of sawtooth or sinusoidal (Liu 2005).

The loads that a spacecraft structure will experience throughout its life depend on the scenario and the structure category. According to Griffin & French (2004), primary structures experience only few events such as proof loading, transportation and launch transients while tertiary structures are subjected only to random vibrations during ground tests and launch.

There are some important parameters and terms that should be considered for fatigue loading. A reversal occurs when a load changes direction.

- **Load reversal** occurs when a load changes direction (every maximum or minimum peak)
- **Stress ratio** or load ratio which is the ratio of minimum stress (load) by maximum stress (load) Equation [3.1] (-1 for reversible loads).

$$R = \frac{\sigma_{min}}{\sigma_{max}} \tag{3.1}$$

- **Stress range**

$$\Delta\sigma = \sigma_{max} - \sigma_{min} \tag{3.2}$$

- **Mean stress**. Positive mean stresses reduce fatigue strength (tensile), while negative mean stresses increases it (Liu 2005, p. 153).

$$\sigma_m = (\sigma_{max} + \sigma_{min})/2 \tag{3.3}$$

- **Stress amplitude**

$$\sigma_a = (\sigma_{max} - \sigma_{min})/2 \tag{3.4}$$

- **Amplitude ratio**

$$A = \frac{\sigma_a}{\sigma_m} \tag{3.5}$$

3.1.2 Fatigue analysis

Fatigue is an event caused by repeated or varying stresses, with a value lower than the material's ultimate tensile strength, which leads to fracture whether it behaves in a cyclic softening or hardening manner.

Fatigue designed data can be determined empirically from cyclic load tests of standard specimens and represent it with a S-N curve showing maximum stress versus the number of cycles to failure, an example of which showing aluminium-lithium alloys (8090 and 2090) and aluminium-zinc alloys can be seen in Figure 4.

The number of loading cycles to failure depends on the peak stress σ_{max}, stress concentration factor K_t, which represents the steep stress gradients, and the stress ratio.

This curve is therefore a scatter plot representing typical fatigue life instead of a lower-bound, introducing the necessity of scatter factors as mentioned earlier which increment the expected number of loading cycles. It has been seen that fracture control programmes use a scatter factor of 4 service life cycles. Other scatter factor techniques can be employed when the data is not available for the appropriate R such as a Goodman diagram or the appropriate K_t by multiplying the gross-section stress by it. Although this approaches are conservative but not enough when considering fretting explained in 2.3.3.

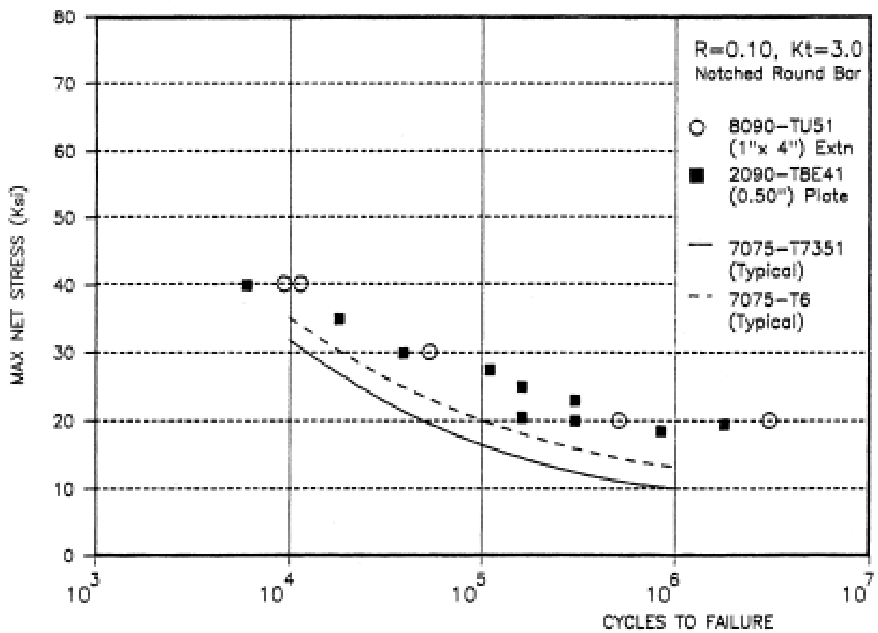


Fig. A7.10 Comparison of *S-N* curves for 8090-TU51 extrusion with 2090-T8E41 plate and typical 7075-T6/T7351 aluminum, *R* = 0.1, K_t = 3. Source: Ref A7.11

Figure 4: S-N curve of Aluminium alloys (Liu 2005)

S-N curves become asymptotic to the fatigue endurance limit, where at such stress level the material sustains infinite cycles, although many materials such as aluminium don't have a real one (Sarafin 1995, p. 285).

Other means for presenting data sometimes are necessary, such as the constant life diagrams representing stress amplitude vs means stress, or against maximum stress for a constant number of cycles

Fatigue phenomena can therefore be divided into high-cycle fatigue, where low stresses are present, and low-cycle fatigue (Liu 2005). The last one is typically concerned with significant cyclic plasticity being present.

Spacecraft components usually work above the material endurance limit, so a cumulative fatigue damage should be employed such as the Miner's rule mentioned in ECSS (2009*a*) for fail-safe PFCI. This method uses a loading spectrum with different load cycles identified and computed a cumulative fatigue-damage ratio (Equation [3.6]), making sure that it stays below 1 (above, a fatigue failure occurs) even when design scatter ratios are multiplied (a factor of 4 life cycles employed in FCP). m represents the number of loads, n represents the number of cycles of a load value, and N the corresponding number of cycles from a S-N curve. However it does not account for load sequence effects

$$D = \sum_{i=1}^{m} \frac{n_i}{N_i} \, (\times 4 \quad \text{service lives as scatter factor}) \qquad [3.6]$$

Factors such as the specimen size, loading condition, load transfers, local geometry, corrosion and temperature should be accounted when performing this damage cumulative damages. For example, for a complex structure, a fatigue quality index is employed to account for complex geometries and loadings which is multiplied by the far-field stress. Also, a stress severity factor can be employed which additionally accounts for variations in material properties, product quality and other analytical uncertainties (Liu 2005, p. 146).

There are other ways to assess fatigue strength, such as the strain based $\epsilon - N$ curves which according to Liu (2005) account for load sequences causing residual stresses and improves cumulative damage assessment as plastic and elastic deformations are shown.

3.1.3 Fracture mechanics

Fracture mechanics incorporates fracture toughness and stress analysis of cracks (Liu 2005) and en-globes fields such as linear elastic fracture mechanics (LEFM), elastic plastic fracture mechanics (EPM) and time-dependent fracture mechanics. When applied stresses are more than 80% yield strength, LEFM becomes inaccurate as the plastic zone at the crack tip becomes bigger, bringing non-linearities and so, EPM should be used to describe the behaviour (Figure 5). As spacecraft industry relies more in LEFM theory as limit stresses are normally below 80% (Sarafin 1995, p. 293) of yield stresses, this section will be centered in it. The analysis is focused in isotropic materials.

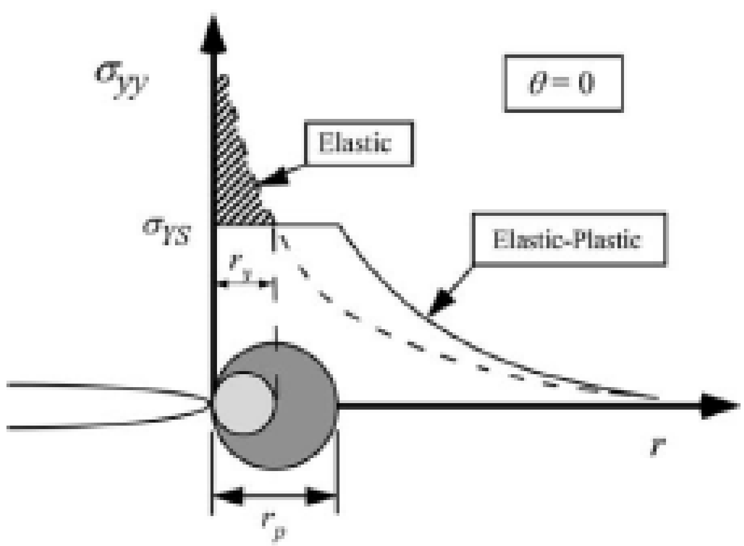


Figure 5: Elastic and Elastic-Plastic field representation (Zenóglio de Oliveira 2013)

LEFM theory relates stress σ and cracks size a, assuming stress is proportional to strain ϵ (elastic region) to obtain the stress intensity factor K. The crack is loaded in one or more of the three basic modes (Figure 6), being the first one loaded normal to the crack and the other two due to shear stress. Mode I is by far the most common (Sarafin 1995, p. 288), so it will be the one explained in this section and analysed, being its stress intensity factor K_I. Some problems may have more than one mode presented on a crack (McNary 2009).

