

Evaluation of the Multiple Load Sequence methodology for GFRP fatigue life prediction

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Dissertation accepted in fulfilment of the requirements for the
degree *Magister Engineering* in Mechanical Engineering at the
North-West University

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Graduation: May 2022

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ABSTRACT

The current preferred static certification method used in the sailplane industry as defined by aviation governing bodies limits the allowable service life to 12 000 Flight Hours (Fh) which is problematic for current Australian sailplanes in the field that is approaching the service life boundary faster compared to European sailplanes (Patching & Wood, 1991, pp. 100-104; Van der Walt, 2020, p. 2). The current certification legislation allows for the use of a dynamic fatigue certification procedure to extend the service life past the 12 000 Fh barrier where a life factor of at least three is required to prove structural operational safety for the desired flight hour amount. (Pommerer, 2000, p. 3; M&D Flugzeugbau, 2020, p. 4).

The problem with the dynamic fatigue certification procedure is the extremely long certification test duration which can be a daunting task to perform. Alternative methods are henceforth required to decrease the costly dynamic fatigue testing for Fibre Reinforced Polymer (FRP) composite material sailplanes that require more than 12 000 Fh. (Van der Walt, 2020, pp. 2, 9-10).

Researchers such as Kensche (2002(a), pp. 32-43), Rodzewics (2009, pp. 452-460), Trappe (2010, pp. 94-99) and Van der Walt (2020, pp. 1-122) proposed alternative methods where a certain amount of Variable Amplitude (VA) load sequences could be exchanged with Constant Amplitude (CA) load sequences to determine the fatigue life of composite sailplane structures at an accelerated rate.

The proposed methods are uncertain about the correlation between CA and VA load sequencing at sailplane loading conditions. The research of Kensche (Kensche, 2002(a), pp. 32-43) initially hypothesized that the relationship between CA and VA load sequencing is linear, irrespective of the loading sequence.

Early research performed by Kensche (2002(a), p. 42), concluded that a certain amount of VA cycles may, in practice, interchange with a certain amount of CA cycles due to the equivalent damage state of the two spectrums. An exchange of the spectrums could result in sailplane manufacturers shortening the fatigue certification duration of sailplanes requiring a flight hour barrier exceeding 12000 Fh.

Research conducted by Kensche (2002(a), pp. 32-43) resulted in an empirical technique capable of comparing the different load spectrums to be proposed by Van der Walt (2020, pp. 1-122) for the extension of sailplane service life past the 12000 Fh barrier.

The research outcome of Van der Walt (2020, pp. 88-92) was that the Multiple Load Sequence (MLS) method showed promise and required further investigation.

Further investigation into the MLS method commenced with testing similar to that was done by Van der Walt (2020, pp. 1-122) with additional Residual Strength and Stiffness (RSS) testing for validation purposes of the MLS method. CA and VA fatigue testing was done at 6Hz for a coupon type specimen at 50, 65 and 75 % of the specimen's Ultimate Tensile Strength (UTS). The VA fatigue testing test interval quantities was increased compared to Van der Walt (2020) to achieve a more accurate MLS data curve.

The residual strength method was applied to determine the load-carrying capability of the specimen during the fatigue life cycle. The Residual stiffness method was used to determine the deformation resistance capability of the specimen subjected to loading during the testing cycle.

The RSS methods yielded equations that converged linearly as according to the PM assumption (Nijssen, 2007), whereas, the MLS methodology yielded a 4th degree polynomial equation which is not the linear function desired by Van der Walt (2020, p. 88), who obtained a quadratic equation with less testing interval points.

The outcome of the experimental testing yielded an estimated result of 22920 certifiable Fh compared to the current 12000 Fh for a UTS of 50 %. The obtained 22920 Fh translates to either 48605 CA or 3189674 VA cycles which is 5,7 KoSMOS II cycles for the coupon type specimen.

The evaluation concludes that although the MLS methodology equation was found to be non-linear, it is just as accurate compared to the validation and verification methods. There is henceforth merit in CA testing instead of the conventional VA testing for accelerated certification of sailplanes after an adequate Building Block Approach (BBA) database is established according to sailplane certification legislation requirements.

ACKNOWLEDGEMENTS

First and foremost, I would like to thank the following people for their guidance during this study:

- Prof A.S. Jonker
- Mr. L.P. Van der Walt

Secondly, I would like to thank the following entities without who the research conducted for this study would not have occurred:

- The CSIR
- Jonker Sailplanes
- The NWU Mechanical Engineering Department

Finally, I would like to thank the love of my life, Mrs. C. J. Senekal, for her love, motivation and support during the completion of this dissertation.

DEDICATION

This thesis is dedicated in loving memory to my father: F.P. Senekal, and my grandfather, F.P.P. Senekal, who both inspired me to study Mechanical Engineering.

KEYWORDS

Constant Amplitude, CS-22, Fatigue, Fibre Reinforced Polymer, Flight Hours, Multiple Load Sequence, Service Life and Variable Amplitude.

LIST OF ABBREVIATIONS

BBA	Building Block Approach
CA	Constant Amplitude
CCL	Combined Cycle Loading
CFRP	Carbon Fibre Reinforced Polymer
CS-22	Certification Specification 22
EASA	European Aviation Safety Agency
FAA	Federal Aviation Administration
FCP	Fatigue Compliance Procedure
Fh	Flight hours
FRP	Fibre Reinforced Polymer
GFRP	Glass Fibre Reinforced Polymer
LBA	<i>Luftfahrt-Bundesamt</i>
MLS	Multiple Load Sequence
PM	Palmgren-Miner
RCA	Residual Constant Amplitude
RSS	Residual Strength and Stiffness
UTS	Ultimate Tensile Strength
VA	Variable Amplitude

NOMENCLATURE

A	gauge cross-sectional area [mm ²]
D	damage number
F	failure force [kN]
i	cycle type
n	cycle count [#]
η_i	number of cycles of type i in the loading sequence [#]
N_i	expected number of cycles to failure for type i [#]
nCA	amount of CA cycles [#]
nRCA	amount of RCA cycles [#]
nVA	amount of VA cycles [#]
Q	Stress [kN/mm ²]
vc	the volume of the composite [m ³]
V_f	the fibre volume ratio [-]
vf	the volume of the fibres [m ³]

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1. INTRODUCTION

1.1. Background

Competitive soaring dates to the late 1800s according to the Soaring Society of America (2004, p. 4) with the development of sailplanes noticeably divided into several sub-eras. The historical sub-divisions are owed mainly due to unprecedented improvements in the design and manufacturing regulations governing the sport.

The current European Union certification, regulation and standardisation guidelines used for sailplanes are the Certification Specifications for Sailplanes and Powered Sailplanes (CS-22) which are based upon the Joint Aviation Requirement (JAR-22) design rules implemented when the European Union Aviation Safety Agency (EASA) obtained full functionality in 2008. (Horváth, 2011).

Type certified sailplane designs as defined by De Florio (2016, pp. 85-122) tend to challenge the barriers of the current sailplane certification legislation since these sailplanes are designed optimally for competitive reasons. (Torode, 2010). The current EASA CS-22 sailplane certification legislation requires that sailplane types based on previous verified designs require at least static certification testing with dynamic full-scale fatigue testing reserved for structures not historically proven by gruelling dynamic fatigue tests. (Ronig, 2015, pp. 1-13).

The problem with the current preferred certification method, the static method, according to Pommera (2000, p. 3) is that the method limits the maximum allowable sailplane service life to 12 000 Flight Hours (Fh). In warmer regions, such as Australia, the flight hour restriction can be problematic since sailplanes tend to accumulate flight hours at an unusually rapid rate. (Mileshkin, *et al.*, 1987, pp. 507-512; Payne, *et al.*, 1986; Van der Walt, 2020, p. 7). Sailplanes that accumulate flight hours too fast will have to be replaced sooner than planned by pilots at a high cost in countries like Australia compared to European countries.

Sailplanes in the field will have to be retired within a few years, due to rapid flight hour accumulation close to the specified maximum allowable flight hours. Sailplanes that acquired the maximum allowable flight hours, nonetheless, will tend to be capable of a safe flight for an extra few thousand flight hours due to the over-designed nature of sailplane structures and strict maintenance plans. (Patching & Esson, 1978, pp. 1-4).

The current international certification legislation allows sailplane manufacturers to extend the service life barrier of the static certification test past 12 000 Fh by employing intensive full-scale dynamic fatigue testing on the sailplane structure. (EASA, 2009).

The problem with the full-scale dynamic fatigue testing method is that a life factor of at least three (Patching & Wood, 1991) is required for certification which is costly and frequently not worth the trouble since it increases the sailplane base price far above the affordability barrier. (Langer, 2020).

Some sailplane manufacturers and research entities investigated other means to extend the flight hour barrier without compromising safety while lowering the dynamic fatigue certification cost. (Stafiej, 1994, pp. 99-103; Fox, *et al.*, 2004, pp. 1-11).

The most identified variable that results in lowering of dynamic fatigue certification cost is to shorten the time-intensive flight load spectrum. (Van der Walt, 2020). Kossira and Reinke (1997) replaced the Franzmeyer flight load spectrum with the KoSMOS I flight load spectrum to extend the service life of sailplanes. (The KoSMOS I & II flight load spectrum is described in Chapter 2 of this dissertation). In the 1980s the KoSMOS I spectrum was succeeded by the current KoSMOS II spectrum which comprises of only the crucial flight loading parameters found within the original spectrum. (Kossira, 1997(a); Kossira, 1997(b)).

The problem with the KoSMOS II spectrum is that although it reduces the testing duration and cost compared to the original, the cost is still not below the affordability barrier due to the strenuous test duration. A new spectrum type thus needs to be implemented to shorten the fatigue certification testing duration even further.

Research on wind turbine blade certification under the German Aerospace Center (DLR) - WISPER wind turbine research program was conducted to reduce the certification duration of large rotor wind turbine blades with an operational service life of about 20 years and an estimate 10^8 load cycles. (Kensche, 2006; Kensche & Seifert, 1990; Nijssen, *et al.*, 2002; ten Have, 1988). Since wind turbine blades are similar in structure to sailplane wings, the research field can overlap extensively. (Kensche, 1985, pp. 57-62).

Kensche (2002(a), pp. 32-43) investigated the Constant Amplitude (CA) test spectrum on two identical sectional wing specimens. The research goal was to accelerate the fatigue damage state within the composite structure. The research outcome showed promise for a certification procedure that if implemented successfully could extend sailplane service life at an accelerated rate compared to the certification procedure prescribed by the Joint Aviation Authorities (JAA) prior to 2003.

When substituting the KoSMOS II load spectrum, which is a Variable Amplitude (VA) load spectrum, with that of a CA load spectrum equivalent to Kensche (2002(a), pp. 32-43), a problem arises due to uncertainty of the damage rate correlation between the two different load spectrum types. The difficulty in determining the damage rate correlation between the

different loading spectrums is a possible result of different failure mode sets found within fatigue subjected Fibre Reinforced Polymer (FRP) composite structures. (Talreja, 2016, pp. 25-40).

The occurrence of different failure mode sets can be seen in the extensive data scatter results of fatigue tests where the fibre reinforced polymer structure fails at different lifecycles (Pfaller, *et al.*, 2012, pp. 1-10) due to damage accumulation build-up that reached a critical residual strength level ensuing in complex structural failure. (King, 1989, pp. 720-726).

In 2020, Van der Walt collaborated with an internationally known sailplane manufacturer to establish an empirical method based upon the research of Kensche (2002(a), pp. 32-43), Rodzewics (2009, pp. 452-460), Soinne (2015, pp. 5-68) and Trappe *et al* (2010, pp. 94-99). The research of Van der Walt focused on interchanging the KoSMOS II loading spectrum with an equivalent CA loading spectrum. In theory, the spectrum exchange causes an increase in the fatigue damage accumulation rate while being subjected to the Palmgren-Miner (PM) linear damage accumulation assumption used currently in aviation certification (Hult, 1992, pp. 63-70).

This proposed empirical Multiple Load Sequence (MLS) methodology's primary goal according to Van der Walt (2020), is to reduce the dynamic fatigue test time within acceptable levels.

The original empirical MLS methodology was only tested on an elemental type specimen as prescribed in the Building Block Approach (BBA) hierarchy illustrated in Figure 1, as used commonly in the general aviation industry for aircraft development (Matthews, *et al.*, 2018, pp. 907-929).

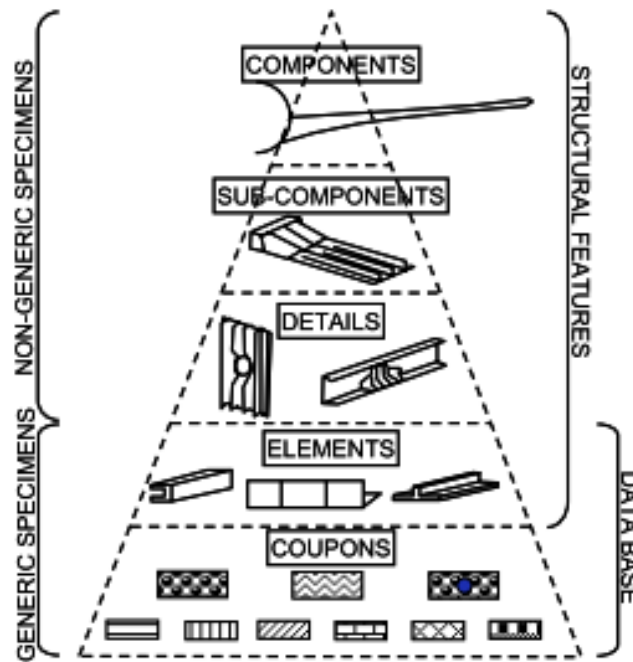


Figure 1: Building Block Approach Hierarchy (Matsumura, et al., 2010)

The original Van der Walt (2020) specimen illustrated in Figure 2, represented a shear web model of a sailplane, which was determined by Kensche (2002(b), pp. 51-55) to be one of the most critical subcomponents in a sailplane structure.

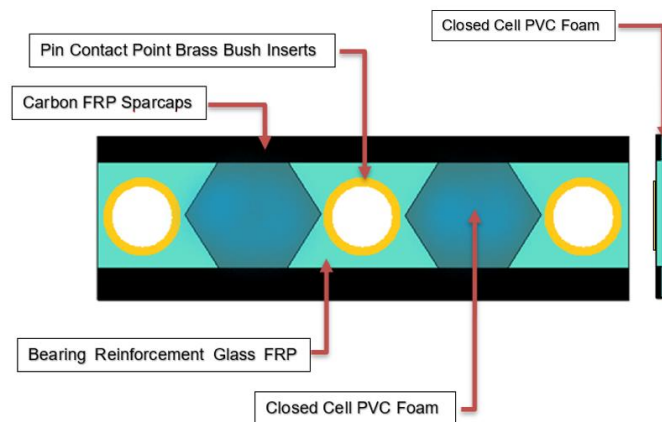


Figure 2: Original Van der Walt Shear Web Specimen (Van der Walt, 2020)

The elemental type specimen in Figure 2, however, resulted in an extensive data scatter during high loading conditions not associated with sailplane flight loads (Van der Walt, 2020, p. 89). The experimental results of Van der Walt (2020) caused confusion on the validity of the PM linear damage accumulation assumption used by aviation certification authorities for Fibre Reinforced Polymer (FRP) material aircraft.

The data trend noted by Van der Walt was classified as a quadratic function, resulting in considerable uncertainty on whether the MLS method is suitable to replace the accepted VA method.

The non-linear correlation between the CA and VA spectrums, as found by Van der Walt (2020), might possibly be the result of the fact that only two points was used to predict the actual failure cycle count. According to Van der Walt (2020, pp. 22-25, 88-92), the MLS method can be adapted to include multiple data points for a small specimen to form a detailed graph that can be used to determine the failure of a full-sized specimen for example, wing structures.

Van der Walt (2020) showed that the MLS method can be used to reduce full scale fatigue test times by months and the cost tenfold, if the relationship between VA and CA load spectrums can be accurately determined.

1.2. Problem Statement

The relationship between the CA and VA methods are not accurately known and must be determined experimentally.

1.3. Aim and Objectives

This study aims to evaluate the empirical MLS methodology proposed by Van der Walt in 2020 for FRP material component fatigue life prediction and sailplane service life extension past the 12000 Fh boundary, as defined by current European sailplane legislation requirements.

The following objectives must be reached to achieve the stated aim:

- Design or identify a suitable BBA coupon type test specimen that is easy to manufacture with sailplane manufacturing techniques.
- Gain a clear understanding on what the KoSMOS II spectrum is as used by Van der Walt (2020) to reproduce the study results with the new specimen type.
- Determine the chosen specimen's UTS with tensile testing.
- Conduct both CA and VA fatigue tests until the specimen fails at 50 % UTS, 65 % UTS as well as at 75 % UTS.
- Conduct MLS tests at 50 % UTS, 65 % UTS and 75 % UTS.
- Validate the MLS results with the Residual Strength and Residual Stiffness methods.
- Determine the relationship between CA and VA load spectrums through use of the MLS methodology for future service life extension of sailplanes.

1.4. Layout of the study

The dissertation scope is divided into 5 chapters.

Chapter 1: Introduction

In Chapter 1, a brief introductory was given in regards to sailplane service life extension past the certification legislation barrier of 12000 Fh and the proposed MLS methodology to shorten the testing duration. The dissertation aim, objectives and scope are also defined.

Chapter 2: Literature Study

In Chapter 2, the development of sailplane certification legislation and the current European Union sailplane certification methodology was briefly researched. Research was conducted on the current KoSMOS II load spectrum, the PM linear damage accumulation method and the newly proposed 2020 empirical MLS methodology for sailplane service life extension.

Chapter 3: Methodology

In Chapter 3, an alternate hypothesis was formulated based on the hypothesis of the 2020 MLS methodology, a coupon type FRP specimen was designed for fatigue testing, followed by the manufacturing procedure of the specimen and finally the test methodology including validation used in the dissertation was discussed in detail.

Chapter 4: Results

In Chapter 4 the results obtained for CA, VA and MLS load spectrum testing at different load levels were discussed in regard to data scatter, correlation and validity.

Chapter 5: Conclusion and Recommendations

In Chapter 5, a conclusion regarding the MLS method's correlation between CA and VA load spectrums for FRP materials were made at flight applicable loads followed by a few recommendations in regards to the future of the MLS methodology.

2. LITERATURE STUDY

Chapter 2 of this dissertation focuses on the historical development of sailplane legislation requirements, the current certification methods used for sailplane certification and the Multiple Load Sequence methodology proposed by Van der Walt (2020).

2.1. Historical Development of Sailplane Legislation

Sailplane legislation can be categorised into five eras where the first legislation era was the *Rhön-Rossitten-Gesellschaften* (RRG) design rules. The RRG society was established in the early 1900s, by sailplane designers and pilots. Sailplanes became popular in Germany during the 1920s, due to the Treaty of Versailles that forbade powered flight. (Reitsch, 2009).

The first official gliding competitions date back to 1920 and was located on the Wasserkuppe mountain in the Rhön region of Hesse and at the Rossitten dunes found in East Prussia's Curonian Spit. Both of these competitions were created independently by different aviation hobbyist groups that longed for powerless flight under the Treaty of Versailles. (Griffitts, 2004; Reitsch, 2009).

Although the Rhön region is far apart from the Rossitten dunes, the pilots and designers frequently communicated, sharing design and manufacturing methods. This sharing of knowledge allowed for rapid development of soaring as a sport and aviation in general in the years to come.

The original glider design, flown at the first Wasserkuppe competition that showed promise was the "Black Devil" that managed to soar for more than a minute before landing without complications. The Rhön Wasserkuppe gliding competition continued for years. With each annual competition, the designs and results improved mainly due to extensive research on sailplane aerodynamics and structural design by universities and hobbyists, under the influence of the likes of both Oskar Ursinus and Theodore von Kármán. (Reitsch, 2009).

In 1924, a united organisation was suggested by Oskar Ursinus better known as "*Rhönvater*" to coordinate soaring activities at the Wasserkuppe Rhön region, the Rossitten dunes and any other gliding club that was to be founded. The organisation proposal was made to the Ministry of Transportation's Dr Brandenburg, Secretary of Air transport during 1924. (Reitsch, 2009).

The newly founded Germanic national gliding organisation was henceforth known as *Rhön-Rossitten-Gesellschaften* (RRG). The RRG's primary role was to promote gliding by organising annual gliding competitions, giving flight training at both Wasserkuppe and Rossitten. The organisation also opened research facilities and workshops to develop better

sailplane designs and to investigate the sources of lift, known at the time as ridge-lift and thermals.

In 1925, “*Rhönvater*” appointed Alexander Lippisch as the head of the RRG's technical branch. Lippisch's role in the RRG allowed for the advancement of sailplane design methodology through more sophisticated sailplane designs. During early 1925, RRG started to develop a better launching method compared to bungee-launching method later known as the winch launch method. With the ban on powered aircraft lifted partially in late 1925, the RRG started researching motorised sailplanes and aerotowing. (Reitsch, 2009).

During 1933, the independent RRG soaring society was disbanded by the Nazi coming into power in Germany. The flying club was absorbed by the Hitler youth where the design and research section were preserved under the *Deutsche Forschungsanstalt für Segelflug* which was known at the time as the German Institute for Glider Research. (Reitsch, 2009).

The second era in sailplane design was promulgated strictly under the *Deutsche Flugsicherung GmbH* (DFS) airworthiness requirements for most of the European countries. The first DFS director, Professor Georgii, allowed Lippisch to retain the position of chief designer, which ultimately resulted in the evolution of German glider design. The primary driving force of aeronautical design in the world from 1933 to 1945 was the DFS and RRG. (Reitsch, 2009).

During 1933 to 1936 the DFS airworthiness requirements mainly focused on sailplanes used either for training or transport purposes of the *Luftwaffe* and later started focusing on highly experimental aircraft that resembled flying wings with rocket motors as a propulsion source. An example of such a rocket motor propelled aircraft is the *Messerschmitt Me 163*, also known as the Comet displayed at the National Air and Space Museum's Udvar-Hazy Center and illustrated in Figure 3. (Smithsonian National Air and Space Museum, 2020; Reitsch, 2009).



Figure 3: Messerschmitt ME 163B-1 "Comet" (Smithsonian National Air and Space Museum, 2020)

In early 1938, Wasserkuppe saw a fatal sailplane accident during a competition which led to a requirement for higher efficiency dive brakes (U.S. Department of Transportation FAA Flight Standards Service, 2013). After the competition the DFS airworthy requirements adopted the new airbrake design to promote safety in future sailplane designs. (Evelyn, 2009; Reitsch, 2009).

Today the modern German Aerospace Center (DLR) still focuses their research into sailplane flight performance with sailplanes like the Discus-2C DLR and the retired DLR's Glaser-Dirks DG-300/17 Elan. The DG-300 with a 17-meter wingspan was mainly used to measure and establish corresponding sailplane performance parameters while the Discus-2C DLR is used for digitalisation research and development of a digital twin to gauge new glider type performance. (DLR -German Aerospace Center, 2020).

The third era in sailplane design dates to 1950 and was mainly promulgated by the Sailplane Design Panel (SPD) of 1962. The *Organisation Scientifique et Technique Internationale du Vol à Voile Airworthy Standards* (OSTIVAS) were applicable to both powered and conventional sailplanes of the period. The OSTIVAS are a few decades later still valid with Miner adjustment to the regulations and form the base guidelines for the current sailplane certification specifications of the European Aviation Safety Association (EASA) that is known as CS-22. (OSTIV, 2017; Waibel, 2007). The OSTIVAS is also known to contain information that can be used in to improve the regulations that are not properly captured by the CS-22. (EASA, 2007, p. 8).

The fourth era in sailplane design dated back to 1979 and was promulgated by the Joint Aviation Research panel. The Joint Aviation Requirements (JAR) was a global collaboration that strived to set a world standard for aircraft design in general. The JAR-22 was at the time known as the sailplane airworthiness standard and was mainly based upon the OSTIVAS. The JAR-22 mainly covered general requirements, flight, structure, design as well as construction, equipment, operational limitations and information. (Thomas & Milgram, 1999, p. 62).

The fifth and current era of sailplane design dates back to 2003 and is promulgated by the European Aviation Safety Agency (EASA). The EASA certification specification known as the "Certification Specifications for Sailplanes and Powered Sailplanes CS-22" is firmly based on the Joint Aviation Requirement's JAR-22 with minor alterations to the specifications. (SofemaOnline, 2017; SKYbrary, 2019).

In the current CS-22 documentation, fatigue requirement is severely understated for sailplane structures. The only data that could be found within the documentation regarding fatigue is contained in CS 22.627 and AMC 22.613.

CS 22.627 states that a sailplane structure should be designed to avoid stress concentration points containing variable stress loading exceeding the fatigue limit during standard operational service. (EASA, 2009, pp. 1-D-1).

The CS-22 Acceptable Means of Compliance's Material strength properties and design values AMC 22.613 states that only EASA approved material specification data or documentation should be used. (EASA, 2009).

The material specification data should be carefully adjusted by the manufacturer to suit the manufacturing methods used, for example, the construction method, component geometry, machining method, as well as the heat treatment cycle required for adequate post-curing of the composite parts to match the corresponding "normal" operating conditions. Average operation temperature for composite sailplanes with white paint schemes is stated at 54 °C according to AMC 22.613(c). (EASA, 2009).

2.2. Fibre Reinforced Polymers (FRP) in the sailplane industry

According to Campbell (2010, p. 1), a composite material is defined as a material that is blended from two or more different material types. The main advantages of composite materials compared to traditional aerospace alloys according to Campbell (2010, p. 14) are:

- Greater corrosion resistance;
- Improved fatigue resistance;
- Lower density;
- Increased strength and stiffness.

According to Campbell (2010, p. 14) the main disadvantages of composite materials compared to traditional aerospace alloys are:

- High fabrication costs;
- Lower impact resistance;
- Adverse effects of environmental variables like temperature and moisture;
- Greater manufacturing difficulty.

Aviation composite materials mainly consist out of epoxy and fibres. The fibres can be made from glass or carbon. Epoxy which is the matrix part of a composite is classified as a two-part mixture consisting of a catalytic hardener and resin. The thoroughly mixed hardener and resin is the binding agent of composite materials associated with aviation structures. When the resin is mixed with a catalytic hardener, it results in an exothermic reaction which allows for curing of the bonding agent. (Everitt, *et al.*, 2015).

During the curing phase, the resin and hardener react to form a covalent bond. The covalent bond between the epoxy resin groups and amine hardener groups are responsible for the cross-linkage found within the cured polymer. (Everitt, *et al.*, 2015).

In modern aviation reinforcement fibres are mostly made of glass, carbon or Kevlar, and in some exceptional cases metallic materials such as aluminium, steel and titanium. (Nijssen, 2007).

According to Hamilton and Moring (2014, pp. 189-227) material such as glass, carbon and Kevlar fibres have been used extensively for decades. The first prototype of a composite sailplane was built in 1956. The *Akaflieg Stuttgart's* FS-24 "Phoenix" had its maiden flight in Germany on 27 November 1957. (Akaflieg Stuttgart, 2014). FRP material sailplane design was met with considerable scepticism due to little research on FRP materials suitable for aviation use at the time.

The plastic sailplane design later proved to be an improvement when compared to rag and tube sailplanes of the same era due to lower drag coefficients, greater structural integrity and fatigue resistance. The FS- 24 "Phoenix" can be considered as the father of composite sailplanes since most of the design materials used were from glass fibre reinforced polymer (GFRP) materials and modern sailplanes primarily consist of FRP materials be it carbon, glass or aramid. (Thomas & Milgram, 1999, p. 178). The FS-24 "Phoenix" is illustrated on the postage stamp in Figure 4.



Figure 4: FS-24 "Phoenix" 1957 (Eurocards, 2018).

Modern sailplanes are built mainly from composite materials due to the success of the FS-24 "Phoenix" and other sailplanes that ensued. Examples of well-known composite sailplanes are the 1960's D-34d, Libelle H-301 "Dragonfly" by the legendary Eugen Hänle and the 1970's Kestrel T59.

When designing a modern composite sailplane, a few design factors such as stress concentrations and structural stability need to be considered by sailplane design engineers since sailplanes tend to require long slender wings subjected to high wing loading specially designed for both gliding and thermal purposes. (FAA, 1973).

An important factor when it comes to composite materials is the fibre ratio. The fibre ratio is defined as the amount of fibre present in a composite material. The fibre ratio plays a crucial role in the strength of a composite material since it is closely linked to the mechanical properties. With an increase in the fibre ratio, the UTS improves greatly. (Hull & Clyne, 1981).

According to Hull and Clyne (1981) the formula for the fibre ratio is given by Equation 1:

$$V_f = \frac{v_f}{v_c} \quad (1)$$

Where:

- V_f is the fibre volume ratio
- v_f is the volume of the fibres
- v_c is the volume of the composite

According to Boller (1964, pp. 145-150) the matrix material determines the performance of low fibre volumes resulting in the structural properties to vary between the matrix and the fibres. The tensile strength performance for composite materials is slightly lower than the compressive strength with the tensile strength performance being proportionally related to the number of fibres present.

In 1964, Boller conducted various tests with fluctuating stress levels to test GFRP specimen constructed from different material layup orientations. The result of the study was that the best fatigue strength of a material consisting of 33 [%] resin occurred at an estimated ten million cycles until failure. (Boller, 1964, pp. 145-150).

2.3. General Composite Material Fatigue Literature

Fatigue is a progressive, localized, plastic deformation that occurs within a material subjected to fluctuating stresses which in turn could result in microscopic cracks, or fractures at high stress concentration regions located within a structure if the stress fluctuations are repeated enough times. (Boyer, 1986; Lee, *et al.*, 2011).

According to Weibull (1961, pp. 7-23) the main objective of fatigue testing is described as a process where the fatigue life of a specimen subjected to fatigue loading is determined.

2.3.1. Fatigue failure

The failure state found within the structure subjected to fatigue testing can be divided into initiation, propagation, and final fracture. The initiation, propagation, and final fracture stage of fatigue can be seen for FRP materials in Figure 5.

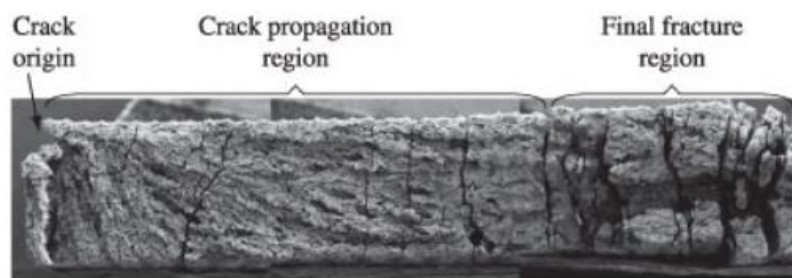


Figure 5: FRP Fractographic image (Srinivasal, *et al.*, 2010, pp. 417-424).

The initiation stage is assumed to be the most complex stage of fatigue failure. The initiation stage is assumed to be the most complex stage of fatigue failure. The initiation stage of failure usually occurs in areas like sharp corners, fillets, holes and grooves found within a specimen subjected to high-stress concentrations (TEC Eurolab, 2005).

The propagation stage is where the microcrack formed during the initiation phase starts to develop in direction and magnitude. In metals the crack growth is perpendicular to the tensile strength. For composite materials, this may not necessarily be the case since crack growth tends to be in the direction of least resistance to breakages. (Boller, 1964; Talreja, 2016).

The crack propagation in composite materials like GFRPs can be seen with the naked eye if a high-intensity light is shined behind the specimen. (TEC Eurolab, 2005).

Final fracture results from a specimen subjected to fatigue testing in the propagation phase with crack growth that occurs until the ultimate strength equal the maximum applied stress. The final fracture occurs mainly due to the brittle nature of the specific material tested. (Azeez, 2013).

In Figure 6 the following three stages: initiation, propagation, and final fracture are indicated. On the vertical axis is the stiffness of the structure and on the horizontal axis is the cycle count.

- Phase 1 is defined as the initiation phase, where the material stiffness degrades rapidly before reaching phase 2.
- Phase 2 is defined as the propagation phase, where the material stiffness gradually decreases linearly before reaching phase 3.
- Phase 3 is defined as the final fracture phase, where the material stiffness degrades exponentially until fatal failure occurs of the structure.

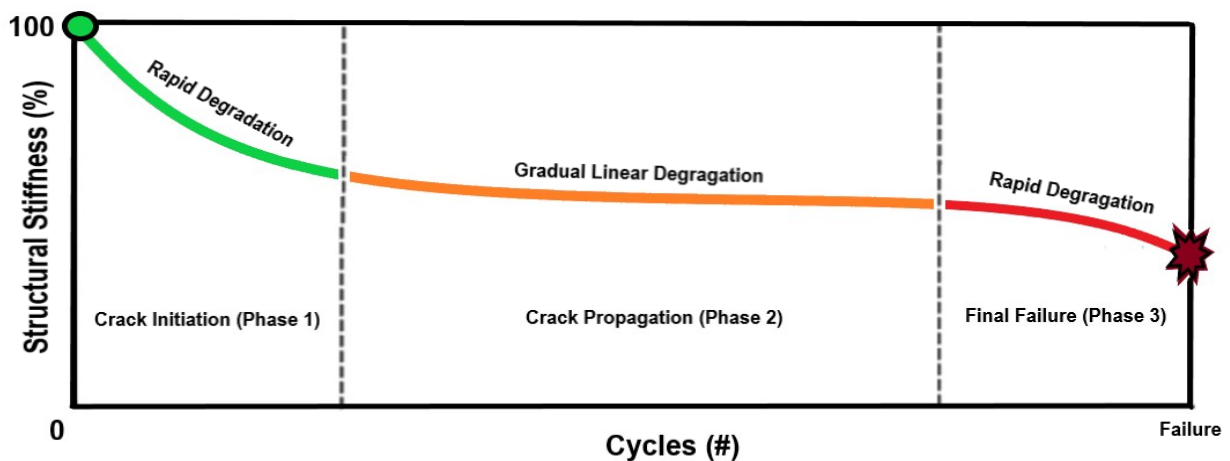


Figure 6: Composite Material Stiffness Degradation

Specimen with surface defects like notches, scratches and shoulders tend to reduce the fatigue life of the defect specimen. The fatigue life of the specimen can, however, be extended by applying defects on internal load paths. (Azeez, 2013).

Factors that influence fatigue are the load type, load intensity, specimen geometry, material type and environmental effects such as air moisture. The type of loading can be tension, compression, shear and or fatigue loading like Constant - or Variable Amplitude loading.

Several steps can be taken during the design and manufacturing phase to prevent fatigue, these steps include but are not limited to avoiding of stress concentrations, choosing the correct material for a specific application and designing for manufacturing ease.

Composite materials are quite complicated when it comes to failure mechanisms due to multiple stage failure that may occur. (Azeez, 2013). Different type of failures illustrated in

Figure 7 that occur in fibre reinforced polymer composites are delamination, fibre kinking, fibre pull out, fibre rupture, matrix rupture, bridging and matrix/fibre debonding.

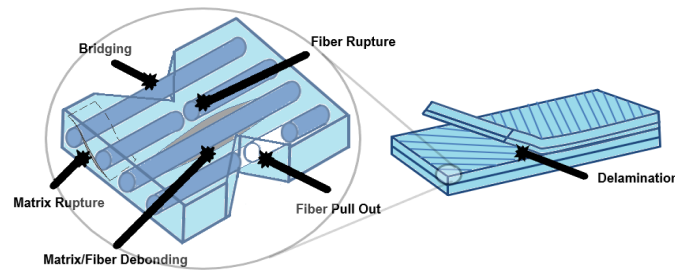


Figure 7: Composite Material Failure Illustration

Any type of failure can occur during composite material fatigue testing, making it difficult to accurately determine the fatigue life of a specimen. The specimen's manufacturing thus needs to be easy and accurate to deliver equivalent specimens for accurate fatigue life data.

2.3.2 Fatigue testing

Fatigue testing will be performed for two different load envelope types for this dissertation. The two load envelopes that are used in this dissertation are the CA and VA load spectrum. The CA load spectrum results in fast-paced fatigue tests with each load cycle being equivalent to the previous. The VA load spectrum consist of different load cycles with most of the load cycles having smaller loads compared to the CA load spectrum.

Fatigue tests are generally performed to predict the life of a structure subjected to a loaded envelope and are classified into the following categories, cumulative/ progressive damage, empirical/ phenomenological, micro-mechanics.

2.3.2.1 Cumulative Fatigue Damage (Palmgren-Miner damage accumulation) Models

In this subsection, accumulative damage literature as per aviation standard is researched. This literature research allows the researcher of this dissertation to conclude whether the Palmgren-Miner (PM) linear damage accumulation method remains valid for the use within the MLS methodology.

The PM linear damage accumulation method seems to be the standard when it comes to certification of sailplanes and aircraft in general. The PM method is not without limitations and at best is only an approximation that should be used with a pinch of salt for FRP materials. The limitations of the method include but are not limited to the data in Table 1.

Table 1: Palmgren-Miner Damage Accumulation Limitations

Limitation	Source
The model does not consider the microdamage mechanisms within a specific material, reducing the accuracy drastically. (The model can be used for any material).	(Harris, 1999, p. 136)
Uncertainty in the life estimation exists due to actual damage accumulation not necessary being linear of nature, hence resulting in the inaccurate interpolation of the life estimation.	(Hayman, <i>et al.</i> , 2008, p. 348)
The PM method uses a strength equivalence assumption that is non-conservative. The non-conservative assumption is that a composite coupon's strength is representative of full-scale composite structure irrespective of manufacturing defects or structural imperfections.	(Hayman, <i>et al.</i> , 2008)
The damage parameter does not physically quantify the damage accumulation and only indicates whether failure occurs.	(Nijssen, 2007, p. i)
The effects caused by sequential loading is not considered.	(Nijssen, 2007, p. i)
In Block or Spectrum applications the estimated result yields to be either conservative or extremely optimistic.	(Passipoularidis & Broendsted, 2009, p. 18)
The number of cycles to failure is gained from empirical data and hence requires extensive testing of a specific material before the Palmgren-Miner method may be used to estimate the applicable life.	(Van der Walt, 2020)

The sequence of failure events according to Table 1, drastically changes how the damage within a composite structure is compiled, resulting in irregular damage accumulation rates that are variable of nature. The Palmgren-Miner linear damage accumulation rule written in mathematical notation can be seen in Equation 2.

$$D = \sum_{i=1}^k \frac{n_i}{N_i} \quad (2)$$

Where:

i denote the cycle type.

n_i denotes the number of cycles of type i in the loading sequence.

N_i represents the expected number of cycles to failure for type i .

D represents the damage number and failure is expected if D equals unity ($D=1$)

The total damage in a structure at a given point is calculated by the sum amount of loads present for a specific load type divided by the actual expected number of cycles until ultimate failure is reached. A graphical illustration of Equation 2 can be seen in Figure 8 (Krause, *et al.*, 2003).

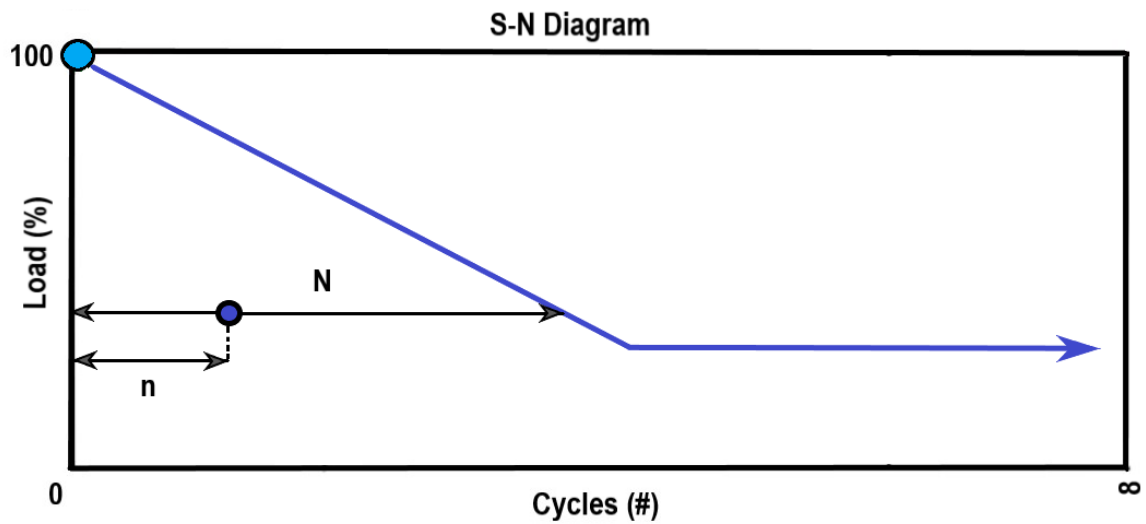


Figure 8: The PM Method applied on a Stress-Life diagram adapted from Van der Walt (2020, p. 39)

2.3.2.2 Empirical Models

In empirical fatigue models, a physical measurable like strength or stiffness is tracked to describe the strength and stiffness degradation of a specimen subjected to fatigue loading to predict the fatigue life.

According to Nijssen (2007), the drawback of empirical models is that extensive test data is required for accuracy. Nijssen (2007) went further, stating that no guarantee can be given that the models' parameters of a particular composite specimen type will be accurate for another specimen type when life prediction is required.

2.3.2.3 Micro- Mechanical Models

Micro- Mechanical models describe the fatigue degradation within composites from a primary ply material-properties angle. According to Nijssen (2007), the difference between Micro-Mechanic models and empirical fatigue models are minimalistic since Micro- Mechanical models are known to consist of considerable modelling detail to establish the constituents. Hence, the Micro-Mechanical models used for fatigue life prediction are mainly exploratory for single failure nodes.

2.4. Fatigue Testing Envelope Type

With composite material fatigue criteria briefly researched the literary focus shifts towards the general fatigue testing envelope types applicable to this dissertations study. The applicable envelopes are the constant and variable fatigue testing envelopes.

2.4.1 Constant Amplitude Fatigue testing envelope

CA fatigue testing is the most straightforward envelope since a single load fluctuation is repeated multiple times over a given period until failure occurs. A graph of a CA fatigue envelope can be seen in Figure 9.

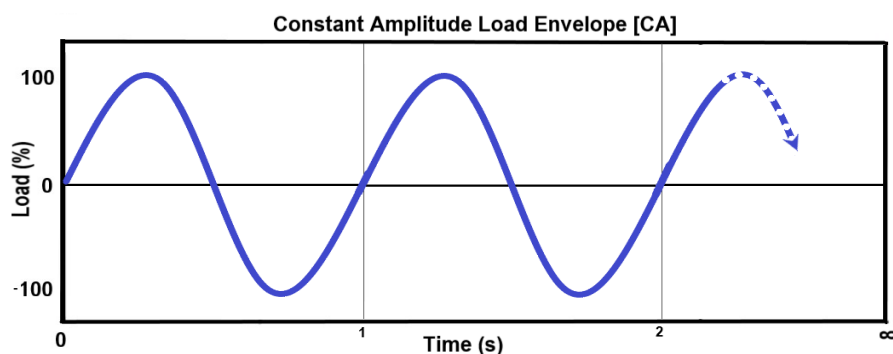


Figure 9: Constant Amplitude Load Envelope Example adapted from Van der Walt (2020, p. 47)

In Figure 9 it can be noted that the spectrum is like a sine waveform with the load on the vertical axis and the time on the horizontal axis. The applied load cycle from time instance zero to one is equal to the load cycle from time instance one to two for a CA load envelope.

According to Weibull (1961, pp. 7-23) CA fatigue testing is divided into several categories that are mentioned in Table 2.

Table 2: Constant Amplitude Fatigue Testing Categories

Test Category	Test Category Identifier
Routine Fatigue Tests	Specimen is expected to fail after a moderate cycle count of between 10^4 to 10^6 cycles due to the applied fluctuating stresses.
Long Life Fatigue Tests	Specimen is expected to fail between a cycle count of 10^6 to 10^7 due to the applied stresses located near the fatigue limit.
Short Life Fatigue Tests	Specimen is expected to fail within a few thousand cycles due to the applied stress levels being close to the yield strength of the given specimen.

2.4.2 Variable Amplitude Fatigue testing envelope

VA fatigue testing compared to CA testing is a more complex envelope since multiple load fluctuation at different load peak values come forth over a given period until failure occurs. The KoSMOS load envelopes as used in aviation certification are classified as VA envelopes due to the 32 random load parameters that occur during testing.

According to Weibull (1961, pp. 7-23) when VA fatigue testing remains complex, it can be classified as service simulating tests. This argument is supported by Portalés *et al.* (2018, pp. 1662-1673) who states that a pure load spectrum consisting of a variable load envelope should be considered for service life estimation since it is more accurate and reliable compared to CA load envelopes to predict the fatigue life of a component.

A representation of a VA fatigue envelope can be seen in Figure 10.

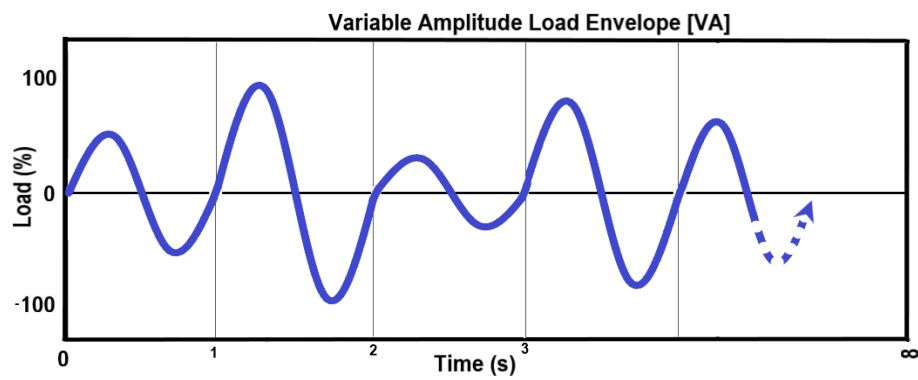


Figure 10: Variable Amplitude Load Envelope Example adapted from Van der Walt (2020, p. 47)

In Figure 10 it can be noted that the spectrum is an alternating waveform with the load on the vertical axis and the time on the horizontal axis.

2.5. Current Sailplane Certification

Sailplane certification in 2021 as prescribed by EASA conform to the CS-22 design requirements. According to EASA (2009), sailplane fatigue compliance certification can be divided into two main categories, as seen in Figure 11.

The first category for sailplane fatigue compliance certification is based on proven certified designs that can be certified through static testing of the full-scale airframe and critical components. In contrast, the second category for sailplane certification is based on dynamic fatigue load spectrum testing of the full-scale airframe.