It was first derived for brittle materials such as glass by A. A. Griffith in 1924 to explain that when a specific stress and crack size was applied, the crack would become unstable in an infinite wide plate with a crack fully through and orientated perpendicular to the uniaxial

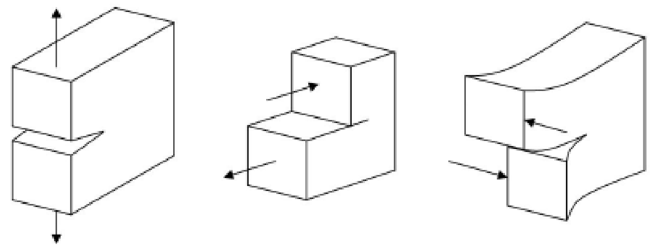


Figure 6: Loading modes (McNary 2009)

tensile stress, but was later adjusted by Irwin and Orowan so it would apply to other materials (including ductile):

$$\sigma_c = \sqrt{\frac{2E\,(T+p)}{\pi a}} \qquad [3.7]$$

Where E is the elasticity modulus, T is the surface energy per unit area of the cracked surface and p is the work done in plastic deformation near the crack tip and a accounts for crack size.

A more general approach was later developed using elasticity theory to calculate stress fields. As the crack can be assumed elliptical (Schijve 2009), the exact solution can be obtained for an infinite wide sheet (Equation [3.8]) with a tension stress loading σ.

$$\begin{aligned}
\sigma_x &= \frac{\sigma\sqrt{\pi a}}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left(1-\sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right) - S \\
\sigma_y &= \frac{\sigma\sqrt{\pi a}}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left(1+\sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right) \\
\tau_{xy} &= \frac{\sigma\sqrt{\pi a}}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\sin\frac{\theta}{2}\sin\frac{3\theta}{2} \\
\text{with} \quad & \sigma_z = 0 \quad \text{for plane stress} \\
\text{and} \quad & \sigma_z = \nu\,(\sigma_x+\sigma_y) \quad \text{for plane strain}
\end{aligned} \qquad [3.8]$$

In the vicinity of the crack tip however. the equation can be used to approximate the stress distribution as stress and strain become independent of the part geometry. So, the theoretical stress distribution of Mode I in the perpendicular direction from a distance r from the crack tip is:

$$\sigma_{y,\theta=0} = \frac{K_I}{\sqrt{2\pi r}} \qquad [3.9]$$

It predicts infinite stress at the crack tip, but in reality a plastic zone at the tip keeps it finite. As this region violates the LEFM theory, a region of K dominance is defined at a determined small distance to the crack, where the stress intensity factor governs the field.

The mode-I stress intensity factor (SIF) is given by Equation [3.11]. It can be multiplied by a factor M account for different stresses (bending), widths, geometry and cracks, being 1 for an infinite width plate and through cracks. The other modes intensity factors are K_{II} and K_{III}. Stress intensity factor definition can be changed when the analysed part is composed of two materials at each side of the crack (McNary 2009), as its also determined by the strain discontinuity at the interface behaving in a more complex way beyond the scope of this project.

$$K_I = \sigma\sqrt{\pi a} \qquad [3.10]$$

The plane strain fracture toughness K_{Ic} is the value of K_I which leads to unstable crack growth. It is obtained by combining Equation [3.11] and Equation [3.7]:

$$K_{Ic} = M\sigma_c\sqrt{\pi a} = M\sqrt{2E\left(T+p\right)} \qquad [3.11]$$

This value refers to thick sections as a state of plane strain (triaxial stress) exists, where the surrounding material constraints the crack tip and keeps the plastic zone small, absorbing less energy by plastic deformation and more by crack growth. Thinner regions, on the other side, experience a plane stress (two-dimensional stress) where more energy is absorbed by plastic deformation and therefore fracture toughness increases. The adjusted for thickness critical stress intensity factor is K_c.

Energy methods can also be employed to categorise crack size by studying the Irwin energy release rate G shown by the Griffith energy balance (Mohammadi 2008).

Crack will grow only when subjected to cyclic loading under normal conditions, but when in a corrosive environment as explained in 2.3.3, even when subjected to a constant load (residual stresses or preload) the critical fracture toughness can be significantly lower than the K_{Ic}, being the stress-corrosion cracking threshold K_{Iscc}.

Paris law

Test data on the other side gives fracture properties and curve-fit parameters for crack growth rates. They have shown that a small crack subjected to high stress will grow almost at the same rate as large cracks at low stress. A crack-growth curve for different aluminium alloys can be seen in Figure 7, showing $\frac{\mathrm{d}a}{\mathrm{d}N}$ which is the change in crack size per cycle versus ΔK which is the difference between the maximum intensity factor and the minimum.

There are two stages:

- **Stage I**, where ΔK is small and cracks grow very little per cycle. At ΔK_{th} cracks don't grow. It varies with R, being ΔK_0 when $R = 0$. In this stage, cracks nucleate and coalesce by slip-plane fracture extending inward at almost 45º to the stress axis (Liu 2005, p. 131), through crystallographic planes with greatest alternating shear stress. It occurs over a limited region near the initiation site.

- **Stage II**, where the crack growth appear linear in a logarithmic scale, following the Paris equation where C and n depend on the material, although some advanced softwares don't use this simplification and try to fit all stages using complex equations (Sarafin 1995, p. 292). Here, a transition from the 45º grow occurs to planes normal to the alternating tensile stress.

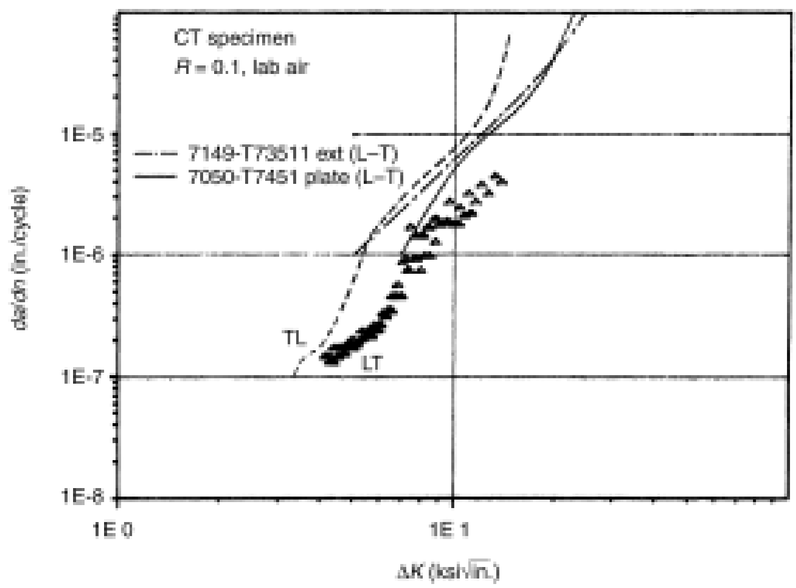


Fig. A7.11 Comparison of da/dN curves for 8090-TU51 extrusion (LT and TL) with 7149-T73511 extrusion and 7050-T7451 plate, R = 0.1. Source: Ref A7.11

Figure 7: Crack-growth curve of Aluminium alloys (Liu 2005)

$$\frac{\mathrm{d}a}{\mathrm{d}N} = C(\Delta K_{th})^n \qquad [3.12]$$

This stage is favoured in materials that exhibit planar slip and some alloys strengthened by coherent precipitates (such as age-hardened aluminum, extensively used in the space sector) with cycles of low stresses(Liu 2005). For the same reason, alloys with bigger grain size such as nickel and cobalt superalloys (for adequate high-temperature resistance) experience little of this propagation stage.

Environmental factors and the microstructure has little effect on this stage unless the stress intensity factors ΔK is low enough, exposing small cracks for extended times to material heterogeneities and environment.

- **Stage III**, where high values of ΔK cause the curve to become nonlinear, with the crack growth becoming unstable at K_c.

This curve (figure 3.12) depends on the crack orientation, material thickness, temperature, environment and shifts with different R ratios.

3.2 Abaqus modelling

It is typical to automate structural life analysis by using computer software such as NASGRO, ESACRACK or finite-element software such as ABAQUS (the one employed for the project) to determine more accurate stress intensities.

The programmes analyse different crack types, geometries and sizes and using a loading spectrum, automatically calculates stress intensity fields in order to iteratively calculate crack-growth using the appropriate material values. It continuously checks if K_{max} overcomes the critical values (K_{Ic} and K_{Iscc}) or if net-section stress overcomes material's yield stress. Therefore, it can compute the critical crack size for a given part and condition.

A brief summary of how Abaqus/CAE crack simulation works will be performed in this section.

3.2.1 FEM simulations

Standard finite element modeling (FEM) requires a use of extremely refined meshes around regions of crack propagations and voids. This adds computational cost when performing crack growth analysis such as in fracture mechanics and fatigue fields. Displacements in finite element modeling (Equation [3.13]) are described by shape functions $N_i(x)$, and the displacements at every node u_i, being n the total number of nodes and x the integration point.

$$u(x) = \sum_{i=1}^{n} N_i(x) u_i \qquad [3.13]$$

FEM analysis works by assembling the global stiffness and force matrices and solving it to obtain nodal displacements and forces and stresses and strains at each element.

3.2.2 XFEM modeling

Extended finite element methods (XFEM) will be used to simulate crack growth in this project, as it simplifies and reduces the steps required as there is no need to re-mesh near the crack-tip and produces better results according to Hedayati & Vahedi (2014). It works by considering more degrees of freedom reducing analysis cost (McNary 2009) by implementing the partition of unity principle.