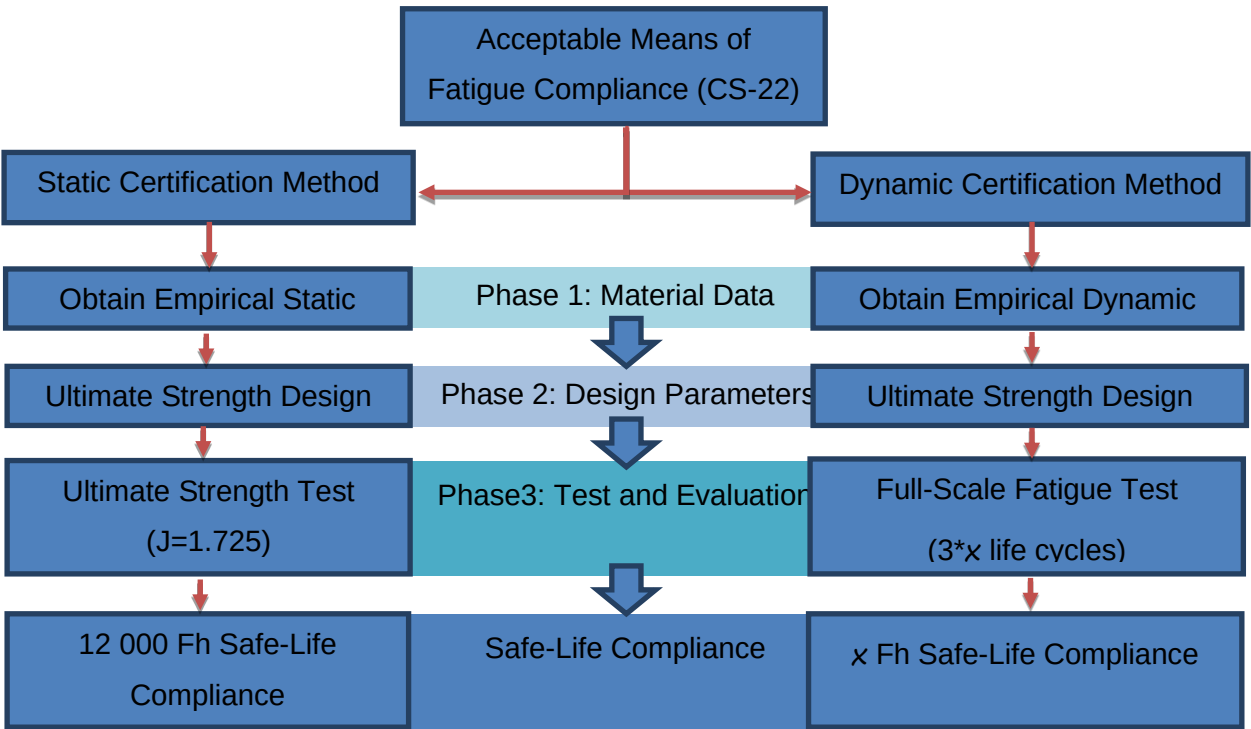


Figure 11: Fatigue Compliance Certification of Sailplanes

The static certification method is a cheaper option compared to the dynamic method since it only requires a fraction of the tests to certify a sailplane. The problem with the static certification method is that it limits the amount of achievable certified flight hours to 12000 Fh before a sailplane needs to be retired from service.

With full-scale dynamic fatigue certification, the problem is that it can be costly to complete due to the amount of work as well as the long test duration required. Although costly, the advantages of full-scale fatigue certification outweigh that of static certification testing in the long run when looking at the achievable certified safe life when using the dynamic certification method

In this sub-chapter, the two categories will be briefly summarized with more focus being applied to service life extension. Theoretical required includes, but aren't limited to, the KoSMOS II load spectrum, cumulative damage measurement, static, and dynamic fatigue testing.

After the current sailplane certification methods are both briefly discussed, the focus will be shifted towards the 2020 MLS method where the original method is summarized thoroughly to formulate a new hypothesis.

2.5.1 The KoSMOS II Spectrum

Current sailplane certification method can be better understood from gaining insight into the current flight spectrum used for certification purposes. For sailplanes, the flight spectrum is known as the KoSMOS II load spectrum created by Kossira and Reinke (1997(b)).

The original KoSMOS load spectrum was first conceptualized in the 1980s and is based on actual sailplane flight cycle data. (Kossira & Reinke, 1983).

A typical flight cycle contains the following three phases: take-off, routine glider operational flight, and landing. The flight spectrum used before the original KoSMOS spectrum was known as the Franzmeyer-programme. (Thielemann & Franzmeyer, 1969). The problem with the Franzmeyer-programme was that it only simulated 3000 Fh for lifetime certification purposes of fibre reinforced sailplanes built from glass, carbon, or aramid materials. (Kensche, 1983, pp. 95-108).

The Franzmeyer-programme maintains validity since minor modifications to the Franzmeyer-programme through flight tests resulted in the original KoSMOS load spectrum's creation, which represented 6000 flight hours. (Kensche, 2018).

The KoSMOS load spectrum was initially established by conducting flight tests with a Janus sailplane. The wings of the Janus were fitted with strain measurement equipment which allowed analogue data to be converted into digital data. The digital data could then be categorized into 32 different load parameters.

These 32 different load parameters were then stored in a 32 x 32 Markov transition matrix (Moffat, 2018). The KoSMOS spectrum was established to cover all the various glider types, forming an envelope type spectrum. (Kossira, 1997(b); Kossira, 1997(a)).

The envelope spectrum contains eight flight pattern operations that form the 32 different load parameters. The flight patterns according to Soinne (2017) are aerotowing up to 700 m, winch

launching, landing, circuit training, cross country flight, mountain flying, wave flying and aerobatic manoeuvres.

The problem arises that certification with the KoSMOS II spectrum is costly, causing sailplane manufacturers to certify sailplanes in general for the stipulated 12000 Fh as contained within the current sailplane certification legislation. (Kensche, 2018).

Three categories are required for lifetime assessment of sailplane structures to extend the service life. The three categories are material-specific data, the KoSMOS II spectrum, and a reliable damage accumulation hypothesis. According to Kensche (1985, pp. 57-62), the damage accumulation hypotheses used in the sailplane and wind turbine and industry is known as the PM linear damage accumulation rule.

According to Kensche (1985, pp. 57-62), the PM linear damage accumulation rule is suitable to estimate the damage of the rotor blades found within wind turbine assemblies. The wind turbine rotor blades can, in theory, be compared to sailplane wings since they are quite similar in structure.

The German Aerospace Center (DLR) demonstrated that a significantly higher service life for composite wing-like structures could be achieved when using the KoSMOS II spectrum than the PM linear damage accumulation rule initially predicts. (Waibel & Schleicher, 2002, pp. 56-61). The higher service life is mainly due to the material-specific properties of the FRP composite materials used within the wing-like structures.

The KoSMOS II flight loading spectrum was put to the test a few years ago, when researchers (Kensche & Wassenaar, 2017) investigated whether simple acrobatic flight manoeuvring and basic flight training resulted in higher damage rates than the KoSMOS II flight loading spectrum. The strain-based flight tests were performed on 18 m acrobatic configured DG-1000 sailplanes that were fitted with fuselage accelerometers and various strain gauges strategically placed throughout the sailplane's wing structure.

The data obtained from the four different experimental configurations were converted individually into the traditional 32 x 32 Markov matrix so that the loading pattern and resulting damage accumulation could be compared to that of the KoSMOS II spectrum.

The result of the spectrum comparison was similar. The researchers (Kensche & Wassenaar, 2017) could not formulate a conclusion regarding the spectrum differences with load characterization charts. Kensche and Wassenaar (2017) eventually demonstrated with a calculation that the four different experimental configurations did not exceed the maximum allowable values of the KoSMOS II loading spectrum resulting in the continued applicability of the load spectrum for certification use.

Wassenaar and Kensche (2017) noted at the OSTIV Congress (2017) about flight load measurement, that it is evident from their (2017) research that accelerometers placed in the cockpit region of the fuselage are not an adequate means to replace strain measurements taken directly from the wing spar and shear web. (Kensche & Wassenaar, 2017).

With literature regarding the KoSMOS II flight loads spectrum covering what the spectrum is used for, how the spectrum data was accumulated and by whom it was created, one question remains; How is the KoSMOS II load spectrum compiled?

The question can be answered by investigation of the known variables, for example; the number of classes present, and the amplitude ratio used.

The KoSMOS II load spectrum is divided into 32 different load classes that are transitioned to the 32 x 32 Markov Matrix for testing purposes (Kossira, 1997(b)). A spectrum with an amplitude R-ratio of $R = -0,55$ and load factor (J) of $J = 0,8$ the amplitude crest will be positive 0,8 times the Ultimate Tensile Strength (UTS) and the amplitude trough will be minus (-) 0,44 times that of the UTS.

An example of the KoSMOS II spectrum of compiled classes can be seen in Figure 12.

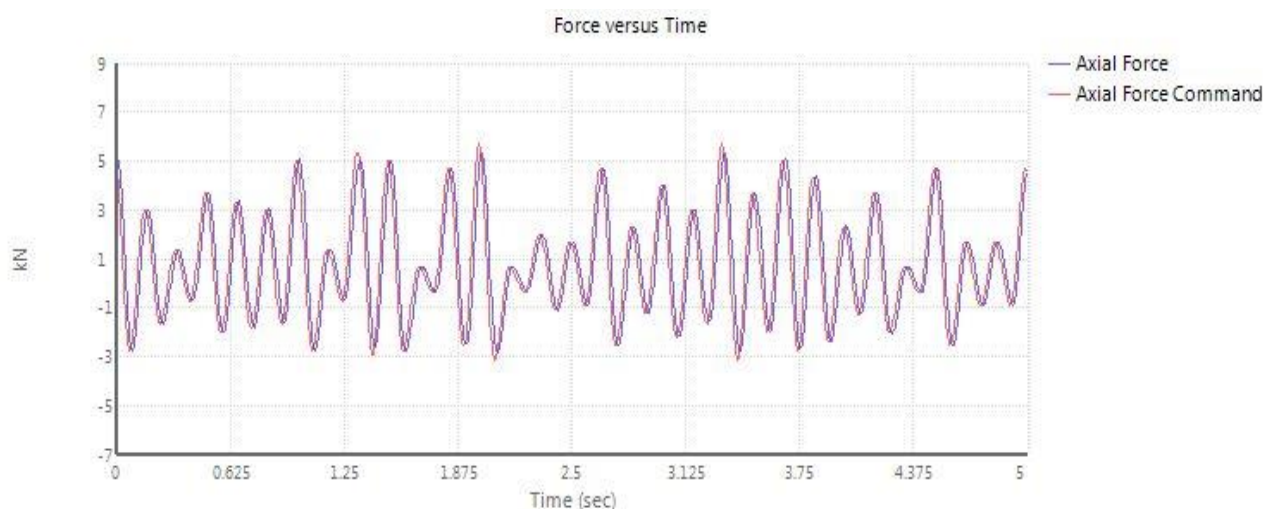


Figure 12: Example KoSMOS II Spectrum

In Figure 12 a sine waveform generated from 32 KoSMOS II ASCII characters: [F L P J K F P E F R G R O P G O I L E J F H O J T G P P G] subjected to a frequency of 6 Hz can be seen. Note that the sinewave form is random of nature and that some of the load classes are repeated in the displayed 29-character snippet out of the full spectrum comprising of 555651 randomly placed ASCII characters. (Kossira, 1997(b)).

With the basics covered, one requires knowledge on what the load parameter for each of the classes are, as well as how many times a particular class comes forth in the KoSMOS II

spectrum. In Table 3; the ASCI character load assignment based upon the original KoSMOS II spectrum (Kossira, 1997(b)) are demonstrated in regards to the amount that a specific class comes forth during the testing cycle:

Table 3: KoSMOS II ASCI Character Load Assignment

CLASS	ASCI	J - LOAD	CLASS	ASCI	J - LOAD
1	A	1,111	17	Q	0,222
2	B	1,056	18	R	0,167
3	C	1,000	19	S	0,111
4	D	0,944	20	T	0,056
5	E	0,889	21	U	0,000
6	F	0,833	22	V	-0,055
7	G	0,778	23	W	-0,110
8	H	0,722	24	X	-0,165
9	I	0,667	25	Y	-0,220
10	J	0,611	26	Z	-0,275
11	K	0,556	27	#	-0,330
12	L	0,500	28	\$	-0,385
13	M	0,444	29	-	-0,440
14	N	0,389	30	*	-0,495
15	O	0,333	31	!	-0,550
16	P	0,278	32	+	-0,605

With the basic knowledge of fatigue and how the damage accumulation rule is applied, research on the current certification techniques can commence.

Van der Walt (2020, pp. 22-33) categorized the specifications according to phase 1 and phase 3 of the Fatigue Compliance Procedure (FCP) illustrated in Figure 12. Both FCP phases 1 and 3 include the testing of materials and structures.

The specifications for phase 1 of the FCP were identified in Table 4.

Table 4: Fatigue Compliance Procedure Specifications

Document	Specification/ requirement	FCP Phase
(EASA, 2009, p. CS22.603 a.)	EASA CS22.603 a.) States that confidence in the selected material data is required for safety-critical components. EASA approved procedures may be used to establish data confidence in the selected material.	1
(Luftfahrt-Bundesamt, 1991, p. 1)	(Luftfahrt-Bundesamt) LBA the German equivalent to EASA requires	1

	that the materials used in the manufacturing of sailplanes require testing. The testing of the materials should be approved by LBA.	
(EASA, 2009)	Full-scale structural testing is required for life cycle substantiation. Either Static Ultimate testing or Fatigue Spectrum testing may be used for the life cycle substantiation.	3

Phase 3 of the FCP for dynamic testing consist of 3 major requirements. The three identified requirements can be seen in Table 5.

Table 5: FCP Phase 3 Dynamic Testing Requirements

Document	Fatigue Spectrum testing requirements
(EASA, 2009)	A life factor of 3 is required to certify for specifically required service life. If a service life of 12000 Fh is required, testing should be conducted for at least 36000 Fh.
(Luftfahrt-Bundesamt, 1991)	Static testing to the Limit Load (J = 1.0) at 54 °C for three seconds after completion of the Fatigue spectrum test to investigate structural deformation.
(EASA, 2009)	Static testing to the Ultimate Load (J = 1.5) at 54 °C for three seconds.

Phase 3 of the FCP for static testing consist of 3 major requirements. The identified requirements can be seen in Table 6.

Table 6: FCP Phase 3 Static Testing Requirements

Document	Fatigue Spectrum testing requirements
(Pommerer, 2000)	Stress levels in the wing spar and other critical components should not exceed the allowable stresses stipulated in (Luftfahrt-Bundesamt, 1991) as 400 N/mm ² for CFRP's and 250 N/mm ² for GFRP's.
(EASA, 2009)	The wing structure should be designed in such a way that it withstands a load of J = 1.725
(EASA, 2009)	The wing design of a sailplane should be similar to historically substantiated wings.

The testing philosophy can be derived for phase 3 of the FCP from Table 5 and Table 6. A theoretical stress life plot can be established to illustrate the testing philosophy for both static and dynamic methods.

Van der Walt (2020, p. 31) illustrated the two methods graphically side by side and noted that the two methods are closely related. An illustration of the testing philosophy for both static and dynamic methods can be seen in Figure 13.

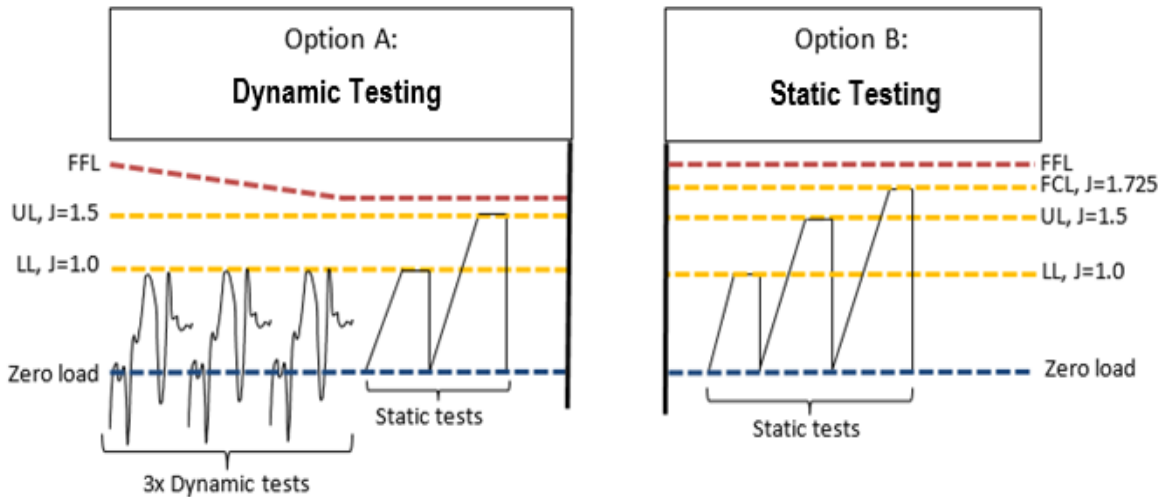


Figure 13: Current Sailplane Testing Philosophy (Van der Walt, 2020, p. 31)

Option A represents dynamic testing that requires a life factor of three, followed by static testing to ensure that the structure is still capable of handling the limit load and ultimate load. Note that the Fatigue Failure Limit (FFL) slope in Option A decreases for the spectrum tests. This is because of the weakening of the material, due to dynamic stress fluctuations over the spectrum timespan. After the spectrum tests, the FFL stabilizes at a constant value close to the Ultimate Load (UL).

When studying Option B, it can be noted that the FFL remains constant throughout testing. Option two represents static testing to limit load (LL), UL and Fatigue Compliance Load (FCL). When comparing the FFL of both methods, it can be noted that the FFL for Option B is theoretically higher compared to Option A.

However, if a further observation is applied, it can be noted that Option B requires static testing to be performed to the Fatigue Compliance Load (FCL), which is close to the FFL. The FCL test is used to compensate for the fatigue damage absent in static testing. Hence, both methods yield an approximate equivalent result compared to one another.

With the testing philosophy discussed, two certification testing examples are given to gain an in-depth understanding of current sailplane certification. The first example applies to the case where static testing (Option B) is required according to the EASA CS-22 sailplane legislation.

2.5.2 Example of a Dynamic Fatigue Test used for certification (Option A)

The well-known 1970's, PIK-20D sailplane underwent in-service fatigue testing as requested by the aviation governing body, EASA, to extend the service life of the sailplane (Soinne, 2015). The fatigue inspection program used during the testing was developed by the Finnish Transport Safety Agency (Trafi) under the stringent supervision of EASA. The fatigue inspection program was required to contain the following base parameters: inspection intervals, basic stress calculations, revised material, and fatigue tests. (Soinne, 2015, p. 4).

The damage state within the fatigue structure was assumed equivalent to the PM linear damage accumulation rule. The inspected components were carefully identified by Soinne (2015) for the fatigue inspection program, who assessed the consequences of component failure. Soinne (2015) stated in the original Trafi report that primary components would result in catastrophic structural failure of the sailplane.

Before each of the primary components were identified, Trafi had to segregate the components associated with composite sailplanes into multiple risk failure categories. (Soinne, 2015).

The inspection areas were identified by means of the maximum amount of quantifiable stress areas, highly loaded stress concentrations, literature strain gauge surveys performed on components of similar nature, comprehensive static and fatigue testing performed on identical components and actual service experience on identical components.

The primary components were identified by Trafi, as the Main Spar, Shear Web, and Spar Caps. (Soinne, 2015).

Trafi's inspection program is divided into two categories, namely: voluntary and mandatory objects. The voluntary objects are classified as common glider maintenance, for example, the aesthetics of the sailplane whereas the mandatory objects are classified as critical flight safety objects. (Soinne, 2015).

Trafi's inspection method comprises of five main steps and is in theory similar to the Saab fatigue inspection method. (Blom, 2005; Soinne, 2015). The five steps are as follow:

Step 1: derivation of the limit load spectrum.

Step 2: normal correction of coupons tested under fatigue loading subjected to the limit load spectrum.

Step 3: adjustment of data acquired in step 2 for the specified application.

Step 4: determination of the scatter factors for life and stress from the data acquired in steps 2 and 3.

Step 5: implementation of the chosen cumulative PM damage summation limit to determine when a specific component will fail under the implemented spectrum loading.

Soinne (2015, p. 11) stated that the fuselage of the PIK-20 is critical in buckling resulting in extremely low acting strains running through the fuselage meaning that the fuselage structural fatigue criticality is low compared to the wings of the aircraft.

Waibel and Schleicher (2002, pp. 56-61) stated that the wings of sailplanes are the critical components when considering fatigue due to the spar located in the wing being subjected to compression loading. Thus, the wing can be used to determine the fatigue life of sailplanes.

When determining the fatigue life of sailplanes, the load spectrum (Preferably KoSMOS II), effects of aerobatic flight conditions, stress concentrations present in the structure (especially in the root section of the wing), material properties (survivability as well as confidence limitations), the effect of the fibre type as well as the resin used in the structure, the altitudinal stress ratio (mainly $R = -0.55$ and -1 [conservative]), the life factor (for sailplanes a life factor of 3 is required), the cumulative damage parameter and summation technique ($D = 0.1$ [conservative] for the PM method) needs to be considered.

After all the necessary considerations are considered the fatigue life calculation can be performed to determine the safe life of the structure within a life factor of 3 followed by an actual fatigue test to validate the data.

The dynamic fatigue certification procedure illustrated in Figure 13 equivalent to Mileschkin *et al.* (1987, pp. 507-512) that is applied on the wing of a sailplane can be seen in Table 7.

Table 7: Dynamic Test Execution Procedure (Van der Walt, 2020)

Step	Instruction
1	Install the displacement, load strain and temperature transducers on the wings and spar of the aircraft.
2	Apply the zero load ($J = 0$) to calibrate and zero all the transducers mentioned in step 1.
3	Initialize data recording of the data loggers.
4	Run the load spectrum (KoSMOS II) for a complete test cycle (1).
5	Apply the zero load ($J = 0$), make sure all the sensors are at the calibrated origins.
6	Run the load spectrum (KoSMOS II) for a complete test cycle (2).
7	Apply the zero load ($J = 0$), make sure all the sensors are at the calibrated origins.
8	Run the load spectrum (KoSMOS II) for a complete test cycle (3).
6	Apply the zero load ($J = 0$), make sure all the sensors are at the calibrated origins.
7	Apply the limit load ($J = 1.0$) and hold the load for a minimum of 3 seconds.
8	Slowly release the load to the zero load ($J = 0$).
9	Apply the ultimate load ($J = 1.5$) and hold the load for a minimum of 3 seconds.
10	Slowly release the load to the zero load ($J = 0$).

It is possible to extend a sailplane's service life with actual operational flight testing and a complex inspection program, similar to that of Trafi (Soinne, 2017). The preferred dynamic certification method remains to conduct full-scale fatigue tests on aircraft structures using

specialised test benches consisting of various actuators sensors and a complex whiffle tree setup similar to that used for static testing (Mileshkin, *et al.*, 1987).

Test benches are mainly used for certification purposes due to the advantages over actual fatigue flight testing. The advantages include but is not limited to shortened test durations, faster concept to production implementation, safety orientated environment and experimental repeatability.

With fatigue testing, as well as, the modern certification procedures, static and dynamic fatigue certification of sailplane structures researched for literary purposes, the focus of Chapter 2 can change to the original MLS methodology.

2.5.3 Example of a Static Test used for certification (Option B)

In 2017, Jonker Sailplanes (M&D Flugzeugbau & Jonker Sailplanes, 2018) had to perform a full-scale static certification test to prove that the newly designed 18 m JS3 wing illustrated in Figure 14 met the EASA CS-22 design specifications.

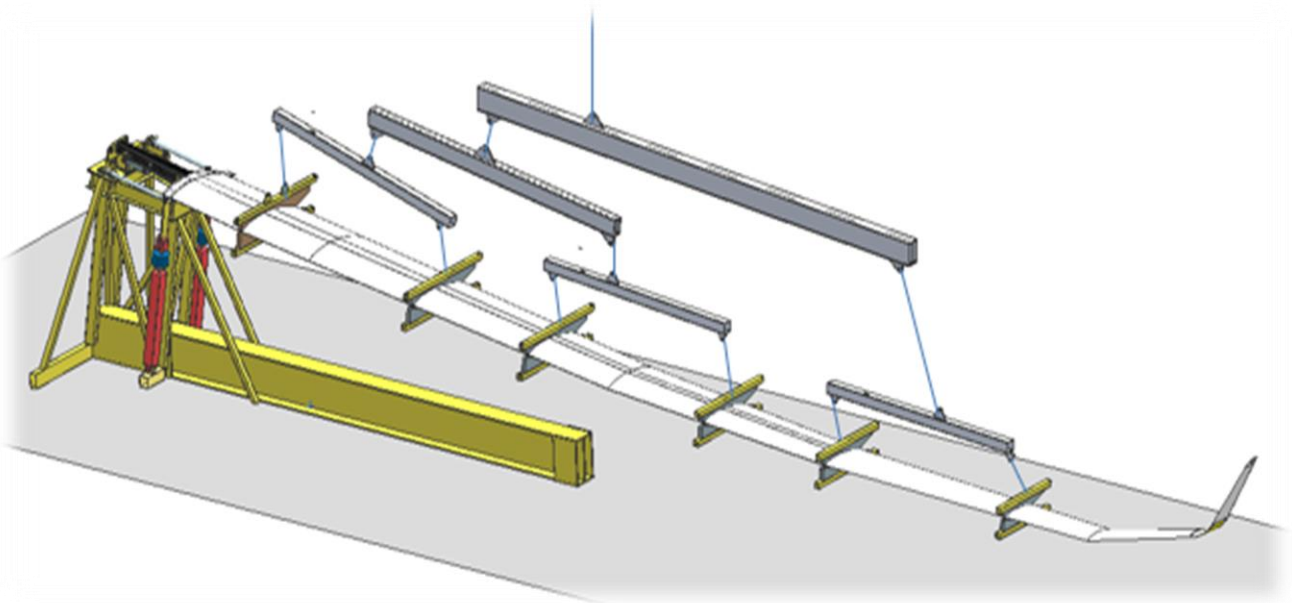


Figure 14: JS3 Wing Static Test (M&D Flugzeugbau & Jonker Sailplanes, 2018)

The JS3 wing had to conform to the CS-22 (EASA, 2009) paragraphs 22.303, 22.305 (a & b), 22.307 (a & b) and 22.601 as well as to I 4- FVK/91 – “Standards for structural substantiation of sailplane and powered sailplane components consisting of glass or carbon fibre reinforced plastics”.

In Table 8 the applicable paragraph is given with a brief description, quite similar to that found within the CS-22 document (EASA, 2009).

Table 8: Static Testing Applicable Certification Regulations

Paragraph	Description
Factor of safety (CS 22.303)	A factor of safety equal to 1.5 should be used when a factor of safety is unspecified.
Strength and deformation (CS 22.305 a)	The structure should support limit loads during testing and not deform permanently. The deformation at any loads including and below the limit loads should not impart safe operation of the structure controls. In other words, the controls within the structure should always remain free moving.
Strength and deformation (CS 22.305 b)	Ultimate loads should be supported by the structure without any form of failure occurring for at least a three-second time duration.
Proof of structure (CS 22.307 a & b)	For each critical load condition, the strength and deformation compliance should be proven via testing. Structural analysis may be used instead if and only if the tested structure conforms to a historically proven structure and or method. Substantiating load tests compliance must adhere to AMC 22.307a as well as subpart D of the CS-22.
General (CS 22.601)	Each questionable design detail including components influencing critical safety during operation must be established through testing. For example, complete wings, ,including control mechanisms, need to be tested.

The 18 m configured JS3 wing was tested under torsional, as well as, bending loads (M&D Flugzeugbau & Jonker Sailplanes, 2018) as perceived during the actual operation and had to

show compliance to CS 22,303 that demands a minimum factor of safety equal to 1.5 times the ultimate load (UL).

The critical load case objective was to prove the main wing spar and flange strength at maximum load conditions. The whiffle tree setup as illustrated in Figure 15 was used by Jonker Sailplanes (2018) to apply the load to the skin. The whiffle tree distributed the applied load to six different points predetermined by intricate calculations to replicate a bending moment and shear load as perceived during actual operation conditions.

In Figure 15 the wing test bench setup can be seen. The figure indicates the single applied force (F_z) implemented by a crane. The single applied load is distributed to the wing via the yellow whiffle tree setup.



Figure 15: Static Wing Test Setup Example

The tested 18 m wing had to support the required UL for at least three seconds at an operating temperature of 54 °C to prove that the wing is designed and built according to the CS-22 specification (EASA, 2009).

Jonker Sailplanes (2018) proved that the wing met the certification requirements by using three insulated thermometers placed strategically on the outer top wing skin, two loadcells with pistons to load the wing in torsion, two loadcells with a wiffle tree to apply the bending moment of the wing, six ultrasonic displacement sensors located under the wing to measure

the overall wing deflection under load, data logging equipment and heating elements with a temperature hotbox (oven) to heat the wing above 54 °C.

The sensor positions are showed in Figure 16. The temperature sensors were insulated from the environment, allowing the accurate reading of the wing skin temperature. The load cells used required accurate calibration to comply with the CS-22 according to EASA protocols.

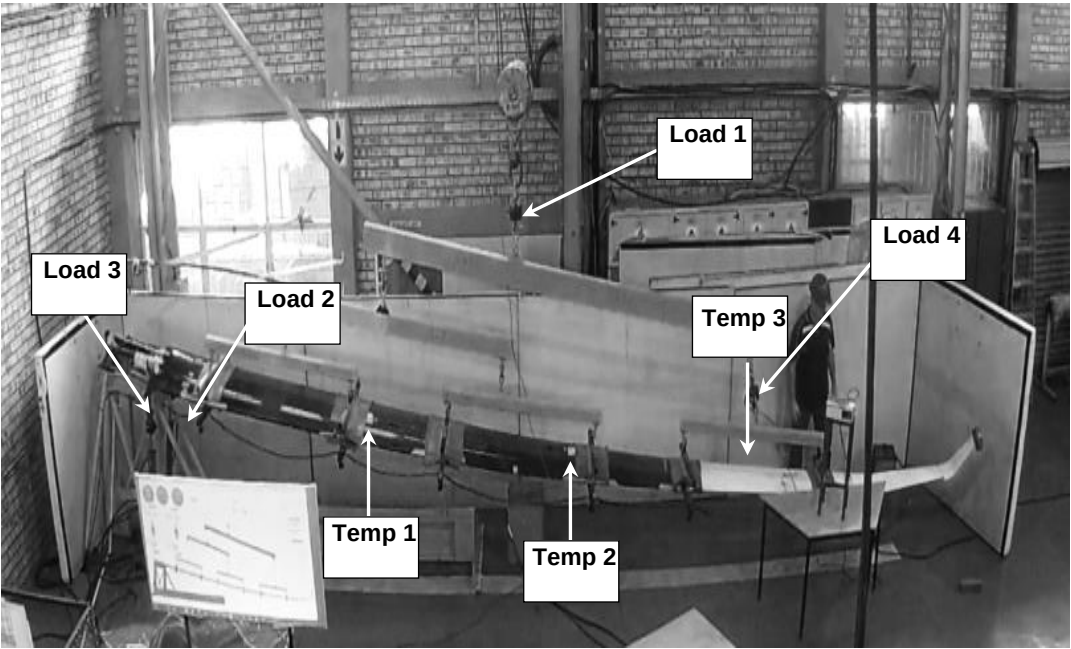


Figure 16: Measurement Equipment Positions (M&D Flugzeugbau & Jonker Sailplanes, 2018)

The positions of the spanwise displacement sensors that measured the vertical position between the wing and the calibrated zero point is shown in Figure 17. The displacement sensors measured the wing deflection with regards to the ground via ultrasonic frequency bands.

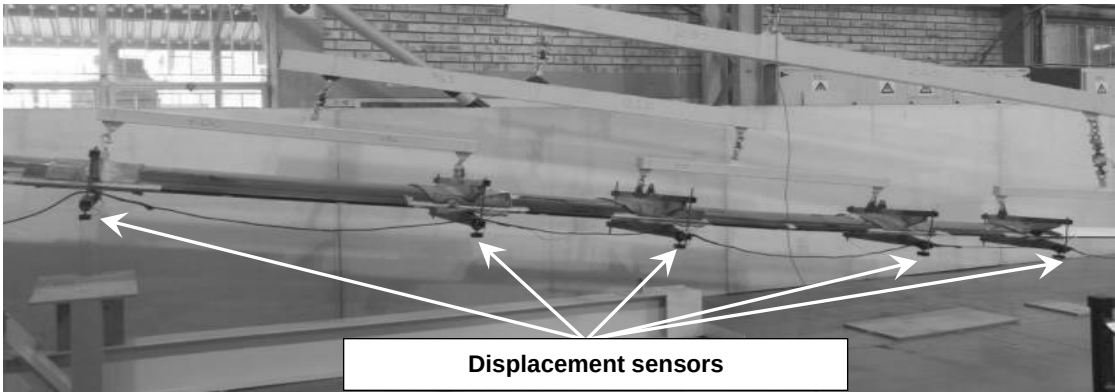


Figure 17: Displacement Sensor Positions (M&D Flugzeugbau & Jonker Sailplanes, 2018)

The test procedure used for the static certification test of the 18 m wing can be broken up into the following two different sections: setup and execution. The setup of the test was a daunting task, requiring a lot of preparation and thorough planning.

The wing had to be installed on a specially designed trestle illustrated in Figure 18, which had to be securely bolted to the floor. The trestle's main purpose was to provide the reaction loads at the wing root when the wing is subjected to a load applied by the overhead crane and whiffle tree load distribution setup.

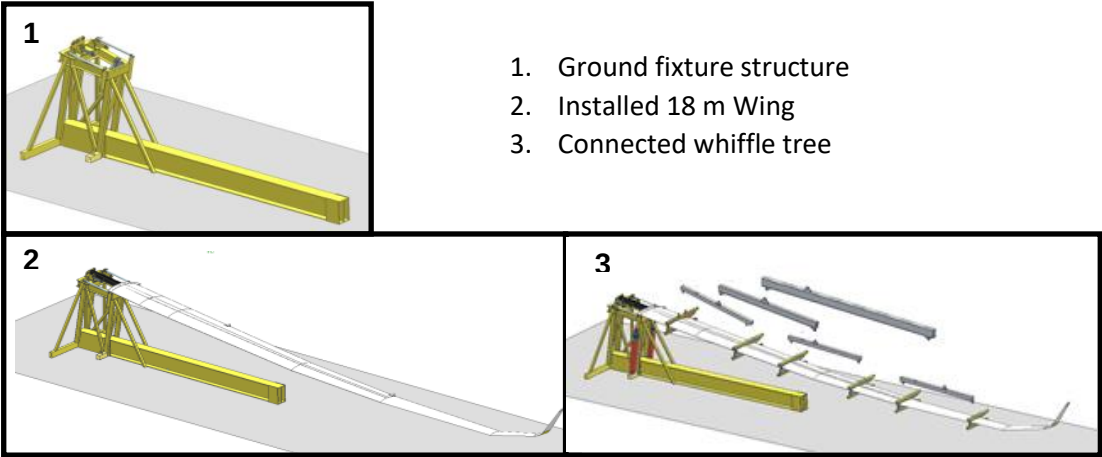


Figure 18: Wing Setup (M&D Flugzeugbau & Jonker Sailplanes, 2018)

After the wing was installed on the trestle, an insulated panel oven had to be constructed enclosing the entire wing. After the oven was constructed, the wing had to be heated to just over 60°C incrementally for 8 hours to prevent thermic shock of the wing skins and internal components.

When the wing reached a temperature of 60 °C, the oven was swiftly deconstructed and the testing commenced.

Figure 19 illustrates the oven used for the heating process of the wing. Note that due to the size of the oven, adequate insulation is required to reach the desired temperature.



Figure 19: Oven Setup (M&D Flugzeugbau & Jonker Sailplanes, 2018)

The test execution procedure can be seen in Table 9. The test execution requires great diligence since the deviation of the procedure could ultimately result in compliance failure.

Table 9: Static Test Execution Procedure (M&D Flugzeugbau & Jonker Sailplanes, 2018)

Step	Instruction
1	Install the displacement, load strain and temperature transducers.
2	Calibrate and zero all the transducers mentioned in step 1.
3	Assemble the oven and seal it off from the external environment with tape.
4	Heat the specimen using a temperature ramping procedure.
5	Initialize data recording of the data loggers and multiple cameras.
6	Disassemble the oven quickly when the skin temperature reaches 60°C. The skin temperature should not fall below 56°C.
7	Connect the whiffle tree setup.
8	Apply the limit load ($J = 1.0$) and hold the load for a minimum of 3 seconds.
9	Slowly release the load.
10	Apply the ultimate load ($J = 1.5$) and hold the load for a minimum of 3 seconds.
11	Increase the load incrementally from the ultimate load until structural failure occurs.

During testing of the wing, the following load phases needed to be reached successfully for CS-22 (EASA, 2009) conformity:

Phase 1: limit load (LL) ($J=1.0$).

Phase 2: ultimate load (UL) ($J=1.5$).

Phase 3: fatigue compliance load (FCL) ($J=1.725$).

Phase 4: final failure load (FL) ($J>1.725$).

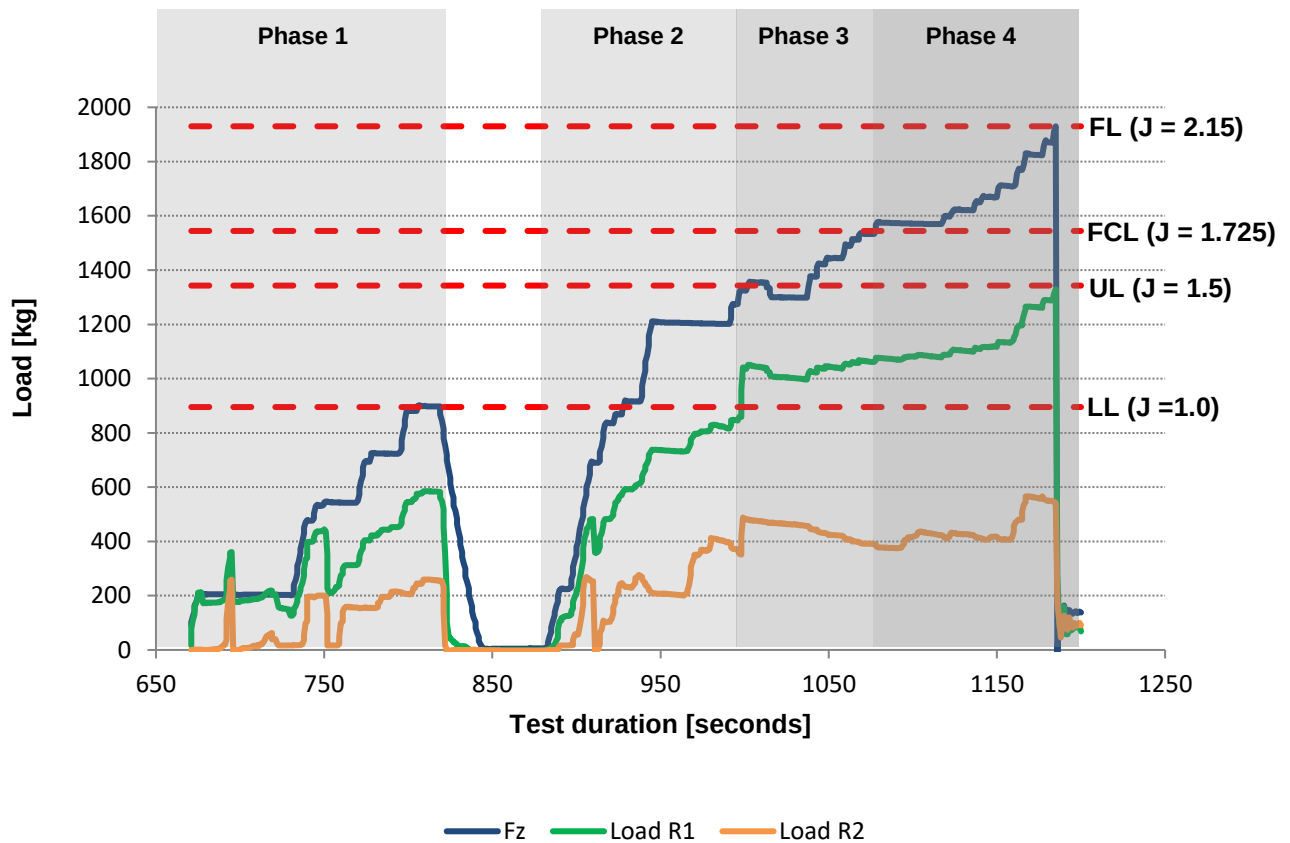


Figure 20: Static Test Load Duration Graph (M&D Flugzeugbau & Jonker Sailplanes, 2018)

In Figure 20 an illustration of the wing test results can be seen. The tested 18 m configured JS3 wing demonstrated that it could maintain loads above the fatigue compliance load, thus conforming to the CS-22 requirements stated in Table 8. The identified failure load (FL) obtained was $J = 2,15$.

2.6. Original MLS Method

Van der Walt (2020, p. 55) investigated if the workload of composite fatigue certification could be decreased, through the use of an alternative method not based on damage calculations, constant life diagrams (CLD) or stress life (S-N) diagrams.

The study of Van der Walt (2020) focused on attaining a relationship between CA and VA fatigue cycling by using an empirical approach. Van der Walt found that CA fatigue cycle testing can under ideal circumstances be used to reduce test duration of a component compared to VA fatigue cycle testing.

The reduction in testing duration is mainly owed to the increased load amplitude of the CA fatigue cycle ultimately accelerating the damage accumulation of a given component.

According to Van der Walt (2020), the approach to replace VA with CA cycling has been documented at least three times by Kensche (2002(a)), Trappe *et al.* (2010) and Rodzewics (2009). All three cases were strictly used for conceptional research and not approved by aviation authorities for actual certification of sailplanes. Van der Walt (2020, p. 47) continued to conduct literature research and found that the wind turbine industry utilizes an equivalent approach for certification purposes of wind turbine blades under the WISPER certification program.

In the wind turbine industry, the VA load cycle was altered significantly through the implementation of the load enhancement approach. The implementation of the load enhancement approach allowed low amplitude cycles to be intensified until all the load cycles mitigated towards a singular load intensity, in other words, a CA cycle envelope. The CA cycle later went on to be used in wind turbine blade certification. (Heijdra, *et al.*, 2013).

Van der Walt (2020) identified one possible significant flaw of the method used to replace VA cycles with CA cycles during certification of aircraft. The correlation between the different cycle types is unknown and therefore the method cannot be used with certainty to determine actual achievable flight.

The only identified way to determine the actual achievable flight hours from CA fatigue cycle tests is through the implementation of the PM damage accumulation method.

Kensche (2002(b), pp. 51-55) compared the damage accumulation of the two cycle types on identical specimens. The experimental setup is shown in Figure 21.



Figure 21: Sailplane Wing Spar Fatigue Test (Kensche, 2002(b))

The specimen tested resembled the spar that is typically used within sailplane wings.

The test results yielded valuable data with failure for the CA spectrum occurring at 10000 cycles. The test results for the VA spectrum were inconclusive with the specimen only tested to 72×10^3 Fh were the specimen showed no strength or stiffness degradation, possibly hinting at infinite life for the specimen. (Kensche, 2002(b)).

The 2010, study conducted by Trappe *et al.* was similar to that of Kensche (2002(b)). Trappe *et al.* compared a CA loading program with a variable loading program by conducting tests on wing sub-sections illustrated in Figure 22, which is categorized as a sub-structural elemental specimen according to the BBA. (Trappe, *et al.*, 2010, pp. 94-99).

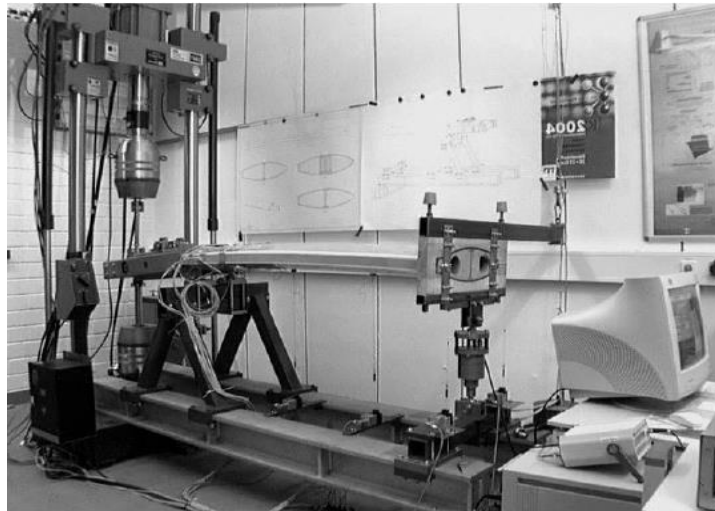


Figure 22: Wing Fatigue Test (Trappe, *et al.*, 2010)

The study objectives were to investigate the damage present in FRP structures subjected to fatigue loading, if the damage between CA and VA cycle loading was equivalent and finally if the initialisation of microcracking indicated the initialisation of total laminate degradation. (Trappe, *et al.*, 2010). The 2010 study of Trappe *et al.* yielded good results only for the first two objectives, with the damage mode at 120 CA cycles equivalent to that of 36000 VA cycles at a maximum strain of 6.4 % for both tests subjected to bending and torsional loading. After the damage mode equivalence was detected, the torsional load was removed, and fatigue testing continued for the different load cycle types. (Trappe, *et al.*, 2010).

The test was conducted until 20×10^3 cycles were reached for CA cycle loading and for the VA loading the test was conducted until 42×10^3 Fh was reached. After completion of the study, it could not be concluded that the initialisation of microcracking indicated the initialisation of total laminate degradation. (Trappe, *et al.*, 2010).

Rodzewics (2009, p. 457) conducted a study where the CA method was used to investigate the high bearing loaded region located within sailplane spars. Rodzewics investigated the residual strength of both GFRP and Carbon Fibre Reinforced Polymer (CFRP) specimen

subjected to fatigue loading and as result identified a strength threshold limit after which laminate degradation occurred rapidly.

Rodzewics (2009) went on to propose a method based on the CA test method which drastically reduces the costly Fatigue Compliance Phase (FCP) phase 3 testing time as well as implementation cost. The equivalent load cycle approach is used to reduce the cycle count in the variable loading test spectrum by eliminating all the load cycles causing negligible damage. (Rodzewicz, 2009, p. 455).

The overall finding of the 2009 Rodzewics study was that through the implementation of the equivalent life cycle approach, the load cycle count can be reduced to a mere 4882 cycles from a full spectrum consisting of $1,75 \times 10^6$ cycles. (Rodzewicz, 2009, p. 455).