Therefore, two functions are added to describe the presence of a crack, saying that they enrich the displacement function Equation [3.14] (Hedayati & Vahedi 2014). These are; a discontinuous function for the crack interior which uses a modified Heaviside function (positive or negative depending at which side of the crack is the integration point)(Abaqus 2014), and an asymptotic function describing the crack tip. The number of nodes cut by the crack surface are denoted by n_Γ and those cut by the tip by n_Λ. a_j and b_k^α describe the enriched degrees of freedom, with α describing the associated elastic asymptotic crack-tip function $F_\alpha(x)$.

$$u(x) = \sum_{i=1}^{n} N_i(x) \left[u_i + H(x) a_j + \sum_{\alpha=1}^{4} F_\alpha(x) b_k^\alpha \right] \qquad [3.14]$$

These extra degrees of freedom added are implemented in the forces and stiffness matrices augmenting their size. During the simulation, some geometric sub-routines determine which elements are affected by the crack interior or tip and partitioned if so.

It is important to note that the XFEM technique is not unique for fracture mechanics analysis, as its ability of modeling moving boundaries allows it to be applied in other fields such as two-phase fluids studies (Cheesa & Belytschiko 2003).

3.3 Simulation procedure

The process used to perform a LEFM crack growth simulation of the identified materials using ABAQUS will be described in the following section. In order to study crack growth of an unstable cracked specimen, a static overload fracture will be studied. This allows to analyse a surface crack growth during the final fracture moments, when cracks unstably grow through the material entire width causing the specimen to fail.

3.3.1 2-D SENT Aluminium plates

Fatigue cracks are typically initially developed near the surface, where nominal stresses can be higher as when bending and geometric variations cause stress concentrations (Liu 2005, p. 128). These variations can be caused by machining, surface flaws, notches, oxygen-enriched surface phases, etc. In this analysis, a surface crack will be simulated by computing the crack growth of a crack initiated at the edge of a 2-D plate. The type of specimen employed is called Single Edge Notched Tension plate (SENT). This will give an insight to material toughness to crack growth. The material's ultimate strength will be used as damage tolerance indicator stating it as the maximum principal stress. Loads are implemented as boundary displacements at the top and lower edges.

Geometry
The geometry analysed is a rectangular shell part of 30 $\times$ 30 mm with and edge crack of 5 mm long horizontally in the middle of the specimen. Therefore, the initial crack size by width ratio is of $a/w = 0.166$. The M factor for this specimen is given in Equation [3.15] by Mohammadi (2008):

$$M(a) = 1.12 - 0.23\left(\frac{a}{W}\right) + 10.56\left(\frac{a}{W}\right)^2 - 21.74\left(\frac{a}{W}\right)^3 + 30.42\left(\frac{a}{W}\right)^4 \qquad [3.15]$$

Mesh
In order to achieve mesh independence, simulations with different global mesh sizes have been carried out. As seen in Figure 8, a global mesh size of 0.4 is sufficient to represent adequate results.

Results
Results are post-processed using Abaqus to obtain crack sizes and visual data and with the matlab toolbox Abaqus2Matlab (Papazafeiropoulos et al. 2017) to obtain forces applied and other useful data.

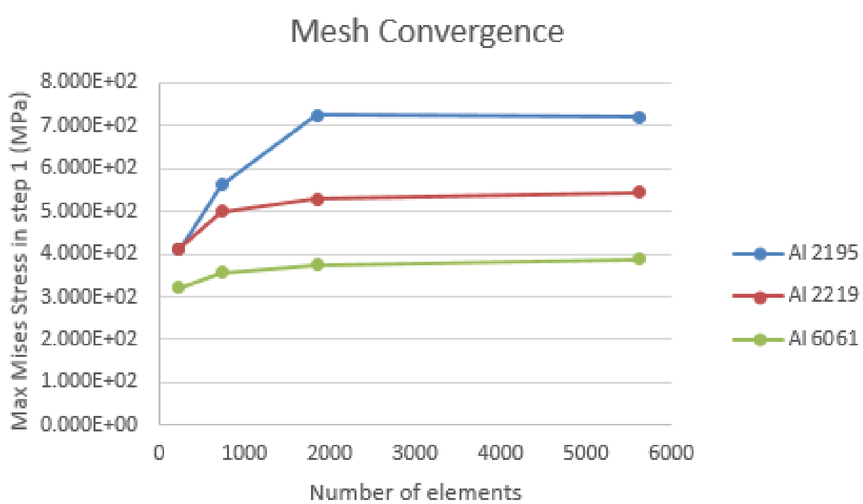


Figure 8: Mesh convergence for the different materials

Chapter 4. Discussion of Results

4.1 Aluminium-lithium alloy

For the Aluminium-lithium 2195 alloy, Figure 9 and Figure 10 show the crack growth development and the Mises stresses, at the beginning of the crack propagation and at the moment of fracture. The first figure shows a typical butterfly zone of high stress near the crack tip (Liu 2005), which could depict a plastic region if the simulation took plasticity into account. It can be seen that when the crack starts to propagate, there is a Mises stress of 719.6 MPa at the crack tip.

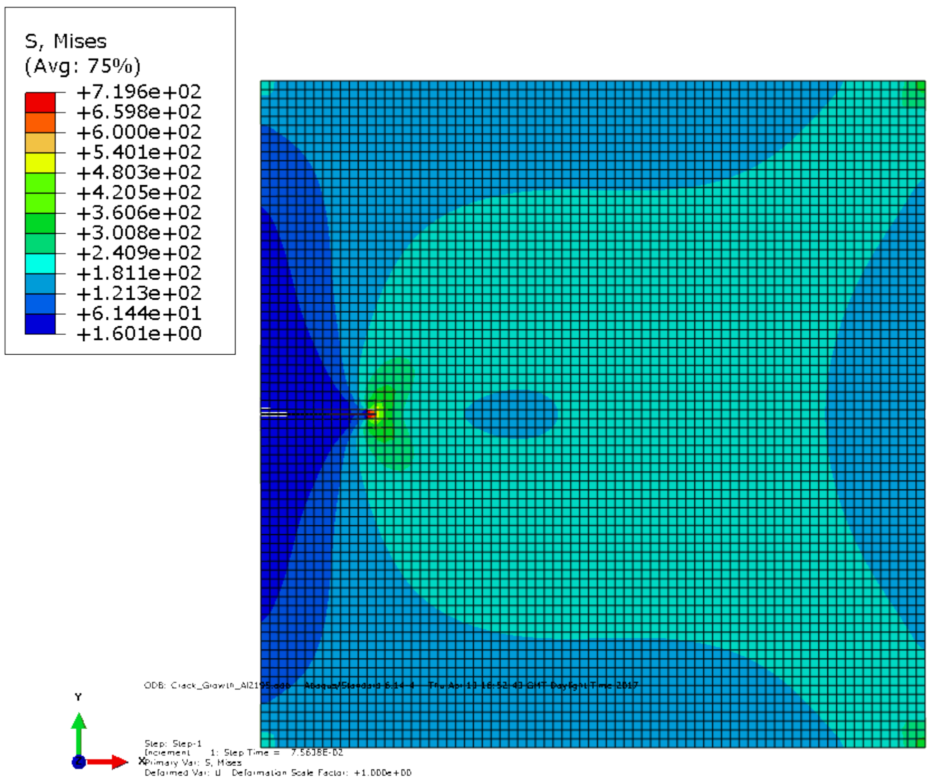


Figure 9: Al 2195, crack initiation, Mises stresses

Regarding the last fracture moment, the simulation shows higher maximum Mises stresses throughout the model, but this is not reliable as it can be seen that they are located at the right corners (Figure 10), probably because of the boundary conditions imposed. Either way, from this last time frame, it can be seen that the specimen has failed 0.2807 *seconds* after the crack propagation started.

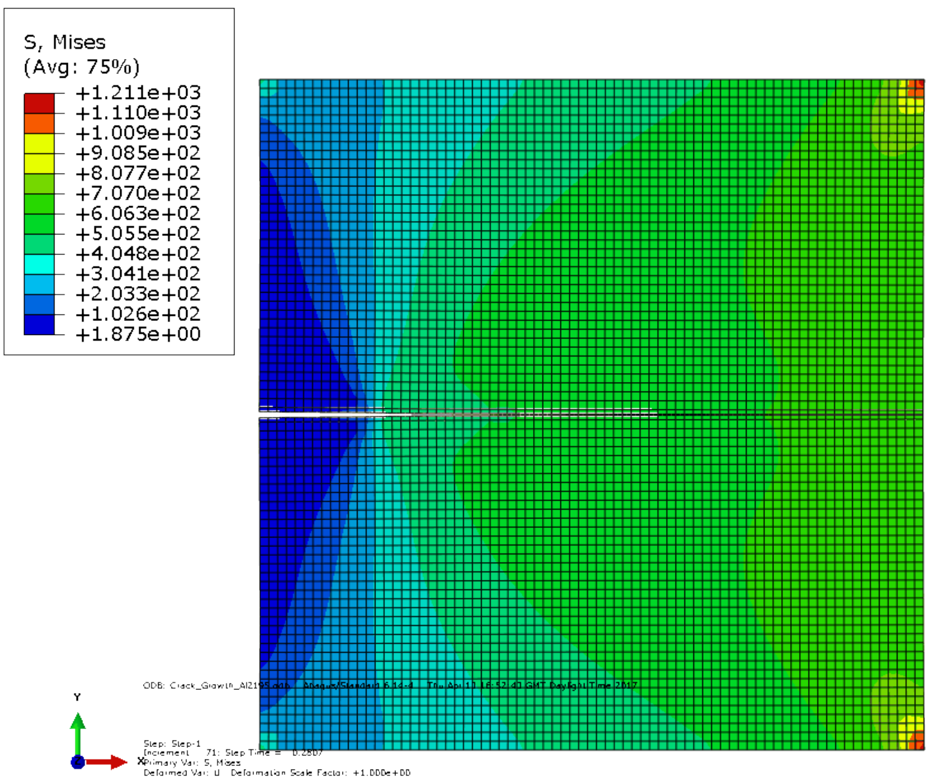


Figure 10: Al 2195, failure, Mises stresses

4.2 Aluminium-copper alloy

For the Aluminium-copper 2219 alloy, Figure 11 and Figure 12 show the crack growth development and the Mises stresses, at the beginning of the crack propagation and at the moment of fracture. Here the mises stress at the beginning of the crack propagation at the crack tip is of 543.9 MPa, a much lower value than the previous result, as the ultimate stress is much lower (Table 4 and Table 3)