After investigation of the three documented cases, Van der Walt (2020) concluded that a probable approach for future sailplane certification is through the use of the PM damage accumulation combined with the CA full-scale fatigue spectrum. This is backed by the findings of Kensche (2002(a), p. 42) where the achieved CA cycles were equivalent to VA cycles in the infinite flight hour range. When compared, the failure modes of both constant and variable loading cycles are similar according to the findings of Trappe *et al.* (2010, pp. 52-58). The CA test cycle method may be used, according to Rodzewics, for accelerated fatigue investigation of regions that are known to be critical. (Rodzewicz, 2009, p. 457).

Van der Walt (2020, pp. 55-60) proposed a new method that is empirical of nature. A certain specified amount of VA cycles is combined with an unknown amount of CA, until a composite specimen reaches failure for the proposed method, later known as MLS method.

The MLS method is based on the PM assumption used in aviation certification and states that damage accumulation remains linear irrespective of the loading order.

The exchange of cycles between VA and CA cycles would, under ideal circumstances, reduce the fatigue testing duration exponentially, hence drastically reducing the certification cost. The hypothesis formulated by Van der Walt (2020, pp. 55-60), states that linearity exists between the amount of CA and VA cycles for composite structures subjected to fatigue testing, resulting in the exchange of a certain amount of VA cycles with CA cycles.

If the Van der Walt (2020) hypothesis can be validated, the possible future implication is that a new composite fatigue certification method for sailplanes can be presented to the governing bodies including Federal Aviation Administration (FAA), South African Civil Aviation Authority (SACAA) and LBA.

The proposed empirical MLS method consists of the identification of critical structural elements found within sailplanes, data acquisition of the VA and CA relationship for the identified critical structural elements employing coupon and elemental testing as prescribed by the building block approach as well as the use of the most conservative VA/CA ratio to perform full-scale CA fatigue tests on sailplane wings as required by certification standard.

According to Van der Walt (2020, pp. 63-71), experimentation was conducted on specimen representing a typical Shear Web construction found within the wing of a sailplane to prove the original MLS method's hypothesis validity. In 2002, Kensche stated that, the Shear Web located within the spar of typical FRP sailplane were more susceptible to fatigue damage when compared to the Spar Cap. The Shear Web can thus be seen as the fatigue life determinator. (Kensche, 2002(b), p. 54).

The Van der Walt (2020) specimen can be seen in Figure 23. The tested specimen consisted of carbon roving Spar Caps located spanwise at the top and bottom of the specimen, three equally spaced brass bushes simulating the spar locking points located spanwise, two foam inserts inserted simulating the foam found within a spar to combat buckling located spanwise between the brass bushes and a glass layered body.

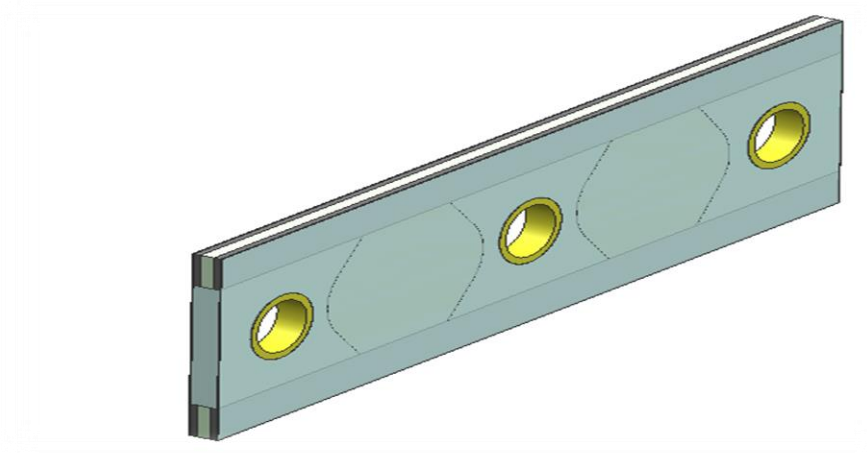


Figure 23: Shear Web specimen (Van der Walt, 2020)

The problem with the Van der Walt Shear Web specimen, however, is the manufacturing complexity as illustrated in Figure 24. The specimen can be classified according to the building block approach hierarchy, as a complex elemental level specimen.

The specimen geometry allows gathering of realistic data since wing structures of aircraft are normally created within a tolerance that allows for small imperfections. A coupon level specimen is thus better for testing of the MLS methodology, due to better accuracy, since the coupon specimen types are in general easier to create within tighter tolerances.

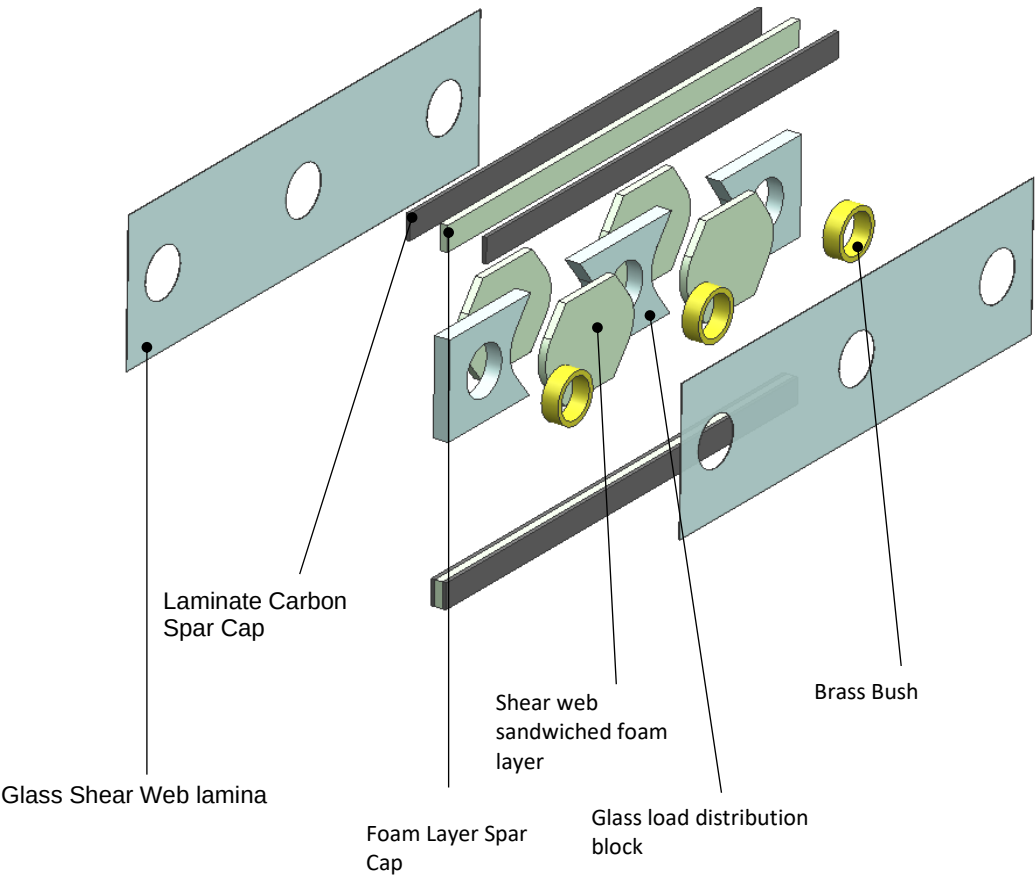


Figure 24: Shear Web Specimen Complexity (Van der Walt, 2020)

The influence of the imperfections found within the original elemental type specimen can be seen in the 15 % VA scatter band found by Van der Walt during the original MLS study. The scatter band however still allows the observance of a correlation between the two different cycle types; CA and VA spectrum as illustrated in Figure 25.

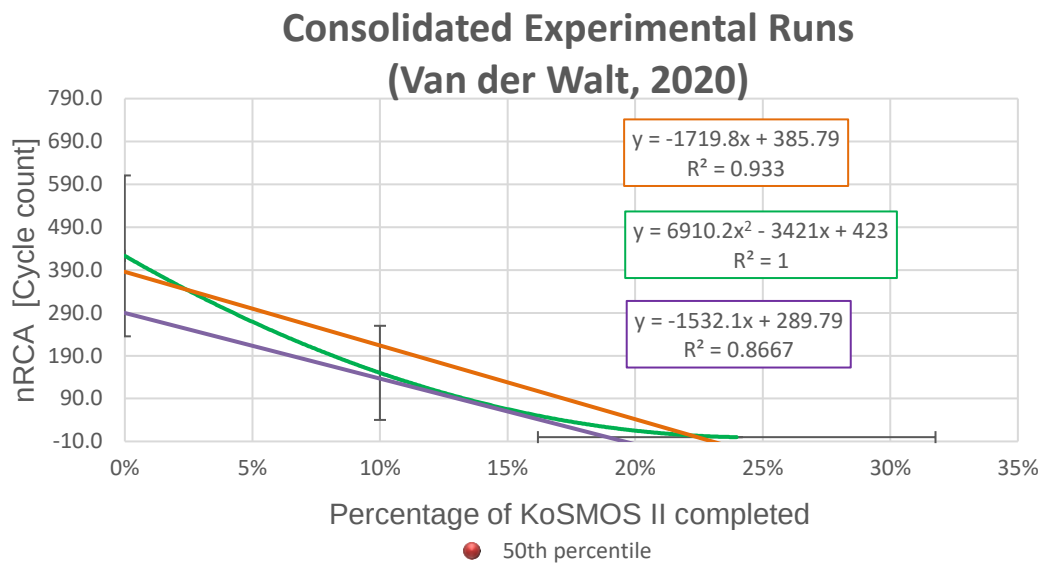


Figure 25: Consolidated Experiment Runs Results adapted from Van der Walt (2020, p. 83)

One of the main findings of Van der Walt’s (2020) work was that a linearity correlation between the amount of CA and VA cycles is probably not accurate. The linear data regression yielded a value of $R^2 = 0.8933$. A second-order function yielded a regression value of $R^2 = 1$. Van der Walt suggested more data test points in the middle range to improve the accuracy. The second-order quadratic fit from Van der Walt (2020) is shown in Figure 26.

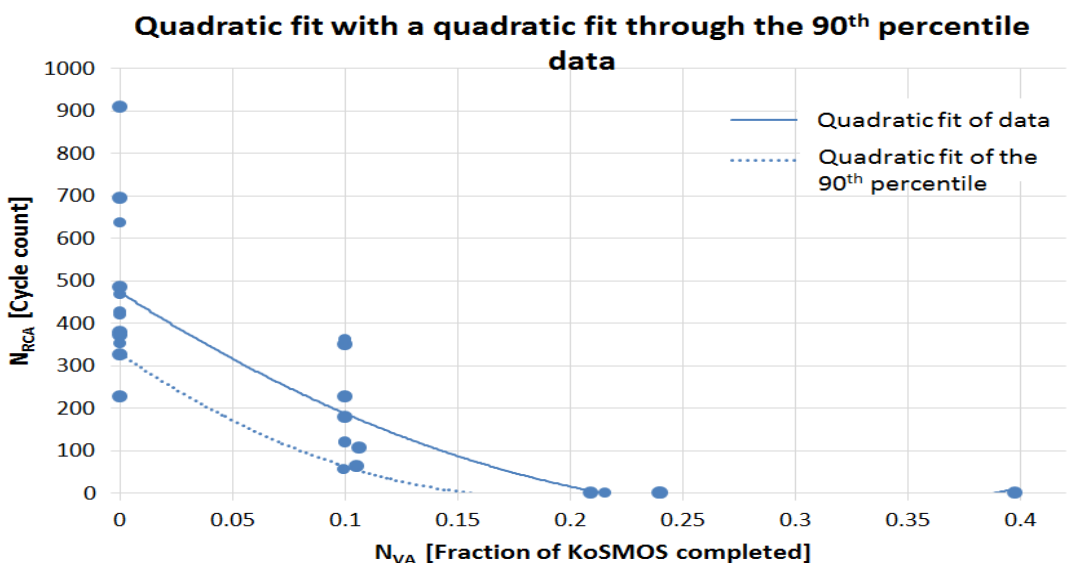


Figure 26: Data trend (Van der Walt, 2020, p. 84)

Although the original MLS study yielded intriguing results the hypothesis was not validated resulting in uncertainty on whether the MLS methodology is suitable for future certification use on FRP sailplanes. The Van der Walt hypothesis hence needs to be evaluated, adjusted, and reimplemented according to Van der Walt (2020, pp. 88-92).

The constraints faced by Van der Walt (2020) that lead to the reformulation of the hypothesis include but aren't limited to:

- Testing on coupon level as required by the Building Block Approach (BBA) as used in aviation was not performed.
- Data acquisition was only at high load intensities where high strain is not applicable for sailplanes.
- The damage state of the equivalent CA and VA cycles remained unknown during testing.

The first identified constraint could be solved by using a specimen that conforms to coupon level according to the BBA and Van der Walt (2020). A proposal to investigate the ASTM D 3039 test procedure specimen was made by Van der Walt (2020). It was noted that the specimen required adaption for fatigue testing subjected to an R - ratio of -0.55 as used in the original MLS study. (Van der Walt, 2020, p. 61).

The second identified constraint could be corrected by Van der Walt (2020), who proposed that tests on the coupon type specimen should be conducted at loads applicable to sailplane flight loads. Lower loads would allow for realistic data and damage rates.

The third constraint, identified by Van der Walt (2020), is a problem worth solving. Residual strength tests were proposed for specified intervals, to determine the damage state present within the specimen after completion of a few thousand cycles. The residual strength test would allow data to be gathered on fatigue resistance of the specimen since one could, in theory, see strength mitigation over the fatigue testing span. (Van der Walt, 2020, p. 89).

With the Van der Walt-MLS proposal discussed, a literary conclusion can follow that focus on the historical sailplane legislation development, current sailplane certification and the future of the MLS method for sailplane certification.

2.7. Conclusion

A future for the MLS methodology in sailplane certification exists only if the technique is validated. The MLS methodology needs to be accurate, cheaper and less time-consuming than the traditional dynamic method for certifying sailplanes with certifiable life above 12000 Fh to make the MLS technique the preferred future technique for sailplane fatigue certification. The Van der Walt - MLS methodology research outcome concluded that a nonlinear correlation exists between CA and VA cycle loading for flight loads far above that experienced by sailplanes during flight. The non-linear function can be seen when investigating Van der Walt's experimental data. Van der Walt recommended that a load integer ($J = 1.0$) closer to that of sailplanes during flight should be used where the ultimate failure load (FL) ($J = 2.0$) equals the ultimate tensile strength of the specimen. (Van der Walt, 2020).

It was further determined that at least three data interval points are required to determine the relationship between CA and VA loading spectrums. With an increase in data interval points the relationship accuracy between the different cycle spectrums will drastically increase making it easier to predict with the CA spectrum when a specimen subjected to realistic flight loading conditions will fail.

The methodology validity for sailplane certification can be determined through further research on coupon level according to the BBA. Further research should entail CA fatigue testing, VA fatigue testing and residual strength validation on GFRP specimen at loads applicable to sailplanes in flight, i.e., 50 % UTS.

3. METHODOLOGY

In Chapter 3 the methodological approach used is discussed regarding the hypothesis, specimen design, testing procedure and validation of the MLS methodology.

3.1. Hypothesis Formulation

The dynamic VA fatigue testing method used in modern aviation for fatigue certification purposes is time-consuming and expensive. An alternative technique proposed by Van der Walt (2020) to reduce the fatigue testing time duration of sailplanes could determine the relationship between CA and VA load spectrums without the use of stress-life (S-N) diagrams, constant-life (CLD) diagrams and PM calculations.

In the empirical MLS method, a certain amount of VA cycles could hypothetically be replaced by an equivalent amount of CA cycles. The equivalent number of cycles is determined for a specific material and structure, be it a primary structure or a representative test specimen.

The proposed empirical MLS method conforms to the PM damage accumulation rule, being that the relationship between damage accumulation of the CA and VA testing is linear in nature irrespective of the loading sequence.

The equivalency between different cyclic sequences can be hypothesised through the assumption that two different loading sequences that both result in the failure of identical test specimen comply with the PM damage accumulation rule. Due to this compliance, a linear relationship is established between these different cyclic sequences.

A mathematical representation of the hypothesis ensues. Specimen type A is tested at a CA load sequence type until failure at cycle count n_x and identical specimen B is tested at VA load sequence type until failure at cycle count n_y . Equivalency can be assumed between the different cyclic sequences in Equation 3 since both identical specimens resulted in the same type of failure at different cycle counts.

$$n_{x(CA)} = n_{y(VA)} \quad (3)$$

Due to the PM assumption for damage accumulation, Equation 3 can be rewritten in the linear equation form as Equation 4.

$$(CA) = \frac{n_y}{n_x} (VA) \quad (4)$$

Equation 4 can be visualized in Figure 27 which represents the equivalency between the amount of CA and VA cycles until failure of identical components at a specific loading magnitude.

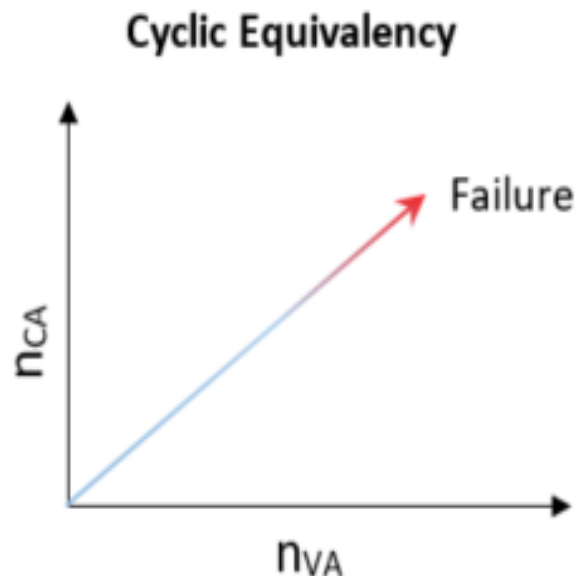


Figure 27: CA - VA Cyclic Equivalency adapted from Van der Walt (2020, p. 56)

By using the cyclic equivalency as illustrated in Figure 27, only one data point is required to determine the equivalency between CA and VA cycles. By comparing the proposed MLS empirical methodology to the conventionally used constant life diagram that exists from a minimum of three SN diagrams with multiple data points, a major reduction in fatigue composite lifetime estimation time can be achieved.

A further reduction in testing duration can be achieved without testing both CA and VA sequences to failure. This proposed method is formulated as follow:

Test 1: Choose a CA loading sequence that shall result in failure in the minimum amount of time that applies to sailplane flight loads. Run the CA loading sequence on a specimen until failure is reached at point n_1 . Point n_1 is the cycle count where a specimen fails subjected to only CA fatigue.

Test 2: Choose a VA loading sequence that applies to sailplane flight loads and run the sequence until a predetermined transition point t_2 where the loading sequence is subsequently changed to a CA loading sequence until failure is reached at point f_2 . Point f_2 is the cycle count where the specimen subjected to CA testing fails. Point t_2 is the cycle count where the fatigue test cycle type is changed from VA to CA.

Both tests are assumed equivalent due to both tests resulting in failure for identical specimen. The formulation of this method follows:

$$n_1 * (CA) = t_2 * (VA) + f_2 * (CA)$$

$$n_1 * (CA) - f_2 * (CA) = t_2 * (VA)$$

$$(n_1 - f_2) * (CA) = t_2 * (VA)$$

$$(VA) = \frac{(n_1 - f_2)}{t_2} * (CA) \quad (5)$$

Equation 5 can be visualized in Figure 28, which presents the cycles to failure interpolation from the amount of CA cycles towards the MLS cycles of identical components at a specific loading magnitude to determine the theoretical VA cycle failure count.

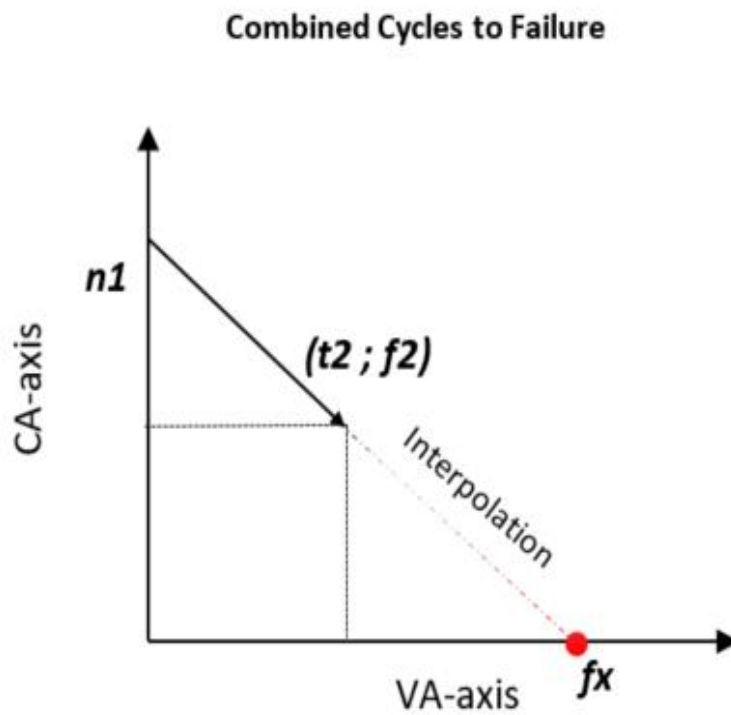


Figure 28: Combined Cycles to Failure Graph adapted from Van der Walt (2020, p. 60)

Van der Walt (2020) concluded in his study that the PM method for damage accumulation is not necessarily applicable, due to a nonlinear relationship that existed at high loads unrepresentative of sailplane flight loads, resulting in the required evaluation of the MLS method feasibility for sailplane fatigue lifetime estimation.

Van der Walt’s hypothesis (2020, pp. 55-57) can be evaluated through the addition of multiple combined cycle test points to the existing Combine Cycles to Failure graph by performing experimentation on a simple unidirectional glass fibre reinforced polymer specimen at sailplane applicable flight loads.

A representation of the revised hypothetical Combine Cycles to Failure graph can be seen in Figure 29.

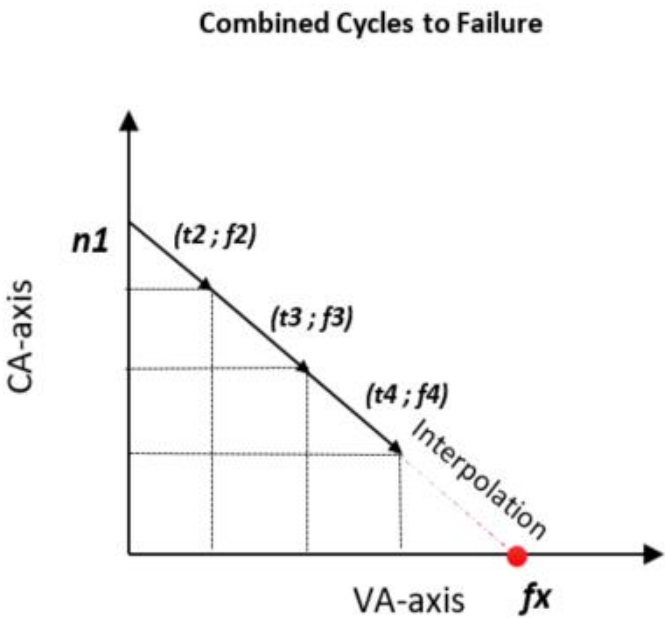


Figure 29: Revised Combined Cycles to Failure Graph adapted from Van der Walt (2020, p. 61)

The proposed Combine Cycles to Failure method illustrated in Figure 29 uses a coupon type test specimen that is tested utilizing VA sequencing until it reaches a predetermined cycle transition point before failure where the load sequencing method then changes towards CA sequencing until the specimen reaches failure.

This method is repeated for a specific material, at different life intervals, to accumulate the damage transition point. The CA cycles for the combined cycle is henceforth called Residual Constant Amplitude (RCA) cycles where the combined load sequencing cycle is referred to as the MLS experimental procedure.

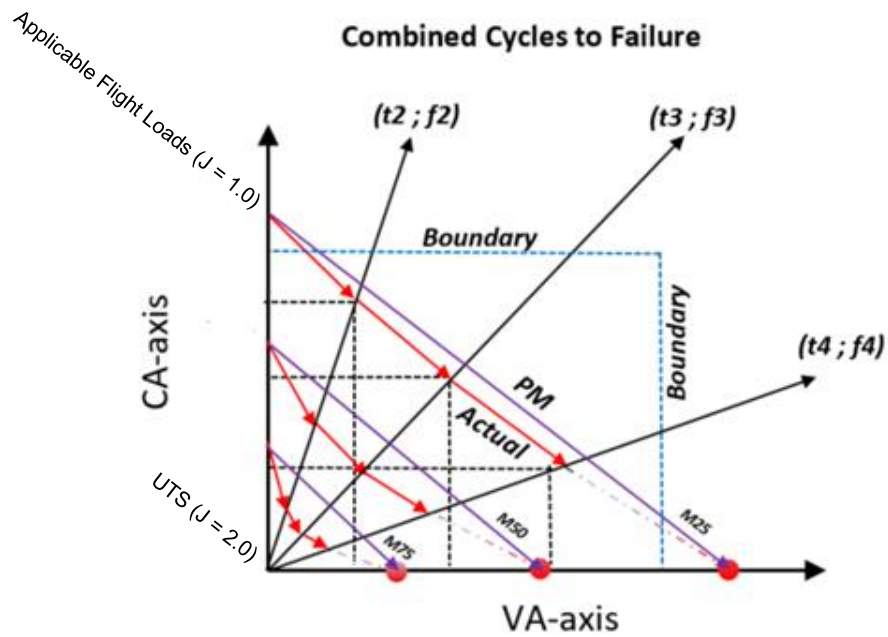


Figure 30: MLS Graphical Hypothesis adapted from Van der Walt (2020)

The MLS procedure can be used successfully in practice if the RCA cycle slopes as illustrated with red arrows in Figure 30 at different load intensities (J) converges towards the purple PM linear line when the load intensity is decreased to that applicable of sailplane flight loads.

The hypothesis of the dissertation can now be concluded. When conducting fatigue tests with the MLS methodology a certain amount of VA can be exchanged with an equivalent amount of CA cycles, due to the correlation between the different data cycles that tend to converge linearly at loads applicable to sailplane flight loads, i.e., 50 % of the UTS.

3.2. Specimen Design and Manufacturing

According to Van der Walt (2020), a specimen categorized as a BBA coupon level was required for evaluative purposes of the original MLS methodology. The first step was to study “American Society for Testing and Materials” (ASTM) standards used for testing of GFRP specimen subjected to fatigue loading, be it: compression-compression, compression-tension or tension-tension fatigue testing.

The most suitable ASTM standard identified was ASTM D 3039 (ASTM International, 2004). Note that the ASTM D 3039 is prone to buckling under compressive testing due to the long gauge area.

3.2.1 Material Selection

Due to the multiple ways a fibre reinforced polymer specimen may fail during testing, it is advised to closely follow the ASTM D 3039 standard, to ensure that the specimen failure results in a small scatter band.

Although, multiple damage mechanisms could occur during operation, FRP composites are adequate replacements for materials used conventionally in aviation structures, like aluminium, steel, and/or even wood. FRP materials offer enhanced physical properties like low density, high modulus of elasticity, high tensile strength, corrosion resistance and low a low thermal expansion coefficient a fraction of the weight of conventional materials, making FRP materials ideal for aviation structures. (Kaddour, *et al.*, 2007).

FRP composite materials consist of fibres and a matrix. The fibres carry most of the load and the matrix ensures interaction between fibres, while carrying a minimalistic load. Fibre materials usually consist of glass, carbon, steel or polymers. Matrix materials are resins, rubber, ceramics and concrete. (ETH Zurich, 2016).

Special emphasis will be on GFRP composite specimen, with a 90-degree bidirectional plain weave fibre pattern, as used in light to medium-sized aircraft for various components for this dissertation.

The properties of the selected sailplane FRP material can be seen in Appendix C.

3.2.2 Specimen Design

The specimen standard used was ASTM D 3039 with a short gauge length that includes two notches in the middle region of the gauge length.

The two small notches are incorporated in the gauge area creating stress concentrations that allow for more accurate life cycle failure rates, due to the failure area being pre-defined as the weakest portion of the specimen. The specimen geometry can be seen in Figure 31. The concern with this specimen is the manufacturing accuracy since variation in gauge width can influence the life cycle failure rate drastically due to varying UTS of the various specimen.

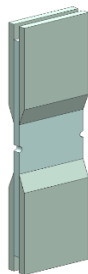


Figure 31: Notched Specimen ASTM D 3039

The theoretical buckling required calculation with the specimen design identified to ensure that the specimen would survive a fatigue R-ratio equal to -0.55 as associated with the KoSMOS II envelope spectrum. When applying the Euler Buckling equation for a specimen with the maximum allowable stress (Q) being 250 N/mm² (EASA, 2009), it is seen from Table 10 that the long gauge – notch-less specimen is prone to buckling under a compressive force. The specimen buckling is due to the specimen gauge area being too long compared to the specimen thickness and width.

Table 10: Euler Buckling of Concept Specimen

Specimen	Critical Load (P_{cr})	Ultimate Compressive Load ($F=Q \cdot A$)	Outcome
Long Gauge (Notch-less)	3,26 [kN]	10 kN	Buckles
Notched	257,81 [kN]	9 kN	Buckling Safe

The overall design limitations according to ASTM D 3039 are classified as specimen bonding, tab design, specimen laminate dimensions, specimen gauge area dimensions, and induced stress concentrations. A detailed description of the design limitations as per final detailed specimen design are given in Table 11.

Table 11: Applicable Design Limitations

Design Limitation	Description
Specimen Bonding	The bonding adhesive of the specimen is acrylic glue. This adhesive was chosen to eliminate misalignment of the tab thickness that could cause unwanted stress concentrations.
Tab Design	The specimen tabs need to be chamfered at 20 ° to ensure minimal stress concentrations exist where the tabs meet the gauge area. A stress of 500 MPa is exerted from the MTS® grips on the tabs of the specimen to ensure that no slippage during testing occur.
Specimen Laminate Dimensions	9 layers were determined necessary to support a tension load of 12 kN for the chosen material with a fibre ratio of 50 % that results in a laminate thickness of 2 mm.
Specimen Gauge Area Dimensions	With fatigue testing duration in mind, it was determined that an 18 mm gauge width would give an optimal result for the required specimen test lifetime. The gauge length of the specimen needs to be as short as possible to prevent buckling under compressive loads. The gauge length should be in between 15 and 20 mm in total length to ensure fracture before buckling if the specimen gauge thickness is 2 mm and the gauge width is 18 mm.
Induced Stress Concentration	In order to ensure that the specimen fails in the middle section of the gauge length and not at the tab ply-drop two small notches need to be inserted of radii 2 mm each in the middle section while maintaining a gauge width of 18 mm. Thus, the specimen tab width is increased to 20 mm. The notches induce a stress concentration in the middle section of the gauge area causing primary failure since the specimen tends to break at its weakest geometrical points.

The final specimen design can be seen in Figure 32.

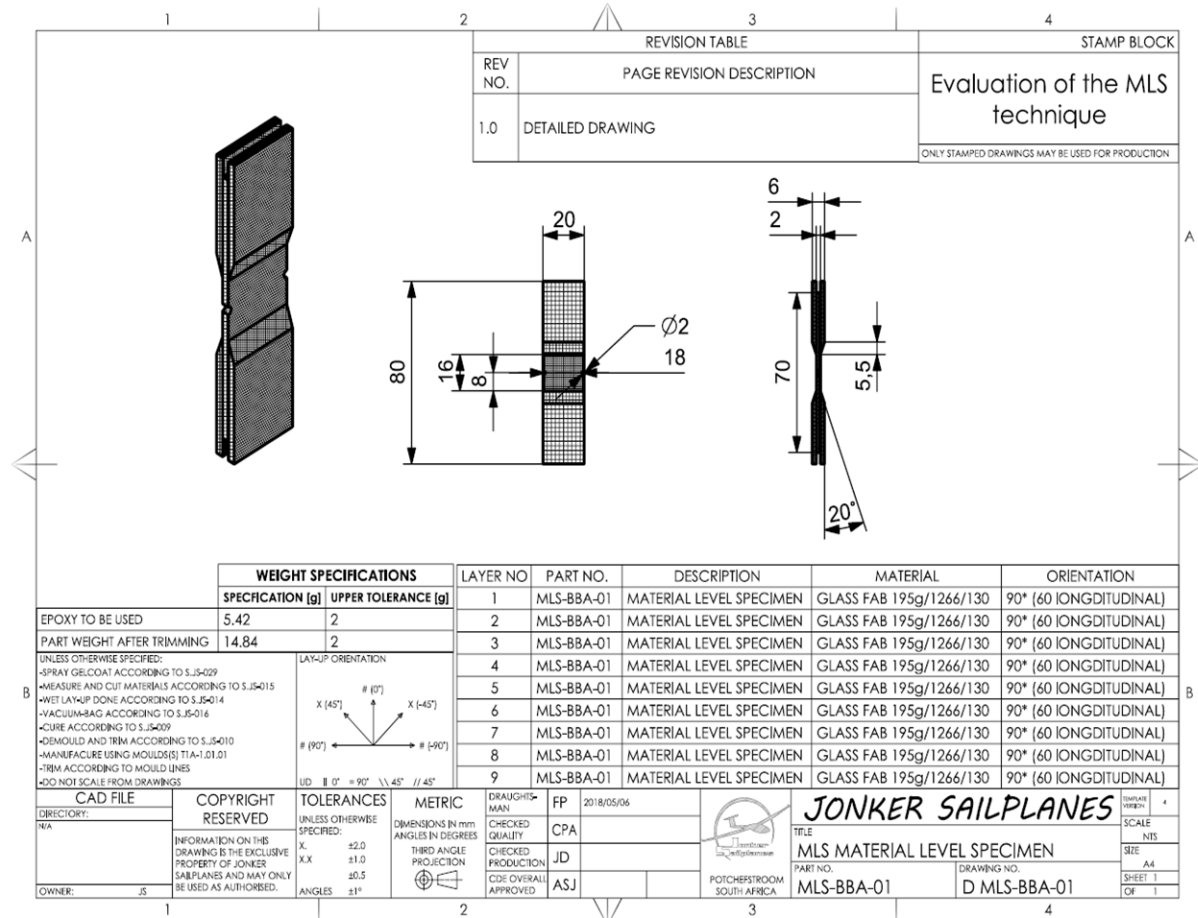


Figure 32: Final Specimen Design

It can be seen in Figure 32 that the specimen consists of 9 layers 90 ° orientated glass fibre layers where 60 % of the fibres in each layer are orientated longitudinally. The approximate specimen weight is 14,84 g and the approximate fibre volume ratio is 63,5 %.

3.2.3 Specimen Manufacturing Selection

The machinery used for manufacturing of the specimen was chosen specifically to simulate the equipment used in a sailplane manufacturing facility for the manufacturing of FRP sailplanes. Machinery required to manufacture the specimen sub-components are listed below:

- Band Saw (Blade For FRP);
- Industrial Belt Sander;
- Industrial Press Drill (1250 RPM minimum);
- Composite Laminate Press.

The Composite Laminate Press was used with eight G-clamps and eight 2 mm spacers to create an uniform laminate. The Band Saw was used to cut the specimen tabs and gauge to with 0.5 mm of the final shape. The Industrial Belt Sander was used to chamfer the specimen tabs at 20° and reduce the gauge width of the specimen to within 0.2 mm without causing large fractures on the sides of the gauge area. The Industrial Press drill was set to 1500 RPM with a 2 mm diameter Dremel® bit to create the two notches required, careful attention was given to align the notches.

3.3. Experimental Data Influencers

The tests that were required can be divided into static- and dynamic testing. Static tests include tensile- and compression testing, which were mainly used to determine the material UTS and compression strength. Dynamic tests include CA - and VA testing which are used to predict the fatigue life of the material. Two additional tests that can be seen as a combination of static and dynamic testing were required for validation of the dynamic test results. These additional tests are the residual strength and residual stiffness tests which may be associated with current sailplane certification.

Before any actual testing of composite specimen could commence, careful attention was given to obtain accurate results. Attention was especially given to gripping effects, specimen orientation, specimen removal, exerted clamp pressure and dimensional inaccuracy. A brief explanation of each influencer ensues.

3.3.1 Gripping effects

When a specimen is clamped in the hydraulic wedge grips, careful attention should be given to gripping effects on the specimen tab area. The machine clamping jaws should be aligned before insertion of the specimen. The specimen should be inserted in such a way that an equal pressure is exerted on the tab area by the jaws to minimize slipping during testing and unnecessary stress concentrations in the specimen.

3.3.2 Orientation of the Specimen in the clamps

The specimen orientation in the clamps is vital to accurately obtain data since load bearing fibres are present in the specimen. If a specimen is orientated at an angle exceeding even 1° the force carrying capabilities of the specimen can be drastically reduced due to different off plane failure mechanisms that could occur during testing. (Bailey & Lafferty, 2015).

3.3.3 Removal of the specimen

Removal of the specimen during testing should be avoided at all cost, due to additional stresses that may be exerted during the removal and re-insertion phase influencing the results drastically. (Bailey & Lafferty, 2015).

3.3.4 Tab bonding and exerted clamping pressure

Specimen tab bonding should be done in such a way that the tabs are parallel to each other, unsymmetrical tabs may lead to the development of eccentric forces or even worse stress concentration development. Clamping pressure should also be set at a specified amount to reduce stress concentration abnormalities in the tab area. (Bailey & Lafferty, 2015).

3.3.5 Specimen dimensional accuracy variation

A slight variation in dimensional accuracy may have a significant effect on the resulting outcome, due to the specimen load bearing capability being equivalent to the number of fibres present in the specimen. If a specimen consists of more than usual fibres it would most likely be stronger and vice a versa. (Bailey & Lafferty, 2015).

3.4. Test Apparatus

The test apparatus used to conduct the required tensile, compression, VA, CA, MLS, Residual Stiffness and Residual Strength tests include:

- MTS Landmark® 100kN servo hydraulic Test System;
- Digital Image Correlation System (DIC);
- I-core 7 Computer running Microsoft Excel VBA.

3.4.1 MTS Landmark® 100kN Servo Hydraulic Test Bench

The MTS Landmark® 100 kN servo hydraulic test bench illustrated in Figure 33 is produced by the MTS System Corporation. The hydraulic servo actuator test bench was used to conduct all fatigue tests required for this study since it is a fully programable servo hydraulic actuator, making it perfect for both static and dynamic testing of the MLS coupon specimen. The specific model type used to conduct tests with for this study is the 370.10 model with the Series 647 hydraulic wedge grips.



Figure 33: MTS Landmark® 100 kN Servo Hydraulic Test Bench (MTS Systems Corporation, 2004)

The MTS Landmark® 100 kN consist of a single load cell, linear potentiometer and two Series 647 hydraulic wedge type clamps. The calibration certificate of the machine used is presented in Appendix A.

The specification of the machine used according to MTS Systems Corporation (2004) is defined in Appendix D.

The operational manual for the test bench can be obtained from MTS Systems Corporation (2004). The software used was the (2019) MTS TestSuite™ Multipurpose Testing Software.

3.4.2 Digital Image Correlation System

The Digital Image Correlation system (DIC) is classified as an optical stress-strain measurement technique that has several advantages over conventional techniques, which according to McCormick and Lord (2010) include:

- Non-contact;
- Cost effective;
- High accuracy;
- User friendliness;
- Wide application range.

The DIC system illustrated in Figure 34 works by comparing digital photographs of a specimen taken throughout different phases of testing capturing the specimen displacement with high accuracy. The DIC software, MTS Advantage™, used with the DIC system tracks pixelated blocks on specimen surface and then converts the data to a multi-dimensional vector field allowing for the creation of strain maps for a specific specimen. (McCormick & Lord, 2010).

The DIC was only used to measure the residual stiffness for validation purposes.



Figure 34: Digital Image Correlation System (DIC)

3.5. Testing Procedures

The tests required to achieve the objectives of this study can be classified into the following three categories: static testing, dynamic testing and combined testing.

3.5.1 Assumptions

The assumptions made for the experimental portion of this dissertation includes, but are not limited to:

- The damage, as well as failure type of CA and VA specimen are equivalent as concluded by Kensche (2002(a)).
- Temperature control to 54 °C as used in current certification is neglectable since sailplanes operate at different temperature zones during flight.
- The test frequency is 6 Hz which is the same as Van der Walt (2020).
- The amplitude ratio is $R = -0.55$ which is the same as Van der Walt (2020).

3.5.2 Static testing

Static testing for this study consists of both compressive and tensile testing that were mainly used to determine the material strength. The formula in Equation 6 was used to determine the material ultimate stress for both compressive and tensile loading of the specimen.

$$Q = F/A \quad (6)$$

Where:

Q is the Stress in kN/mm².

F is the failure force in kN.

A is the gauge cross-sectional area in mm².

3.5.3 Dynamic testing

Dynamic testing for this study entails three test envelope types. The test envelope types are classified as CA, VA, and MLS fatigue testing. The objectives and methodology of each of the test envelope types ensue.

3.5.3.1 Constant Amplitude Fatigue Testing

CA fatigue testing until failure occurred for three different percentage load values of the specimen's UTS was required as part of the MLS method. The objective of the CA fatigue tests is to determine the fatigue cycle failure count of a specimen subjected to constant fatigue load fluctuations. The applicable percentage load values were:

- 75 % of the UTS (Van der Walt, 2020).
- 65 % of the UTS.
- 50 % of the UTS (Similar to loads experience by sailplanes during flight).

The CA testing procedure used is illustrated in Figure 35.

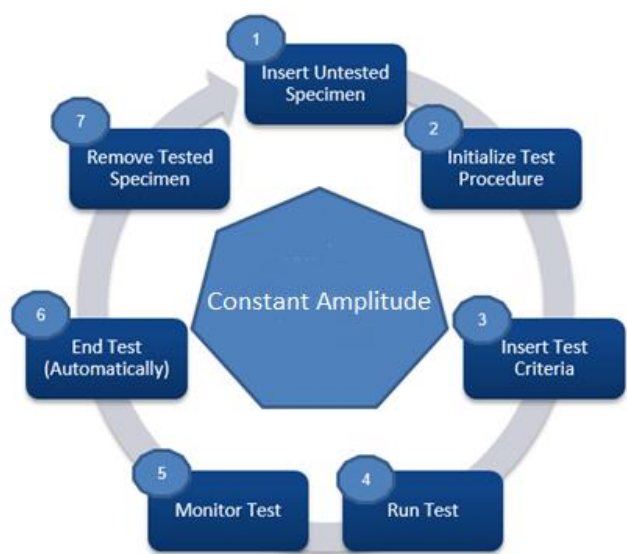


Figure 35: CA Testing Procedure

The test criteria for the CA fatigue tests are shown in Table 12.

Table 12: Constant & Variable Amplitude Test Input Criteria

Description	Symbol	Value	Unit
Average Ultimate Tensile Strength	UTS	12 (Determined through static testing)	kN
Percentage Load	Percentage Load	{75 / 65 / 50} (Depends on test load)	%
Frequency	f	6	Hz
Amplitude Ratio	R	-0.55	-

Cycle Count	N	Infinite (Until Failure)	-
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Please note that a minimum of three CA fatigue tests for each percentage load parameter was required.

3.5.3.2 Variable Amplitude Fatigue Testing

VA fatigue testing until failure occurred for three different percentage load values of the specimen's UTS was required as part of the MLS method. The objective of the VA fatigue tests is to determine the fatigue cycle failure count of a specimen subjected to variable fatigue load fluctuations. The applicable percentage load values were:

- 75 % of the UTS (Same as Van der Walt, 2020)
- 65 % of the UTS
- 50 % of the UTS (Similar to loads experience by sailplanes during flight)

The VA testing procedure used is illustrated in Figure 36.

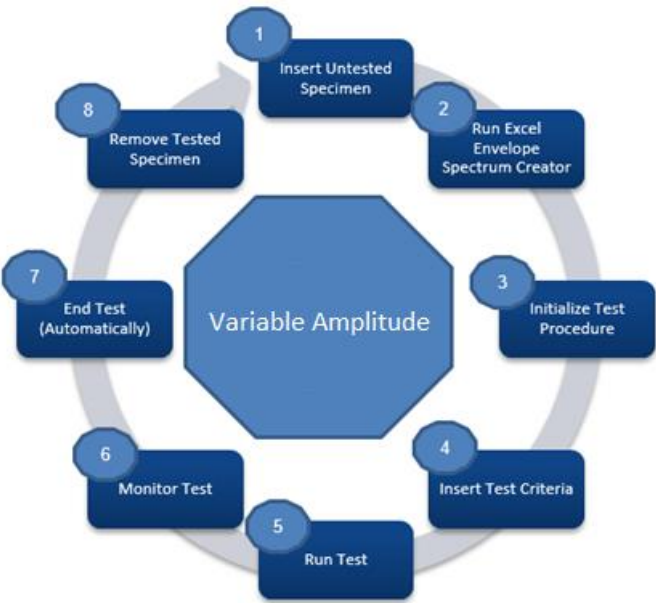


Figure 36: VA Testing Procedure

Please note that the VA and CA test input criteria are the same and that a minimum of three VA fatigue tests for each percentage load parameter was required.

3.5.3.3 Multiple Load Sequence Fatigue Testing

The MLS is a combination of CA and VA fatigue testing. The objective of MLS fatigue testing is to determine the equivalence between CA fatigue testing and VA fatigue testing performed on identical specimen.

The MLS fatigue testing started with VA fatigue testing on the X-axis of Figure 37. After reaching the predetermined transition point, the testing type changed to CA fatigue testing on the Y-axis of Figure 37, resulting in fatigue failure of the GFRP specimen.

The predetermined transition point on the VA fatigue X-axis of Figure 37 can be any percentage of the KosMOS II spectrum that allows for repeatability. The MLS methodology's VA transition points used to gather the residual constant amplitude cycles were in increments of 25% of the KoSMOS II spectrum.

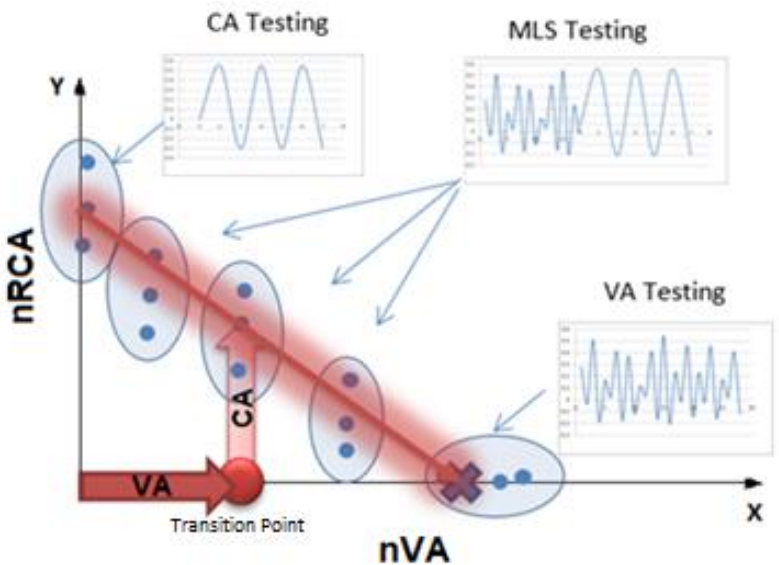


Figure 37: MLS Testing Sequence Illustration

MLS testing was performed for 3 different percentage load values of the UTS of the specimen until failure was reached.