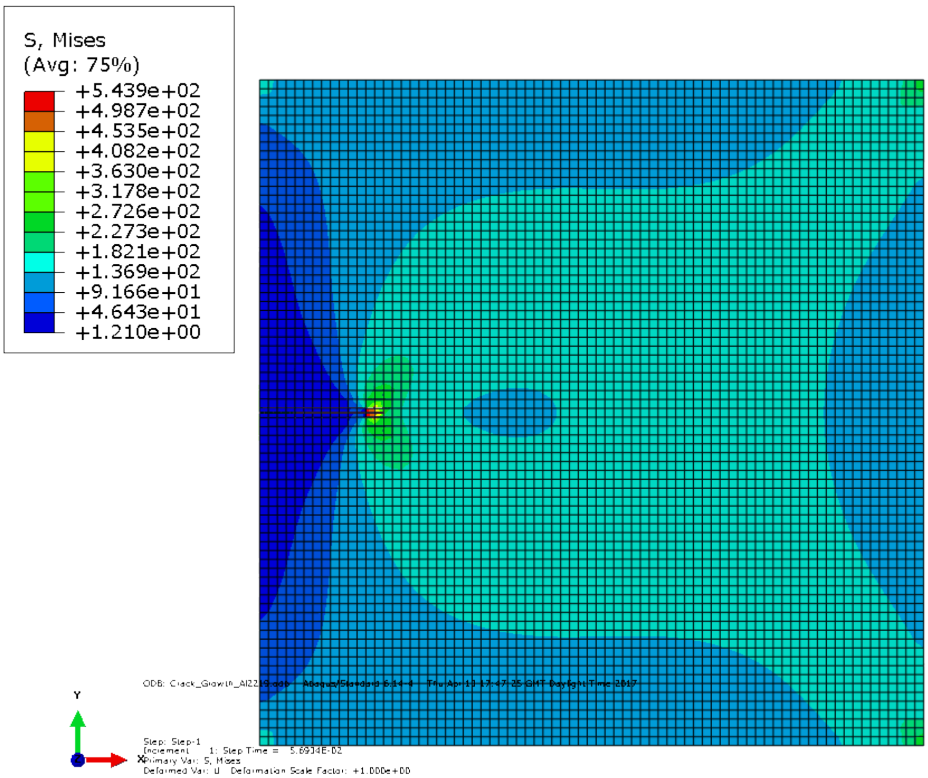


Figure 11: Al 2219, crack initiation Mises stresses

The same effect from the boundary conditions can be seen in Figure 12. Additionally it can be seen that the specimen has failed 0.2096 *seconds* after the crack propagation started, which is earlier than with aluminium-lithium alloy.

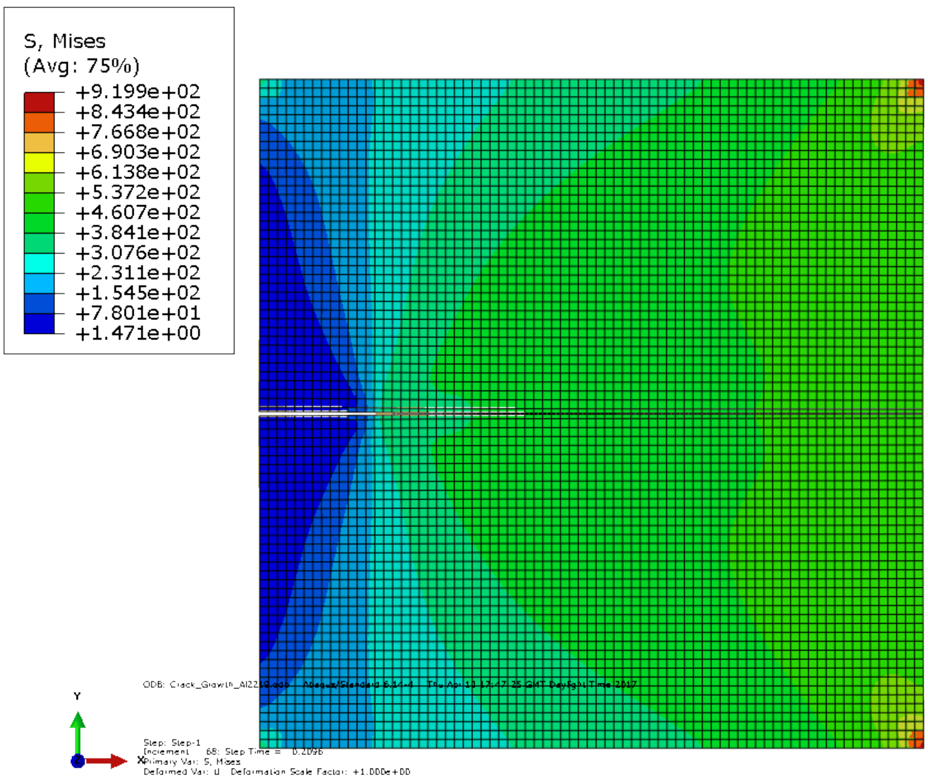


Figure 12: Al 2219, failure, Mises stresses

4.3 Aluminium-silicon-magnesium alloy

For the Aluminium-silicon-magnesium 6061 alloy, Figure 13 and Figure 14 show the crack growth development and the Mises stresses. Here the mises stress at the beginning of the crack propagation at the crack tip is of 387.3 MPa.

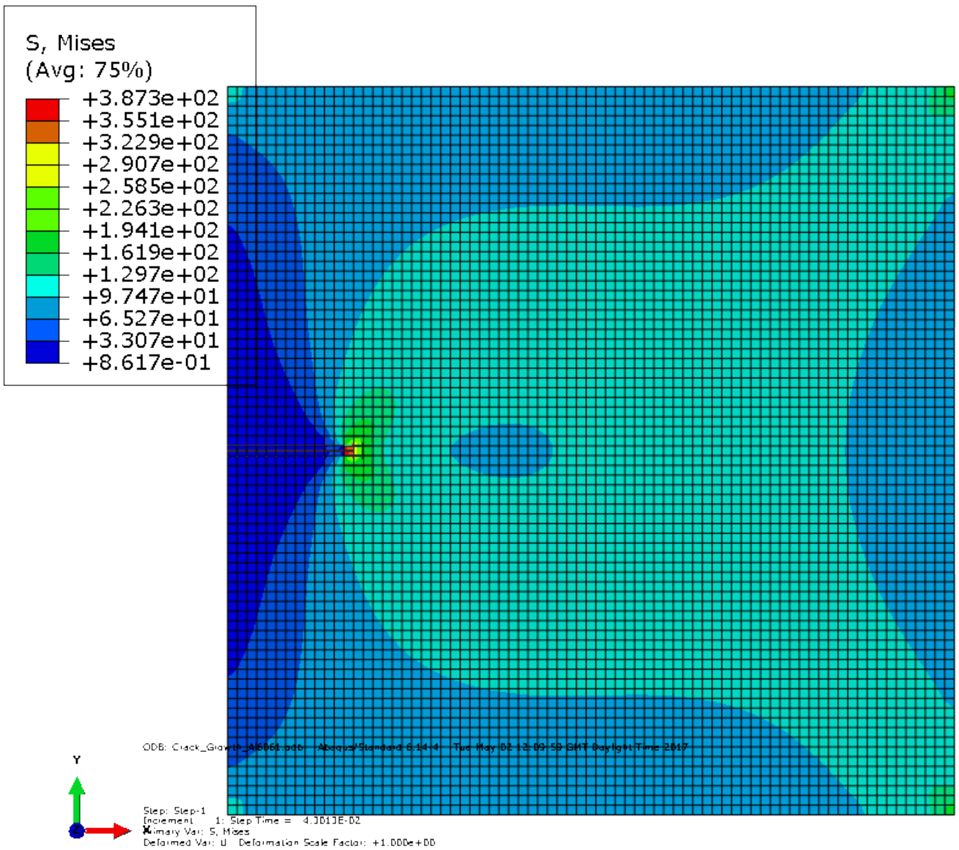


Figure 13: Al 6061, crack initiation Mises stresses

Additionally it can be seen that the specimen has failed 0.1604 *seconds* after the crack propagation started which is much earlier than in the previous materials.