The applicable percentage load values used are:

- 75 % of the UTS (Same as Van der Walt, 2020)
- 65 % of the UTS
- 50 % of the UTS Similar to loads experience by sailplanes during flight

The MLS testing procedure used is illustrated in Figure 38.

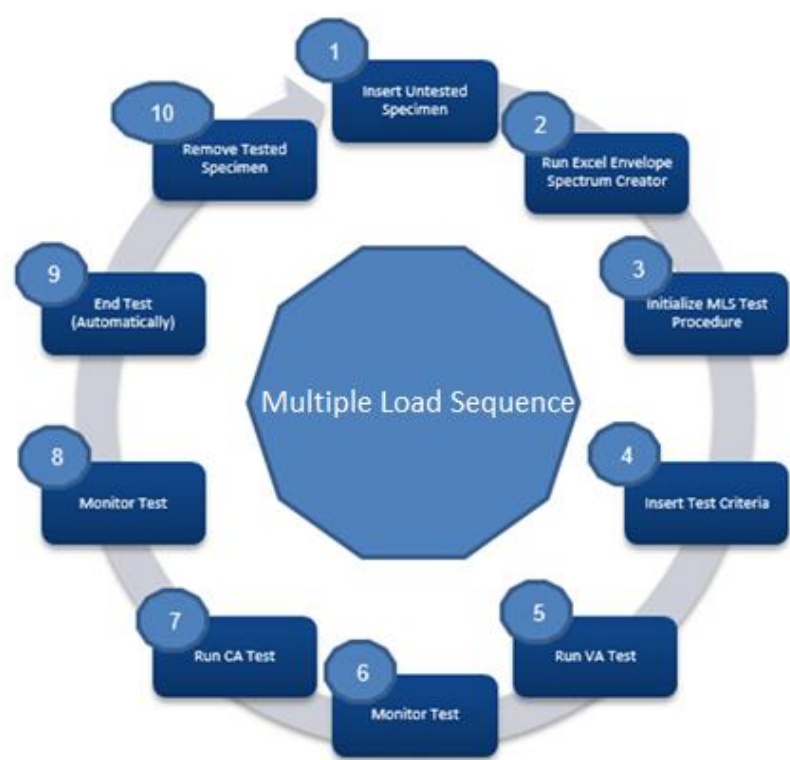


Figure 38: MLS Testing Procedure

In step 2 of both Figure 36 and Figure 38 a custom Excel Envelope Spectrum Creator (EESC) is used. The EESC extract can be seen in Appendix B. The EESC allows for an equivalent scaled distribution of the original KoSMOS II spectrum.

The EESC was used to create KoSMOS II representative spectrums for 0.25 times of the KoSMOS II spectrum to 2.5 times the KoSMOS II spectrum in increments of 25 %.

The input criteria for the MLS fatigue tests can be seen in Table 13. Note that the VA and CA input criteria are the same for the MLS fatigue tests with the only exception being the intermittent cycle count parameter.

Table 13: MLS Testing Input Criteria

Description	Symbol	Value	Unit
Ultimate Tensile Strength	UTS	12 (Determined through static testing)	kN
Percentage Load	Percentage Load	{75 / 65 / 50} (Depends on test load)	%
Frequency	f	6	Hz
Amplitude Ratio	R	-0.55	-
Cycle Count	N	0.25 *(KoSMOS II) 0.5 *(KoSMOS II) 0.75 *(KoSMOS II) 1.0*(KoSMOS II) 1.25 *(KoSMOS II) 1.5 *(KoSMOS II) 1.75 *(KoSMOS II) 2.0*(KoSMOS II) 2.25*(KoSMOS II) 2.5*(KoSMOS II)	-

3.5.4 Verification of the MLS methodology

The MLS methodology was established in 2020, by Van der Walt, who tested a specimen subjected to a fatigue loading of 75 % UTS. The result was a quadratic function that could predict the service life of a sailplane subjected to CA testing.

The verification of this dissertations' MLS test procedure requires the test of Van der Walt (2020) to be repeated on a few specimens subjected to 75 % UTS, which then can be normalised, compared and verified with a few specimens subjected to fatigue testing at 50 % UTS.

If the 75 % UTS specimen test result is a non-linear function, then the test method is verified to be equivalent to that of Van der Walt (2020). The reason why this dissertation's 75 % UTS fatigue results cannot be compared to that of the original MLS methodology is that Van der Walt (2020) used the KoSMOS II spectrum that was calculated for a completely different specimen category.

The MLS methodology of this dissertation was verified further by means of comparing 75 % UTS with that of both 65 % and 50 % UTS for the same specimen design.

3.6. Validation Methodology

The two recommended methods for validation of the MLS method are the Residual Strength and Residual Stiffness methods according to Van der Walt (2020). Both methods combine dynamic fatigue testing with static testing, with the only difference between the two methods being the measurable output parameters being force for the Residual Strength method and displacement for the Residual Stiffness method after a predetermined cycle count.

3.6.1 Residual Stiffness Method for Validation

The Residual Stiffness method is a non-destructive test that measures the permanent displacement between cycle counts for a specified load parameter without specimen failure. The applicable procedure used for the Residual Stiffness method can be seen in Figure 39.

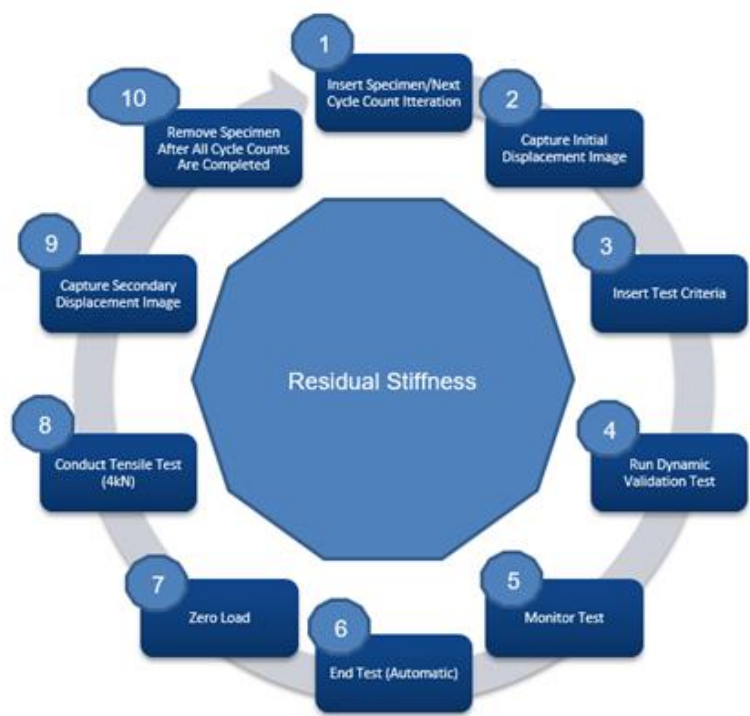


Figure 39: Residual Stiffness Test Procedure

The Residual Stiffness method is used to determine the remaining stiffness of a material, specimen or structure subjected to fatigue after a predetermined life span. An example of how the procedure was used is as follows.

Several specimens were tested with static tests to obtain the UTS and compressive strength averages. The average UTS value was used to calculate the percentage load values.

With the average UTS known, a specimen was tested in tension until a tensile force of 35 % of the UTS was reached to measure the displacement of the specimen that occurred at 35% of the UTS.

With the displacement of the specimen known at 35 % of the UTS, dynamic tests were possible and conducted. VA tests were conducted until a predetermined percentage of the fatigue life was reached. The frequency of the load spectrum for the VA tests where 6 Hz from peak to peak for the waveform of an amplitude ratio of -0.55.

The frequency and amplitude ratio are equivalent to Van der Walt (2020). When the predetermined fatigue life was reached with VA testing, a tensile test was performed with the tensile force being the same as the initial tensile test value of 35 % UTS. The new specimen displacement results were then measured and compared to the previous set of displacement results.

Residual Stiffness test were performed for the VA load spectrum until the desired fatigue life was reached. The Residual Stiffness test for the CA load spectrum was tested with the same methodology as the VA test. The only difference being the cycle type used.

After both VA and CA fatigue loading spectrums tested using the Residual Stiffness method, a comparison between the CA and VA loading spectrums were made and the results were compared to the MLS methodology's test results.

The initial input criteria for the Residual Stiffness tests can be seen in Table 14. The input parameters in Table 14 were inserted into the MTS Landmarks’ MPE Test Suite Residual Stiffness program.

Table 14: Residual Stiffness Test Input Criteria

Description	Symbol	Value	Unit
Ultimate Tensile Strength	UTS	12 (Determined through static testing)	kN
Percentage Load	Percentage Load	{75 / 65 / 50} (Depends on test load)	%
Frequency	f	6	Hz
Amplitude Ratio	R	-0.55	-
Cycle Count	n	27000 CA 30000 CA 33000 CA 36000 CA 39000 CA 278325.5 VA 556651 VA 834676.5 VA	-
Displacement Measurement Force	F _d	4	kN

3.6.2 Residual Strength Method for Validation

The Residual Strength method is a destructive test method that measures the remaining specimen strength after as specific cycle count set. (Sander, *et al.*, 2016). The applicable procedure used for the Residual Strength method can be seen in Figure 40.



Figure 40: Residual Strength Test Procedure

The Residual Strength method is used to determine the remaining tensile strength of a material, specimen and or structure subjected to fatigue after a predetermined life span. An example of how the procedure was used is as follows.

Several specimens were tested with static tests to obtain the UTS and compressive strength averages. The average UTS value was used to calculate the percentage load values. The load values for both the Residual Strength and Residual Stiffness methods are calculated in the same manner.

With the average UTS known, Residual Strength tests were possible and conducted. VA tests were conducted until a predetermined percentage of the fatigue life was reached. The frequency of the load spectrum for the VA tests where 6 Hz from peak to peak for the waveform of an amplitude ratio of -0.55.

When the predetermined percentage of the fatigue life was reached, a tensile test was performed until the specimen failed completely. The result of the tensile test was the Residual Strength of the specimen after a predetermined amount of VA cycles

The Residual Strength test with VA cycles was repeated on new specimen until the required data points were obtained. The Residual Strength test for the CA load spectrum was tested with the same methodology as the Residual Strength for the VA load spectrum test. The only difference being the cycle type used.

After both VA and CA fatigue loading spectrums were tested with the Residual Strength method a comparison between CA and VA loading spectrums were made and compared to the MLS methodology’s test results.

The initial input criteria for the Residual Strength tests can be seen in Table 15. The input parameters in Table 15 were inserted into the MTS Landmarks’ MPE Test Suite Residual Strength program.

Table 15: Residual Strength Test Input Criteria

Description	Symbol	Value	Unit
Ultimate Tensile Strength	UTS	12 (Determined through static testing)	kN
Percentage Load	Percentage Load	{75 / 65 / 50} (Depends on test load)	%
Frequency	f	6	Hz
Amplitude Ratio	R	-0.55	[-]
Cycle Count	n	27000 CA 30000 CA 33000 CA 278325.5 VA 556651 VA 834676.5 VA	-
Residual Force	F _d	Until Failure	kN

3.6.3 The Validation of the MLS methodology

The MLS method was validated by means of both the Residual Strength and Residual Stiffness methods. The validation of the MLS methodology was performed by using the same specimen type for the Residual Strength and Residual Stiffness methods. The MLS methodology's function at 50 % UTS was first compared to the residual stiffness method's function and later compared to the residual strength method's function. Both functions were a result of the procedures followed during testing.

The functions were normalised so that a comparison could be made between the different methodology's functions for the same specimen type. The reason normalisation was used is mainly due to the comparison of the fatigue life data at different percentage of the UTS.. Normalisation makes it possible to draw a conclusion from the data result with a slight increase in accuracy.

By comparing the data through normalization, the data equivalence and trends of the Residual Strength and Residual Stiffness methods could be compared to the MLS method. If the validation trends were over conservative, the validation methods would have predicted that failure occurred earlier compared to that of the MLS methodology's trend and the MLS methodology will be validated.

4. RESULTS AND DISCUSSION

4.1. Static Testing

4.1.1 Compression Testing

The compression tests were performed according to the ASTM D 3039 standard testing procedure, and the standard compression testing template of the MTS Landmark® 100 kN servo hydraulic test bench.

In Table 16 the specimen is safe in buckling for all cases with a minimum factor of 3. The specimen failure type that occurred can be defined as matrix rupture at a peak compressive load average of 10,45 kN, and a minimum peak stress of 259,66 MPa for all the specimens tested. The Euler buckling critical load minimum for the specimens tested were calculated to be 39,25 kN where the failure of the specific specimen in compression occurring at 10,84 kN.

The minimum peak stress noted for the compression tests were higher than the proposed GFRP stress of 250 MPa as defined by sailplane legislation which validates the material compressive strength.

Table 16: Compression Test Results

Specimen	Gauge Length mm	Gauge Width mm	Gauge Thickness mm	Cross-Section mm ²	P_crit {Euler} kN	Peak Load kN	Peak Stress MPa
1	16,20	18,92	1,83	34,62	39,25	10,84	313,03
2	15,31	19,84	2,01	39,88	61,06	12,44	311,90
3	16,57	18,52	1,93	35,74	43,07	10,32	288,75
4	16,43	18,45	1,93	35,61	43,65	9,43	264,69
5	15,57	18,33	1,94	35,56	49,04	9,23	259,66
Average	16,02	18,81	1,93	36,28	47,21	10,45	287,61

the compression results can be seen in Figure 41.

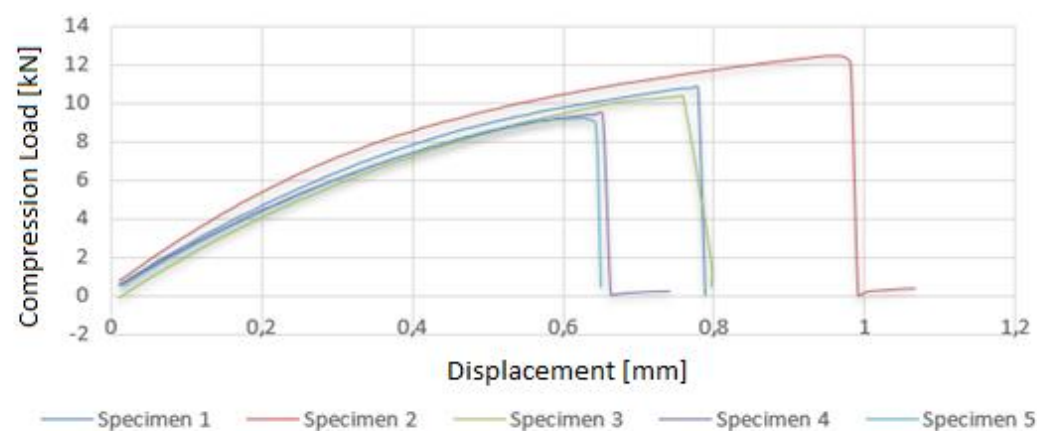


Figure 41: Compression Test Result Graph

4.1.2 Tensile Testing

The tensile tests were performed according to the ASTM D 3039 standard testing procedure and the standard tensile testing template of the MTS Landmark® 100 kN servo hydraulic test bench.

In Table 17 the specimen tested failed at an average tensile peak load of 11,93 kN which was higher than the theoretical failure load of 9 kN. The specimen failure type that occurred can be defined as fibre rupture at the peak tensile load.

Table 17: Tensile Test Results

Specimen	Gauge Length mm	Gauge Width mm	Gauge Thickness mm	Cross-Section mm²	Theoretical Load kN	Peak Load kN	Peak Stress MPa
1	16,40	18,43	1,95	35,94	8,98	12,29	341,95
2	15,48	18,82	1,92	36,13	9,03	12,77	353,31
3	15,89	18,78	1,91	35,87	8,97	12,45	347,07
4	15,66	18,30	1,92	35,14	8,78	10,88	309,64
5	15,61	18,75	1,99	37,31	9,33	11,28	302,33
Average	15,81	18,62	1,94	36,08	9,02	11,93	330,86

The minimum peak stress noted for the tensile tests were higher than the proposed GFRP stress of 250 MPa, as defined by sailplane legislation, which validates the material tensile strength.

The tensile results can be seen in Figure 42.

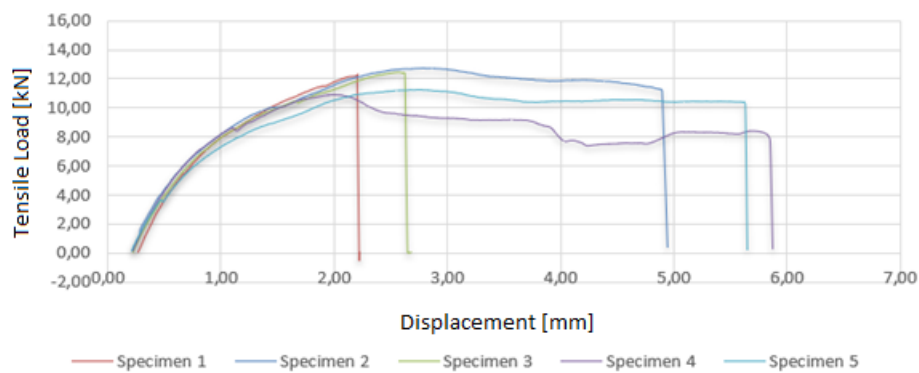


Figure 42: Tensile Test Results Graph

Note that the load peaks for three of tested specimen did not result in sudden failure. The peak load failure can be seen as the matrix failing, while the fibres are still capable of carrying the load until sudden failure occurs, which clearly proves that multiple failure types can occur on a specimen. The differences in the orientation of fibres from specimen to specimen could in theory result in a significant difference in the results.

Attention to the fibre positioning was however given to minimize the impact on the end results. The variation in tensile properties can result in large data scatter for the subsequent fatigue testing.

4.2. Constant Amplitude Test

The CA test results can be seen in Table 18. The data in Table 18 was grouped into three load categories. The three different load categories are 50 %, 65 % and 75 % of the UTS.

The statistical outliers were calculated with the statistical Inter-Quartile Range (IQR) technique and the Excel QUARTILE function was used in to calculate the quartiles of each load category separately.

Table 18: Constant Amplitude Test Results

Load	Type	Specimen	Cycles	Fatigue Failure
50	CA	5	56925	Yes
50	CA	4	54554	Yes
50	CA	1	46931	Yes
50	CA	2	45621	Yes
50	CA	3	45568	Yes
50	CA	7	42033	Yes
50	CA	6	29028	Yes
65	CA	6	11489	Yes
65	CA	5	8250	Yes
65	CA	1	6394	Yes
65	CA	4	6226	Yes
65	CA	3	3077	Yes
65	CA	2	2929	Yes
75	CA	2	2360	Yes
75	CA	4	1993	Yes
75	CA	1	1186	Yes
75	CA	3	1115	Yes
75	CA	6	935	Yes
75	CA	5	399	Yes
75	CA	7	315	Yes
100	Static	Test	0.5	No

The CA test results were plotted and can be seen in Figure 43.

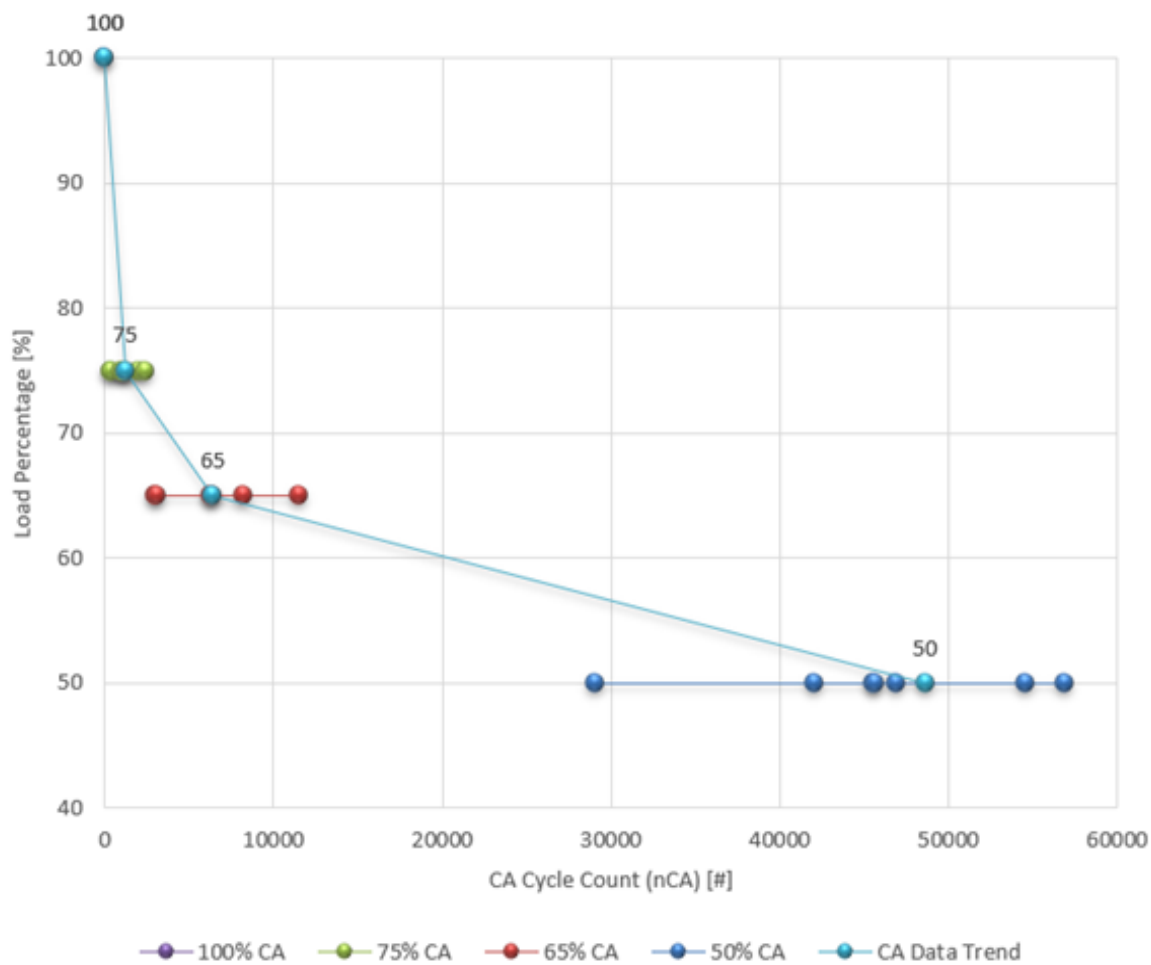


Figure 43: CA Test Results

A large cycle count data scatter band for 50 % loading, compared to 75 % of the UTS can be seen in Figure 43. This large scatter band may be problematic in the future, due to great difficulty arising when trying to predict the safe life of a sailplane for service life extension certification by means of CA testing.

4.3. Variable Amplitude Test

The VA test results can be seen in Table 19. The data in Table 19 was grouped into the same three load categories as used by the CA testing. The three different load categories are 50 %, 65 % and 75 % of the UTS.