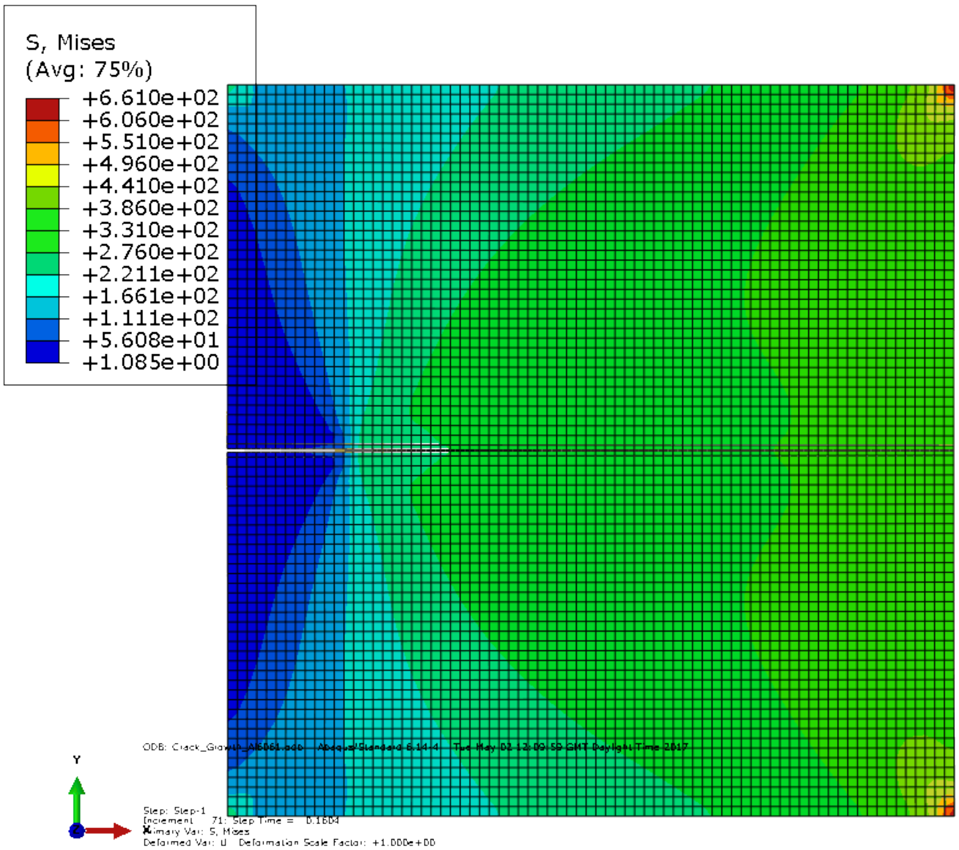


Figure 14: Al 6061, failure, Mises stresses

4.4 Comparison

Table 6 shows the different results for the different materials simulated. It can be seen how the aluminium 6061 plate fails much faster than the other materials, under a lower Mises stress surrounding the crack tip, probably because of its lower ultimate strength. Different parameters are compared, when the maximum force is applied, at at crack initiation and failure of the specimen. These values are specified which circles in the following graphs.

Alloy	Al 2195	Al 2219	Al 6061
Maximum Mises Stress at crack initiation (MPa)	719.6	543.9	387.3
t_0 (s)	0.0756	0.0569	0.0430
t_{Fail} (s)	0.2807	0.2096	0.1604
t^* (s)	0.205	0.1527	0.1174
Force applied at initiation (N)	10417.92	7874.01	5606.98
Force applied at failure (N)	29246.04	22460.67	16206.21
Maximum force applied (N)	29617.67	22569.10	16240.16
Crack size at maximum force applied (mm)	26.8	28.0	28.8

Table 6: Comparison of Results

Moreover, Figure 15 shows how the corresponding cracks grow during each time step. It can be seen that the crack starts to grow much later in aluminium 2195, while in aluminium 6061 the crack starts growing much earlier.

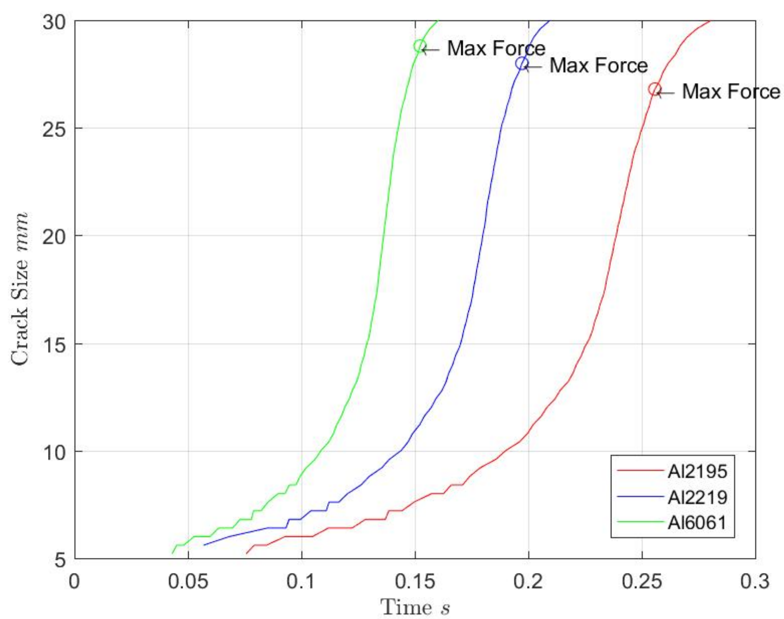


Figure 15: Crack size at each time step

From a first look, it seems that the crack growth speeds is bigger the lower the ultimate strength of the material is, as the slope observed in aluminium 6061 seems bigger than the one of aluminium 2195. This can be seen approximately by to the crack propagation time, as it takes longer for the aluminium 2195 specimen to fail (t^*) since the crack starts propagating (t_0), until it fails at t_{Fail}. Moreover, it seems necessary to find a correlation between crack initiation and another factor to explain the further difference in the material's lives.

Figure 16 shows the force applied for each time step for each material. The data is recorded after crack has started to grow. It can be seen that the crack starts to propagate when bigger loads are applied in aluminium 2195, while in 6061 it starts with a smaller load. The difference in initial slopes could be related to the modulus of elasticity of the materials, being a lower slope for aluminium 6061 as it deforms more for the same load as its modulus of elasticity is lower.

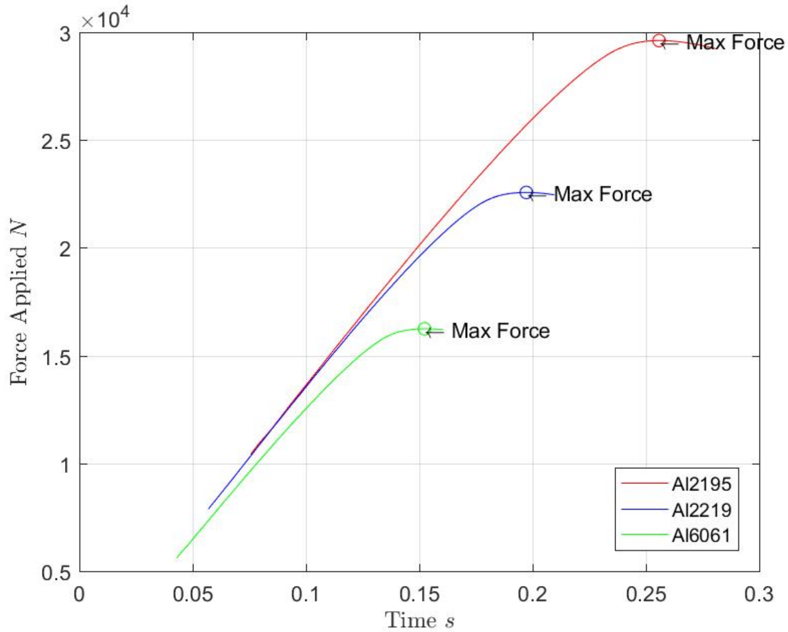


Figure 16: Force applied at each time step

Moreover a maximum force applied can be observed for each of the materials, after which the crack propagates under lower forces, possibly due to an unstable crack growth. This maximum force is bigger for the aluminium 2195 as it has the biggest ultimate strength compared with the rest.

The relation between force applied and crack size can be seen in Figure 17. It shows that the crack starts to propagate at a bigger force for aluminium 2195, and how more force is required for it to develop than for the rest of the materials. A mid section where almost the same maximum load is applied can be seen where almost all crack growth occurs, possibly because of unstable crack growth until failure.

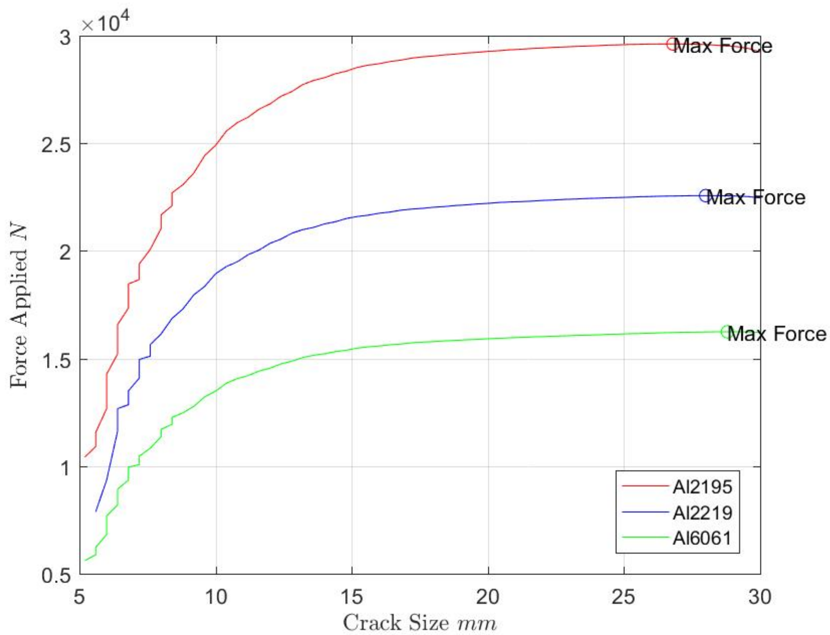


Figure 17: Force applied and crack size

Chapter 5. Conclusion

The previous results (section 4) show interesting fracture properties of the materials analysed. This properties can help in fatigue and fracture structural analysis of the components selected in this project in order to obtain accurate estimations of their service lives, as required for the fracture control programmes employed in the space industry.

It should be noted that, although aluminium 2195 show excellent properties in this fracture analysis compared with the other materials, a bunch of other factor should be taken into account in order to conduct a reliable analysis. For example, plasticity has not been taken into account for this analysis, where the fracture properties of this materials could dramatically changed. Also, as explained in section 2.3.2 and 3.1.1, the maximum loadings and the loading spectrum should be taken into account for this analysis by performing cyclic loading analysis. Additionally, the materials have been assumed homogeneous, but heterogeneities have to be taken into account as its variations in material properties, microstructures and thermal fluctuations affects crack propagation (Lengliné et al. 2011). This is especially true for alloys such as the materials considered, with different microstructures as they are conformed of different phases.

Environmental effects such as corrosion explained in 2.3.3 needs to be taken into account for the different applications of each material. For example, aluminium lithium alloys are susceptible to SCC (Balasubramaniam et al. 1991), specially when hydrogen is present by the formation brittle hydrides with composition of $LiAlH_4$. On the other side, the aluminium magnesium-silicon alloys have excellent resistance to corrosion as they contain less alloying elements (Busquets Mataix 2015), explaining why although its lower fracture properties shown in this project, it is preferred for use in rover's wheels in harsh environments such as Mars's atmosphere.

Regarding the aluminium copper alloy analysed, its results are acceptable compared with the other alloys, although it seems reasonable to consider future replacements for pressurised structures of habitable modules with the aluminium lithium alloys, 2195 or another one, in order to reduce weight (NASA 2005). Moreover, Friction Stir Welding (FSW) technology has emerged now days improving weld-ability of this kind of difficult to fusion weld alloys, becoming the preferred method for joining Al-Li space structures (Dunn 2016). Nonetheless, more fatigue analysis should be conducted for this relatively new alloys (aluminium-lithium) as aluminium copper alloys have good fatigue and damage tolerance properties (Campbell 2006).

Other parameters such as the stress intensity factors and critical crack sizes where considered for analysis in this project but were discarded as didn't efficiently helped in performing the required material comparisons.

This project has served to introduce the writer into the field of fracture mechanics and fatigue analysis applied to the space sector. It may be extended to conduct other type of fracture analysis such as in three dimensions considering realistic components and loads and by using other assessments such as the failure assessment diagrams to compare materials fracture properties. Moreover, the field of fatigue analysis could also be studied as it help in performing comparisons for the materials selected for different space missions based on more empirical data. The ultimate goal would be to perform comparisons using real spacecraft fracture control methodology in order to adequately predict spacecraft components service lives, optimising actual or future spacecraft components and systems.

Bibliography

Abaqus (2014), '*Abaqus/CAE User's Guide*', [online]. available from http://abaqus.software.polimi.it/v6.14/books/usi/default.htm [15 April 2017].

Baker, B. (2012), Reconfigurable wheels: Re-inventing the wheel for the next generation of planetary rovers, Master thesis, Massachusetts Institute of Technology.

Balasubramaniam, R., Duquette, D. J. & Rajan, K. (1991), 'On stress corrosion cracking in aluminum-lithium alloys', *Advances in Engineering Software. Vol 39, Issue 11* . available from https://www.sciencedirect.com/science/article/pii/095671519190075C [03 May 2017].

Baseda, D., Burany, J., Chapman, R., Martin, G., Murphy, K., Patton, C. & Saunders, K. (n.d.), Tracked multi-terrain exploratory vehicle (tramtev), Preliminary design review, National Space Concepts Union, Potential Advance Concepts Kickoff Group, Raleigh, North Carolina.

Busquets Mataix, D. J. (2015), '*4: Aleaciones de Aluminio*', [online lecture]. 11889 Aerospace Materials, Valencia, Universidad Politécnica de Valencia.

Campbell, F. C. (2006), *Manufacturing Technology for Aerospace Structural Materials*, 1st edn, Elsevier, London, UK.

CANNAE (n.d.), '*Space Freighter*', [online]. available from http://cannae.com/space-freighter/ [04 March 2017].

Cheesa, J. & Belytschiko, T. (2003), 'An extended finite element method for two-phase fluids', *Journal of Applied Mechanics* . available from http://appliedmechanics.asmedigitalcollection.asme.org/article.aspx?articleid=1414746 [15 April 2017].

CRP MECCANICA S.r.l. (n.d.), Aluminum 2219-t851, Material data, Via Cesare Della Chiesa 21 - 41126, Modena.

Dunn, B. D. (2016), *Materials and Processes: for Spacecraft and High Reliability Applications*, 1st edn, Springer-Praxis, Chichester, UK.

ECSS (2009*a*), *Space Engineering: Fracture Control*, Technical Standard ECSS–E–32–01C Rev. 1.

ECSS (2009*b*), *Space product assurance: Material selection for controlling stress-corrosion cracking*, Technical Standard ECSS–Q–ST–70–36C Rev. 1.

ESA & Fletcher, K. (2016), '*Space economy, ESA: Poweing European Growth*', [online]. available from http://esamultimedia.esa.int/multimedia/publications/Space_economy/offline/download.pdf [03 March 2017].

European Space Agency (n.d.*a*), Cupola, Document, Project: International Space Station ESA–HSO–COU–004 Rev. 2. available from http://wsn.spaceflight.esa.int/docs/Factsheets/4%20Cupola%20HR%20web.pdf [22 March 2017].

European Space Agency (n.d.*b*), Node–3 (tranquility), Document, Project: International Space Station ESA–HSO–COU-006 Rev. 2. available from http://wsn.spaceflight.esa.int/docs/Factsheets/6%20Node%203%20HR%20web.pdf [22 March 2017].

Firrao, D. & Doglione, R. (2001), 'K-r curves of thin 2195-t8 aluminium alloy plate'.

Griffin, M. D. & French, J. R. (2004), *Space Vehicle Design*, 2nd edn, AAIA, Washington, USA.

Hedayati, E. & Vahedi, M. (2014), 'Using extended finite element method for computation of the stress intensity factor, crack growth simulation and predicting fatigue crack growth in a slant-cracked plated of 6061-t651 aluminum', *World Journal of Mechanics* . available from http://wsn.spaceflight.esa.int/docs/Factsheets/6%20Node%203%20HR%20web.pdf [15 April 2017].

Jewett, R. P. & Halchak, J. A. (1991), 'The use of alloy 718 in the space shuttle main engine', *The Minerals, Metals and Materials Society* .

Lengliné, O., Toussaint, R., Schmittbuhl, J., J., E., Ampuero, J. P., Tallakstad, K. T., Santucci, S. & Maloy, J. K. (2011), 'Average crack front velocity during subcritical fracture propagation in a heterogeneous medium'.

Liu, A. F. (2005), *Mechanics and Mechanisms of Fracture: An Introduction*, 1st edn, ASM International.

Matweb (n.d.), '*Aluminum 6061-T6; 6061-T651*', [online]. available from http://www.matweb.com/search/datasheet_print.aspx?matguid=1b8c06d0ca7c456694c7777d9e10be5b [01 May 2017].

McNary, M. J. (2009), Implementation of the extended finite element method (xfem) in the abaqus software package, Master thesis, Georgia Institute of Technology.

Mohammadi, S. (2008), *Extended Finite Element Method: for Fracture Analysis of Structures*, 1st edn, Wiley-Blackwell.

NASA (1971), *Spacecraft solar cell array*, Design Criteria NASA SP-8074.

NASA (2005), 'Super lightweight external tank'.

NASA (2008), *Nondestructive evaluation requirements for fracture critical metallic components*, Technical Standard NASA-STD-5009.

NASA (2016), *Fracture Control Requirements for Spaceflight Hardware*, Technical Standard NASA-STD-5019A.

Papazafeiropoulos, G., MuñizCalvente, M. & Martínez-Pañeda, E. (2017), 'Abaqus2matlab: A suitable tool for finite element post-processing', *Advances in Engineering Software. Vol 105* . available from https://www.sciencedirect.com/science/article/pii/S0965997816306512 [10 April 2017].

Sarafin, T. P. (1995), *Spacecraft Structures and Mechanisms*, 2nd edn, Microcosm, Inc. and Kluwer Academic Publishers, California, USA.

Schijve, J. (2009), *Fatigue of Structures and Materials*, 2nd edn, Springer Netherlands.

SIA & The Tauri Group (2016), '*State of the Satellite Industry Report*', [online]. available from https://www.nasa.gov/directorates/heo/scan/engineering/technology/txt_accordion1.html [04 March 2017].

Space X (2015), '*Falcon 9 Launch Vehicle Payload User's Guide*'. Rev 2.

Upgraded Spacex Falcon 9.1.1 will launch 25% more than old Falcon 9 and bring price down to $4109 per kilogram to LEO (2013), [online]. available from http://www.nextbigfuture.com/2013/03/upgraded-spacex-falcon-911-will-launch.html [04 March 2017].

Wayne, H., Helen, L., Gail, C. & Kamlesh, L. (2011), *Wings in Orbit: Scientific and Engineering Legacies of the Space Shuttle, 1971-2010*, 1st edn, NASA, Washington, USA.

Zenóglio de Oliveira, F. (2013), 'Crack modelling with the extendend finitie element method'.

Appendices

Appendix I Resumed Log Book

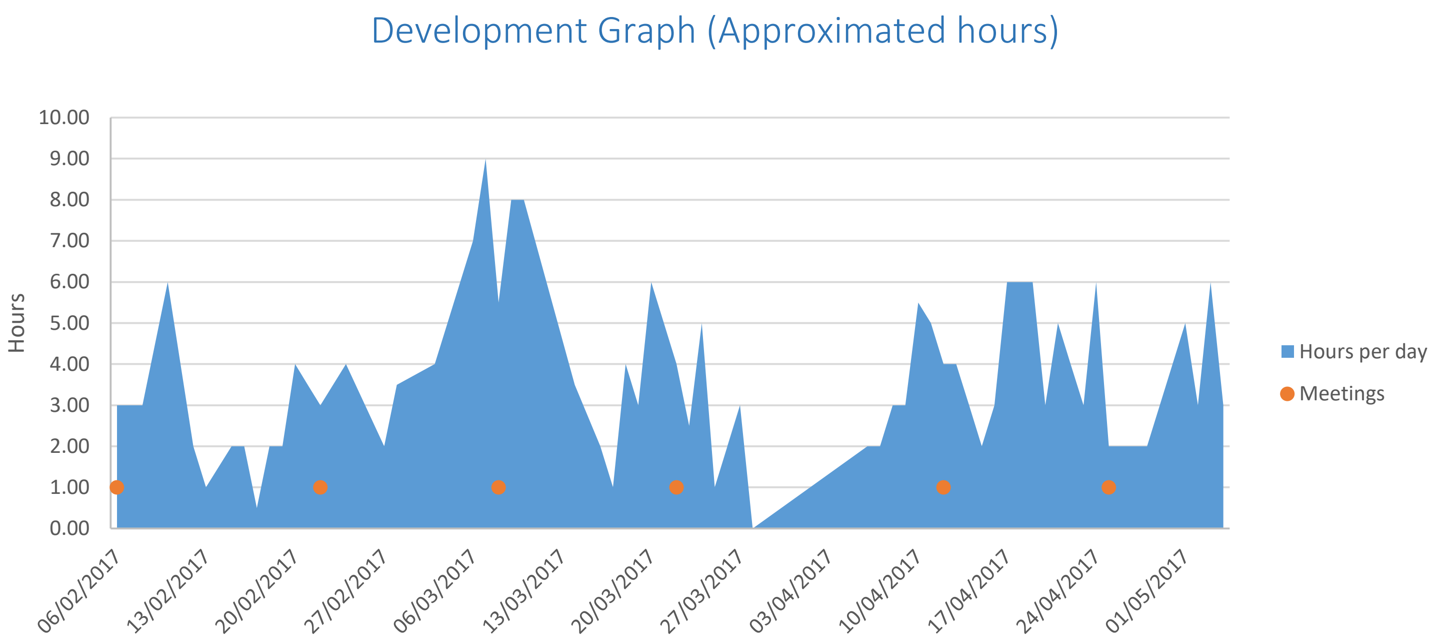
Resumed Log Book

Total hours	300

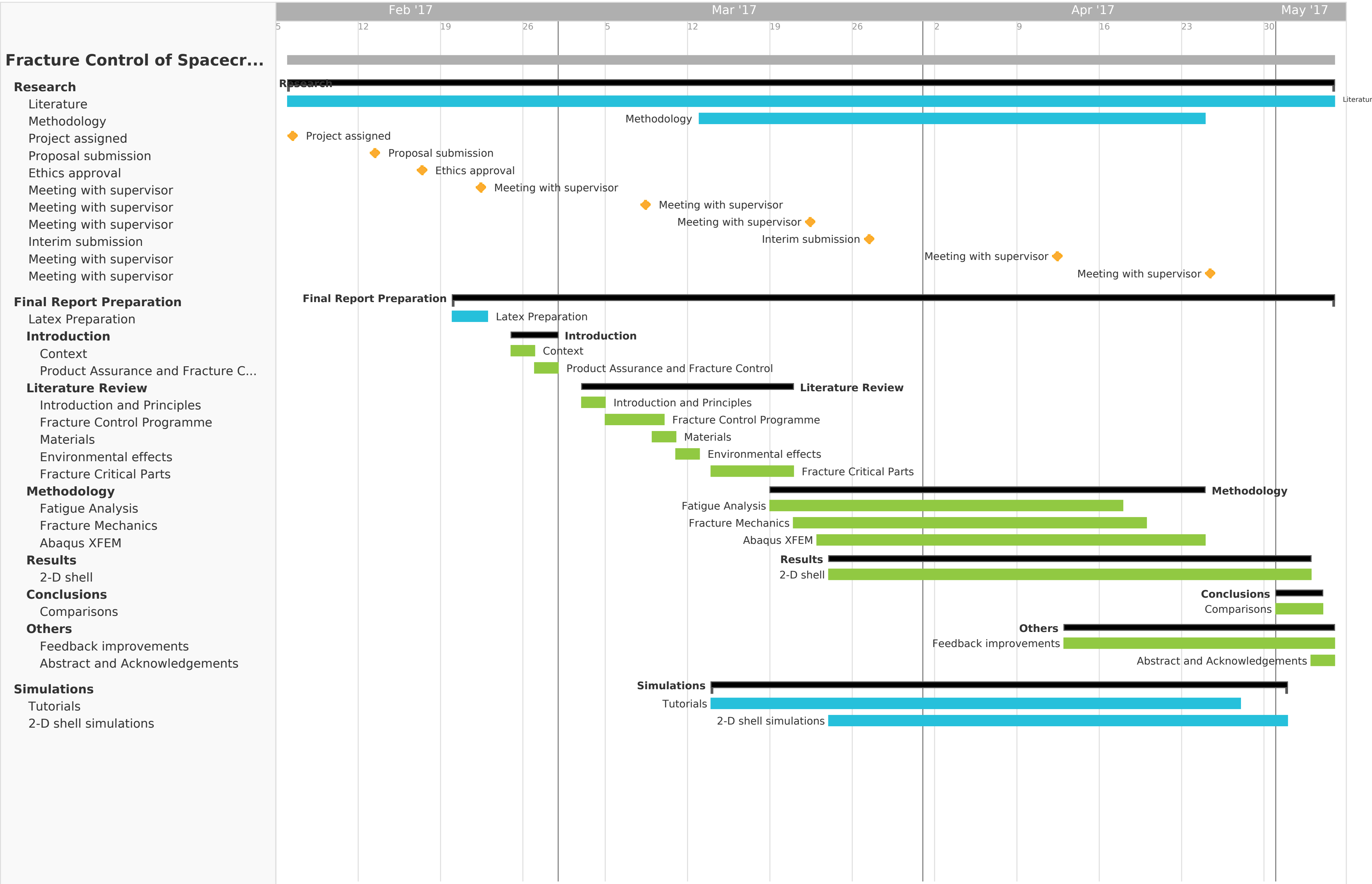
Date	Comment	Hours
06/02/2017	Supervisor assigned, project briefing. Research, set of books found.	3.00
10/02/2017	ECSS, NASA and JPL standards on fracture mechanics received. Questions on project development and objectives answered.	6.00
12/02/2017	Proposal finished.	2.00
13/02/2017	**Proposal submission.** Ethics developed.	1.00
17/02/2017	Ethics approval.	0.50
20/02/2017	Document development starts. Initial structure layout performed and bibliography prepared.	4.00
22/02/2017	Meeting with supervisor. Discussion about: How to identify the main components, probably the fuel and propellant tanks. Why titanium alloys are employed. Guide on how to develop the report. Some fundamentals of fracture control, why it is important in space industry (because of very thin structures employed).	3.00
24/02/2017	Extension Given.	4.00
27/02/2017	Half of introduction finished. Start to introduce product assurance programmes and fracture control.	2.00
28/02/2017	Development of introduction, new books and papers reviewed.	3.50
03/03/2017	Chapter 2, Literature review started.	4.00
06/03/2017	Some Components Identified, 1. Propellant tanks of aluminium lithium and 2. Combustion chamber of Inconel 718. Fracture control programme summary in development	7.00
08/03/2017	Meeting with supervisor. Discussion about: How to organise project. Methodology section; introduce linear elastic theory and link it to the project and software, explain methodology performed in Abaqus. Make a list of critical components for spacecraft. Component 1 is okay, but 2 not due to a super alloy being employed at very high temperatures, making analysis very complicated as hydrogen embrittlement is a very complex phenomena. 5. Okay to describe phenomena which affect quality and flaws such as SCC and hydrogen embrittlement. Summary of fracture control programme finished	5.50

09/03/2017	Explanation of materials employed. Notes of previous material subject revised. Revision of other sections developed.	8.00
10/03/2017	Material section finished, new components identified. Next days, explain the environments experienced and corrosion effects.	8.00
14/03/2017	Tutorials and recommended literature for methodology received by supervisor. Tutorials revised. Table of identified components and description of the selected ones started.	3.50
19/03/2017	Methodology section started, description of fatigue analysis developed.	3.00
20/03/2017	Previous sections revised. Finished fatigue analysis introduction, fracture mechanics introduction started	6.00
21/03/2017	Fracture mechanics section finished.	5.00
22/03/2017	Meeting with supervisor, all questions solved regarding theory and literature review. Instructions for a case analysis (simulation of a crack growth in a 2-D plate under simple cyclic tensile load of the materials identified)	4.00
23/03/2017	New component identified, others discarded. Tutorials revised.	2.50
25/03/2017	Interim report revised. Gantt chart started.	1.00
27/03/2017	Presentation finished.	3.00
28/03/2017	**Interim Review submission**	0.00
06/04/2017	Started more Fracture mechanics reading	2.00
08/04/2017	Some results obtained	3.00
10/04/2017	More simulations performed, improved results	5.50
11/04/2017	Mesh convergence checked	5.00
12/04/2017	Meeting with supervisor. Discussion about project feedback, simulation and results. Main focus now in static case with edge crack, evaluating crack growth vs stress applied and maximum stress. Stress intensity factor could be also obtained. For the different materials identified. Could the Griffith’s criterion be employed for material toughness? Or only for a middle crack?	4.00
13/04/2017	Feedback improvements started	4.00
17/04/2017	Other literature books started to help with feedback	3.00
20/04/2017	More results obtained by programming	3.00
24/04/2017	Methodology finished	6.00
25/04/2017	Meeting with supervisor. Scope of the project revised. Focus on obtaining crack length vs load applied on the 2D static case and compare both materials. A video tutorial on how to obtain them received.	2.00

27/04/2017	Tutorials finished, specific comparisons can now be made	2.00
01/05/2017	Simulations and results post processing finished	5.00
03/05/2017	Discussion of results finished	6.00
04/05/2017	Conclusions finished, materials comparisons fully complete	3.00
05/05/2017	Checking of some results and report optimisation finished. Gant chart finished.	3.00



Appendix II Gantt Chart


Feb '17
Mar '17
Apr '17
May '17
5
12
19
26
5
12
19
26
2
9
16
23
30
Fracture Control of Spacecr...
Research
Literature
Methodology
Project assigned
Proposal submission
Ethics approval
Meeting with supervisor
Meeting with supervisor
Meeting with supervisor
Interim submission
Meeting with supervisor
Meeting with supervisor
Final Report Preparation
Latex Preparation
Introduction
Context
Product Assurance and Fracture C...
Literature Review
Introduction and Principles
Fracture Control Programme
Materials
Environmental effects
Fracture Critical Parts
Methodology
Fatigue Analysis
Fracture Mechanics
Abaqus XFEM
Results
2-D shell
Conclusions
Comparisons
Others
Feedback improvements
Abstract and Acknowledgements
Simulations
Tutorials
2-D shell simulations
Research
Literature
Methodology
Project assigned
Proposal submission
Ethics approval
Meeting with supervisor
Meeting with supervisor
Meeting with supervisor
Interim submission
Meeting with supervisor
Meeting with supervisor
Final Report Preparation
Latex Preparation
Introduction
Context
Product Assurance and Fracture Control
Literature Review
Introduction and Principles
Fracture Control Programme
Materials
Environmental effects
Fracture Critical Parts
Methodology
Fatigue Analysis
Fracture Mechanics
Abaqus XFEM
Results
2-D shell
Conclusions
Comparisons
Others
Feedback improvements
Abstract and Acknowledgements
Simulations
Tutorials
2-D shell simulations

Appendix III Project Proposal

Final Year Undergraduate Project Proposal Form

Student Name	*Guillermo Joaquín Domínguez Calabuig*
Course	320EKM_1617JANMAY Project
Email	*domingu3@uni.coventry.ac.uk*
Project Module	320EKM / 330EKM

Supervisor	Kashif Khan

Project Title:
A proposed title for the project (Should be meaningful, relevant and concise)

Fracture control of spacecraft components

Synopsis:
Explain the background to the project, and provide an overview of what you intend to do (approximately 500 words)

The space industry´s main role in today's society is to contribute to attain a smart, sustainable and inclusive growth. It does so by contributing to scientific progress, which drives innovation by supplying other sectors with the knowledge acquired, and by targeting major issues as climate change, limited resources and health. Additionally, it plays an important role in every nation's security budget.

Throughout the last decade we have witnessed a steep increase in space commercialization, as many multinationals have expanded their business model to the space sector, especially in the communication and navigation sector. It seems that the exponential growth in technological advances worldwide will continue to encourage private space businesses into looking for new applications and opportunities.

It is then necessary to accompany this growth with some safety specifications, such as a fracture control programme which can assure mission success and protection against critical failures which could result in tremendous economical loses and, more *importantly, live loses. For this reason, NASA and other space agencies and organizations standardizes their fracture control specifications.*

The purpose is to assure the structural integrity of safety critical components from a usage failure due to mechanical loading, thermal loading and environmental influences which affect the propagation rate of pre-existing defects to critical sizes sufficient to cause a catastrophic failure. By and efficient fracture control, the risks of failures can be mitigated by establishing a safe interval of operation, providing adequate margin on the required service life and critical defect size in the structure.

The project will be focused on identifying and analysing the main components of satellites and spacecraft which require a fracture control criteria implementation. Moreover a study of the materials employed during the manufacturing of those components will be performed in order to analyse with a finite element analysis tool the damage mechanisms by which they can produce cracks.

Client:
Provide a description of your client (if any), and contact details.

Not applicable.

Objectives (provide from 5 to 8): *List the overall objectives of the project. These should be measurable, and will be used to assess the level of achievement of the project.*

- Review the different fracture control requirements specified for different space organisms.
- Identify the fracture prone satellite/spacecraft components
- Identify and analyse the loadings applied on these components during a space mission.
- Study the materials used to manufacture those components
- Identify and study the damage mechanisms by which those components can produce cracks.

Project Deliverables (provide from 5 to 8):
Provide a list of key deliverables of the project (which may be one for each of the above objectives). These can be studies, reports, recommendations, etc.

- Final report
- Interim review
- Summary of fracture control program
- Main components identification
- Fracture criteria
- Simulation files from the finite element analysis tool (FEA) from one or two main components

Why are you interested in the project?

Provide a reason for your interest, and describe what greater general interest it serves. Who else could benefit from it?

Back in my childhood, I had the typical young dream of becoming an astronaut. As I kept growing up, I was lucky to keep the dream of space exploration and rocket launches in my mind. For this reason, I have applied to Delft University of Technology (TUDelft) for the Space Flight track in the Master in Aerospace Engineering. By doing this project I get to work in a field I am very passionate about and which will help me in my future studies. Moreover I was very interested throughout my degree of Aerospace engineering in all the subjects related to the materials employed and structure analysis. With this project I get to apply all the knowledge in a specific sector.

Moreover it is very important to reduce launching and operating costs in order to improve space accessibility to different application and to improve space exploration. Materials are key for every space mission, and their requirements vary completely depending on the mission objectives. Therefore by studying their different properties and specifications one can optimize a mission reducing its costs dramatically. Also, new materials and processing techniques are always being developed and by understanding their necessary requirements new space applications can be discovered.

What are the key questions the project attempts to answer (provide from 1-3)?

- Which components of a spacecraft/satellite need a fracture control criteria implementation?
- Which materials are commonly used in spacecraft/satellites?
- What are the damage mechanisms which produce cracks?

How will you judge whether your project has been a success?

In my opinion the project will be a success if I understand the key aspects of a fracture control implementation as I will know how risk can be minimized in a space mission.
Also by acquiring an extensive knowledge on the materials employed in the space sector I will be able to obtain a wider view, key for every space programme. Moreover by performing the FEA analysis I will gain expertise on these type of software which is crucial in today's engineering.

What research methods do you intend to use?

A deductive approach will be employed in order to identify which components are in need of a fracture control criteria implementation using information from space organizations and the theory. Consequently a qualitative analysis of the damage mechanisms which conduct to cracks in satellites and spacecrafts components will be performed by simulations using a FEA software called ABAQUS.

What primary and/or secondary data sources do you intend to use?

I will work with secondary data extracted from different published materials such as journals, technical reports and standards. Also secondary information from different books will help to acquire the necessary theoretical background. Simulations will be performed to obtain primary data using models.

Estimate the number of hours you expect to spend on each of the major project tasks:
(The tasks below are only examples. You will need to edit the table to suit your own project).

Introduction	20
Objectives	15
Literature Review	100
Case Studies	40
Research	50
Final report preparation	50
Total number of hours	275

Signature:

Date: 13/02/2017

Appendix IV Ethics Approval

Certificate of Ethical Approval

Applicant:

Guillermo Dominguez Calabuig

Project Title:

Fracture control of spacecraft components

This is to certify that the above named applicant has completed the Coventry University Ethical Approval process and their project has been confirmed and approved as Low Risk

Date of approval:

17 February 2017

Project Reference Number:

P51644