Table 19: Variable Amplitude Test Results

Load	Type	Specimen	Cycles	Fatigue Failure
50	VA	8	3322562	Yes
50	VA	10	3314401	Yes
50	VA	11	3204284	Yes
50	VA	12	3126907	Yes
50	VA	9	3113102	Yes
65	VA	8	1113302	Yes
65	VA	7	1056651	Yes
65	VA	9	1046135	Yes
65	VA	11	1025970	Yes
65	VA	10	1024414	Yes
75	VA	11	431941	Yes
75	VA	10	429326	Yes
75	VA	9	372667	Yes
75	VA	12	337514	Yes
75	VA	8	326041	Yes
100	Static	Test	0.5	No

The VA test results were plotted and can be seen in Figure 44.

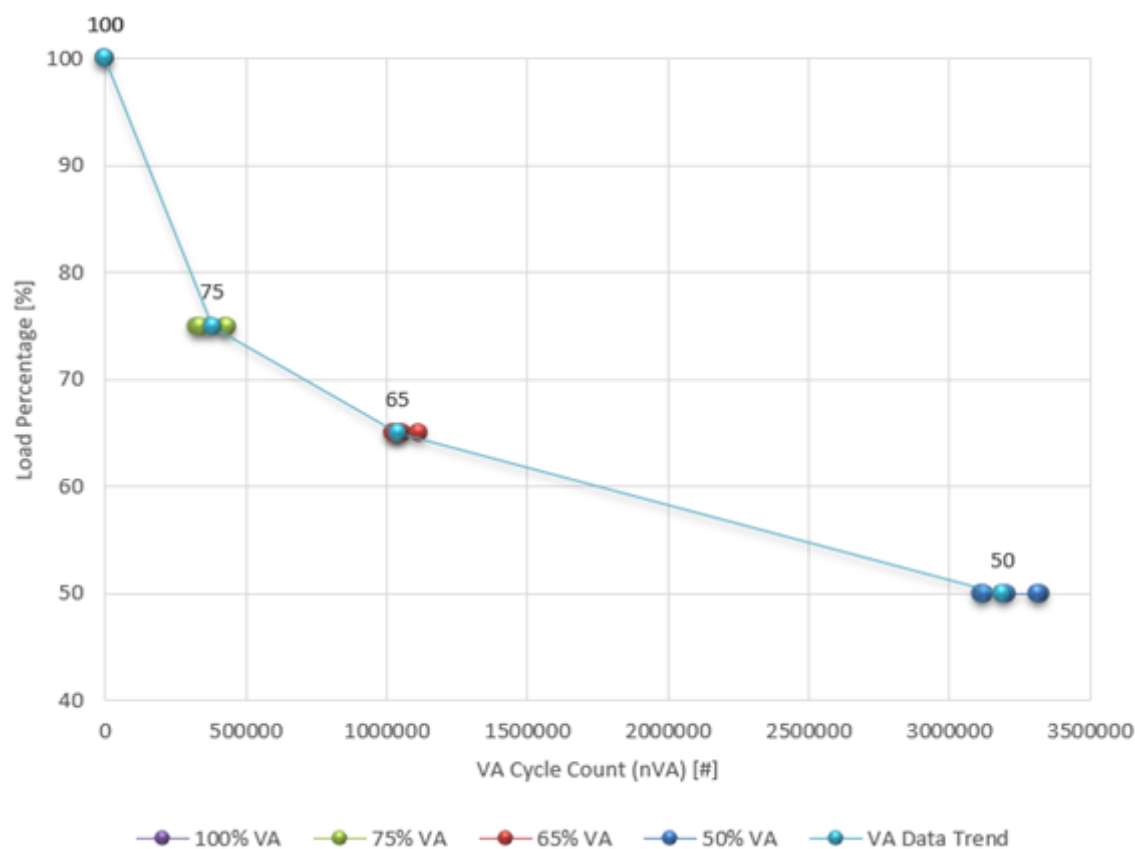


Figure 44: VA Test Results

The VA test methodology yielded an average failure count of 3189674 VA cycles at 50 % UTS in Figure 44 which equals 5,7 KoSMOS II cycles through interpolation. Since sailplane certification legislation requires a safety factor of at least 3 the coupon type specimen may in theory be certified for up to 22000 Fh since 12000 Fh equals 3 times the entire KoSMOS II load spectrum.

4.4. MLS Test

The MLS test results for the 50 % UTS loads can be seen in Table 20. The data contains the combined number of cycles achieved before failure occurred, at 50 % of the UTS, which is similar to the safety factors used by sailplane legislation.

Table 20: MLS Test Results at 50 % of the UTS

Load %	Type	Specimen	nVA	nRCA
50	CA	CA50	0	48605
50	MLS50	13	278326	47788
50	MLS50	14	556651	46931
50	MLS50	15	834977	46195
50	MLS50	16	1113302	39199
50	MLS50	17	1391628	34413
50	MLS50	18	1669954	24935
50	MLS50	19	1948280	17194
50	MLS50	20	2226606	15159
50	MLS50	21	2504932	6427
50	MLS50	22	2783258	1514
50	VA	VA50	3189674	0

The data from Table 20 can be seen in Figure 45. The data trend for the MLS at 50 % UTS was determined through finite differences to be represented by a 4th order polynomial function.

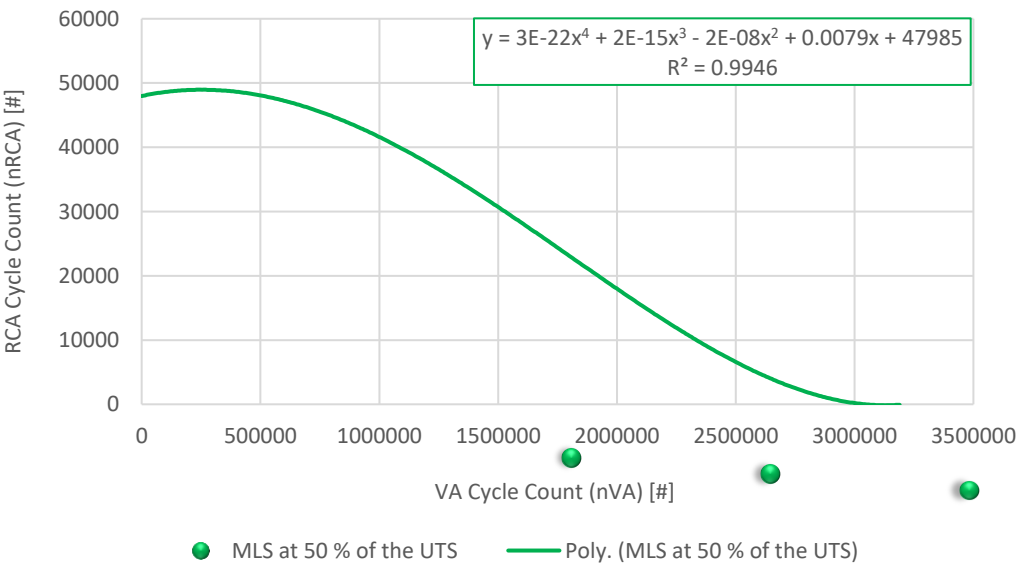


Figure 45: MLS Test Results at 50 % of the UTS

The MLS test results for the 65 % UTS loads can be seen in Table 21. The data contains the combined number of cycles achieved before failure occurred at 65 % of the UTS.

Table 21: MLS Test Results at 65 % UTS

Load	Type	Specimen	nVA	nRCA
65	CA	CA65	0	6394
65	MLS65	12	111331	6081
65	MLS65	13	278326	5598
65	MLS65	14	333990	4888
65	MLS65	15	389656	4415
65	MLS65	16	445321	3578
65	MLS65	17	500986	2768
65	MLS65	18	556651	2393
65	MLS65	19	612316	1853
65	MLS65	20	667981	1231
65	MLS65	21	723646	783
65	MLS65	22	779311	717
65	MLS65	23	834977	489
65	VA	VA65	1038293	0

The data from Table 21 can be seen in Figure 46. The data trend for the MLS at 65 % UTS was determined through finite differences to be represented by a 4th order polynomial function.

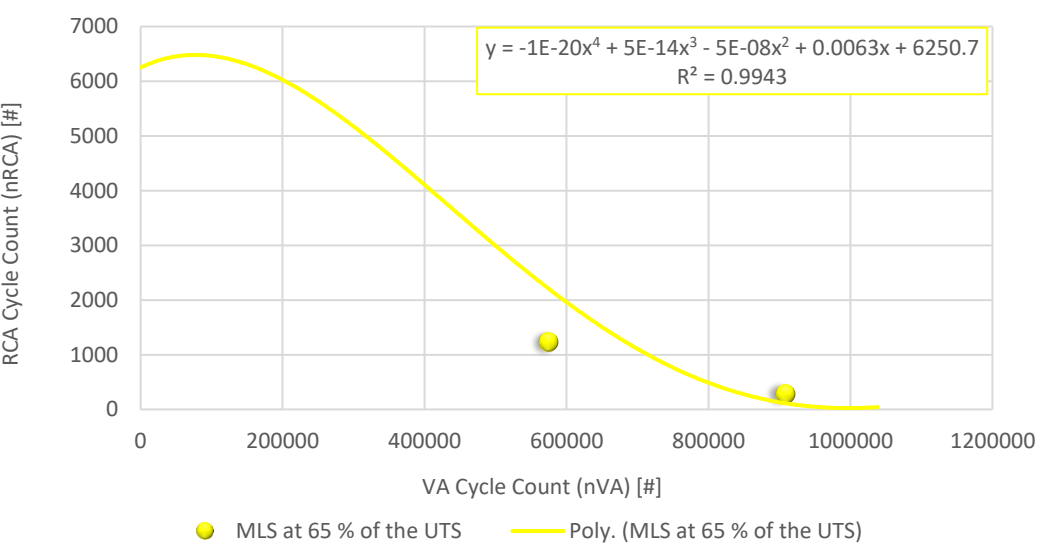


Figure 46: MLS Test Results at 65 % UTS

The MLS test results for the 75 % UTS can be seen in Table 22. The data contains the combined number of cycles achieved before failure occurred at 75 % of the UTS, which is equivalent to the UTS percentage used by Van der Walt (2020).

Table 22: MLS Test Results at 75 % UTS

Load	Type	Specimen	nVA	nRCA
75	CA	CA75	0	1186
75	MLS 75	13	111331	1088
75	MLS 75	14	139162	1032
75	MLS 75	15	166995	924
75	MLS 75	16	194827	844
75	MLS 75	17	212660	742
75	MLS 75	18	222673	604
75	MLS 75	19	250493	457
75	MLS 75	20	278325	248
75	MLS 75	21	306158	153
75	MLS 75	22	333991	125
75	MLS 75	23	353875	27
75	VA	VA75	379498	0

The data from Table 22 can be seen in Figure 47. The data trend for the MLS at 75 % UTS was determined through finite differences to be represented by a 4th order polynomial function.

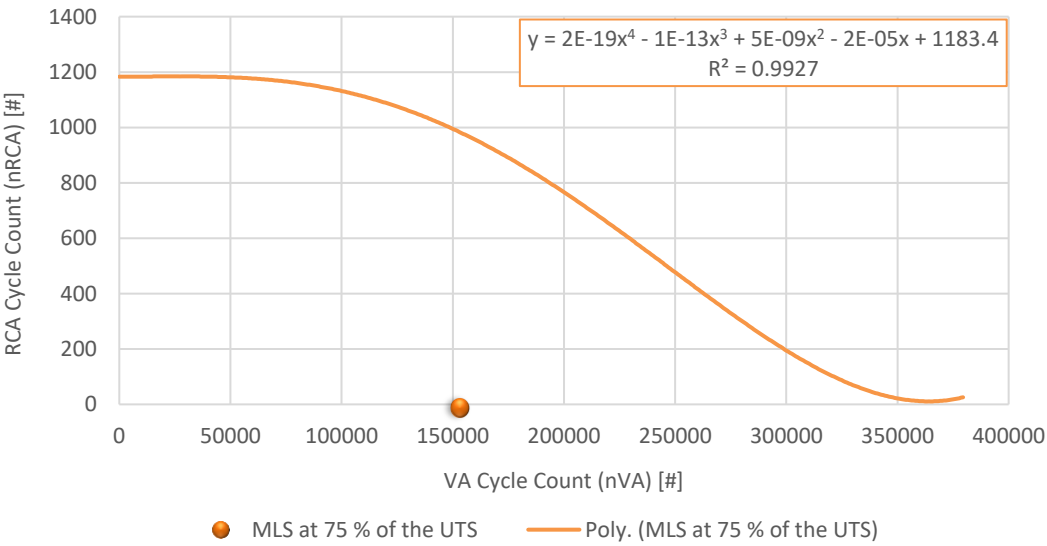


Figure 47: MLS Test Results at 75 %UTS

The different MLS test results for the different UTS loads are illustrated in Figure 48 to give insight on the data trend at different load levels.

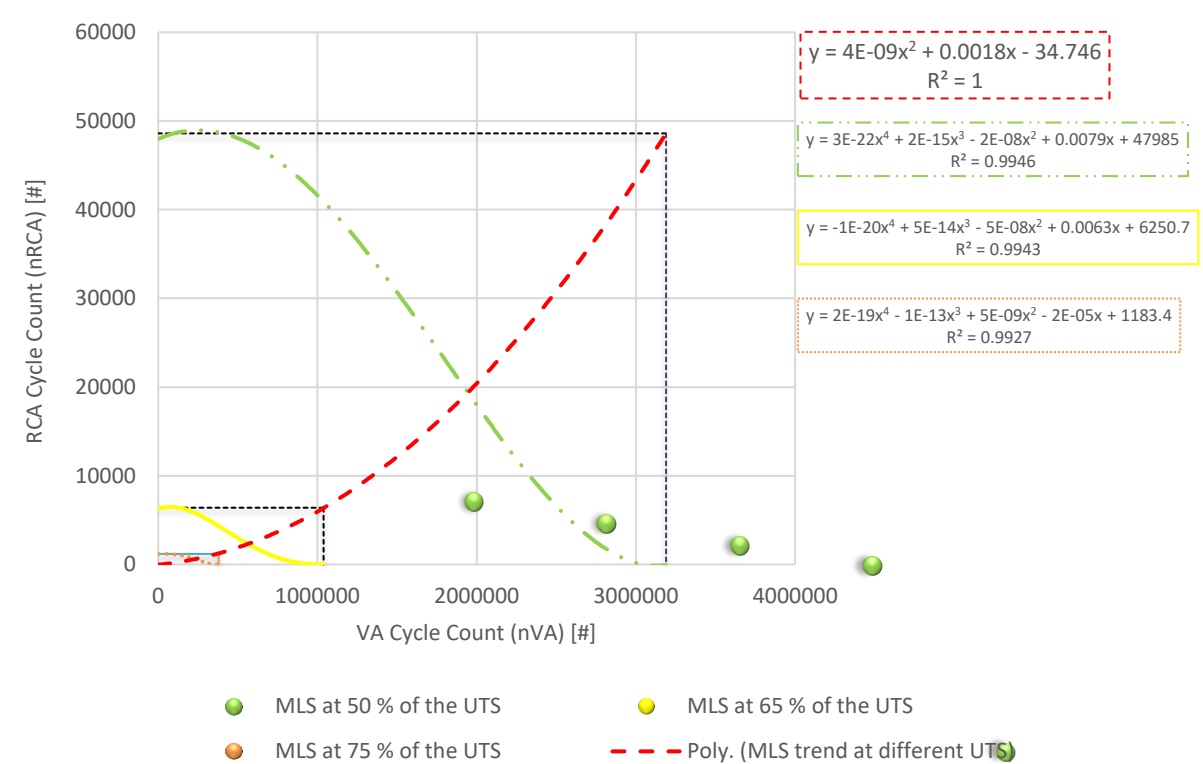


Figure 48: Combined Cycles to Failure Results

A correlation between the UTS load percentages in Figure 48 can be determined by normalising the data sets towards a numerical value range of between 0 and 1. The Combined Cycles to Failure graph in Figure 48 illustrates that the difference in cycle failure on both CA as well as VA axes for each of the load categories are non-linear and that the failure rate increases as the UTS increases.

The normalisation of the MLS test results can be seen in Figure 49.

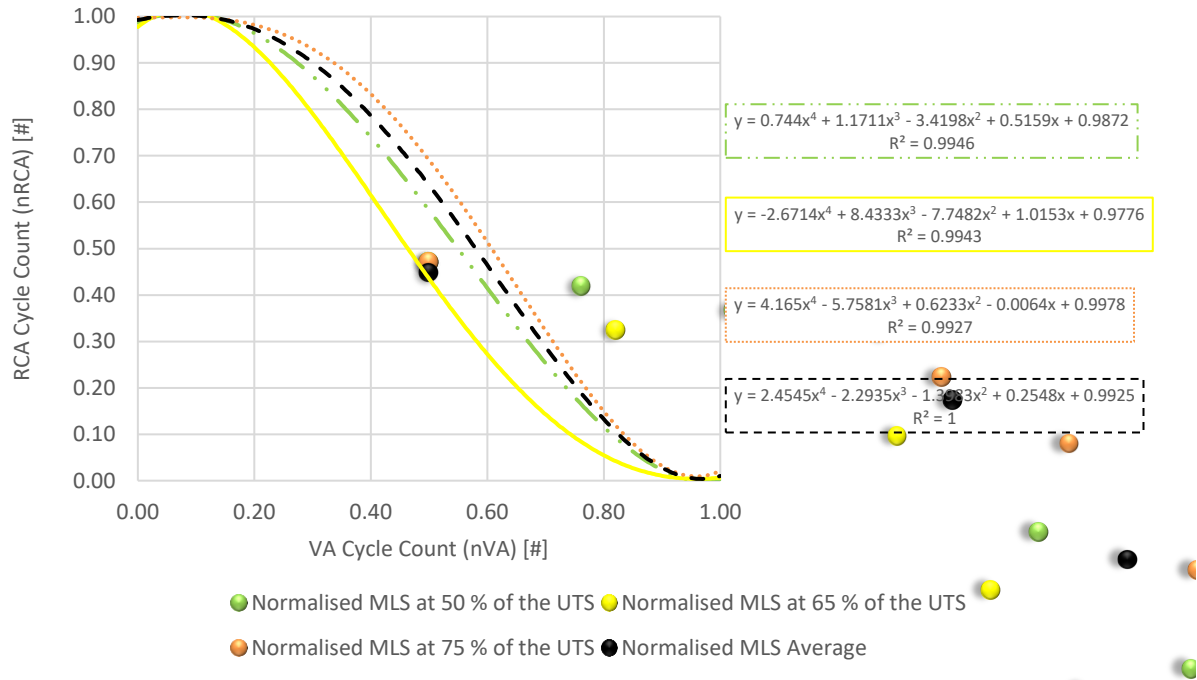


Figure 49: Normalised Combined Cycles to Failure Results

In Figure 49 the data equivalence between the different load cases can be seen where the black dash 4th degree polynomial curve with a regression (R^2) of unity, represents the average of the three datasets and forms the normalised Combine Cycle to Failure equation illustrated in Equation 7.

$$\left(\frac{nRCA}{nCA_{max}}\right) = 0,98753 + 0,50827 * \left(\frac{nVA}{nVA_{max}}\right) - 3,5149 * \left(\frac{nVA}{nVA_{max}}\right)^2 + 1,2821 * \left(\frac{nVA}{nVA_{max}}\right)^3 + 0,74587 * \left(\frac{nVA}{nVA_{max}}\right)^4 \quad (7)$$

Where:

$\left(\frac{nRCA}{nCA_{max}}\right)$ is the normalised residual CA cycle count between zero and unity.

$\left(\frac{nVA}{nVA_{max}}\right)$ is the normalised VA cycle count between zero and unity.

The required residual CA cycle count for flight applicable loads for the MLS method at 50 % UTS can be determined by Equation 8 which is derived from Equation 7 by substituting both nCA_{max} and nVA_{max} with the cycle count where failure occurs during the respective fatigue tests at a UTS of 50 %.

$$(nRCA) = 47999 + 0,007745 * (nVA) - 1,68 * 10^{-8} * (nVA)^2 + 1,92 * 10^{-15} * (nVA)^3 + 3,50 * 10^{-22} * (nVA)^4 \tag{8}$$

The nRCA data obtained from Equation 8 can be seen in the theoretical nRCA column of Table 23 and in Figure 50. The average cycle error was determined to be 308 RCA cycles.

Table 23: MLS CCF Equation Comparison

nVA	Empirical nRCA	Theoretical nRCA	Difference
0	48605	47999	606
278326	47788	48897	-1109
556651	46931	47472	-541
834977	46195	44047	2148
1113302	39199	38997	202
1391628	34413	32746	1667
1669954	24935	25771	-836
1948280	17194	18597	-1403
2226606	15159	11800	3359
2504932	6427	6007	420
2783258	1514	1896	-382
3189674	0	431	-431
Average Cycle Error			308

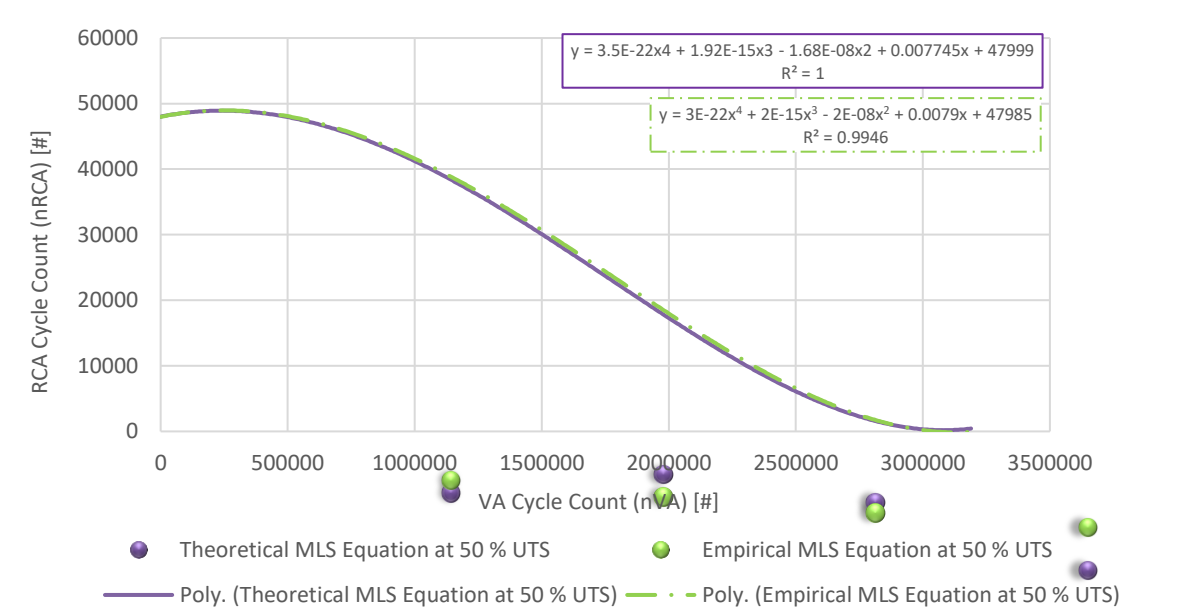


Figure 50: MLS Empirical vs Theoretical Equation Comparison

The data regression of Equation 8 is non-linear in Figure 51. The data regression (R^2) equals 0,9637 for a linear relationship compared to a regression value (R^2) equalling unity for a 4th degree polynomial function. The linearity analysis implies that the relationship between CA and VA cycle counts to failure is not linear. The Palmgren-Miner damage accumulation rule also does not apply when calculating the amount of VA fatigue cycles from CA fatigue cycles through extrapolation.

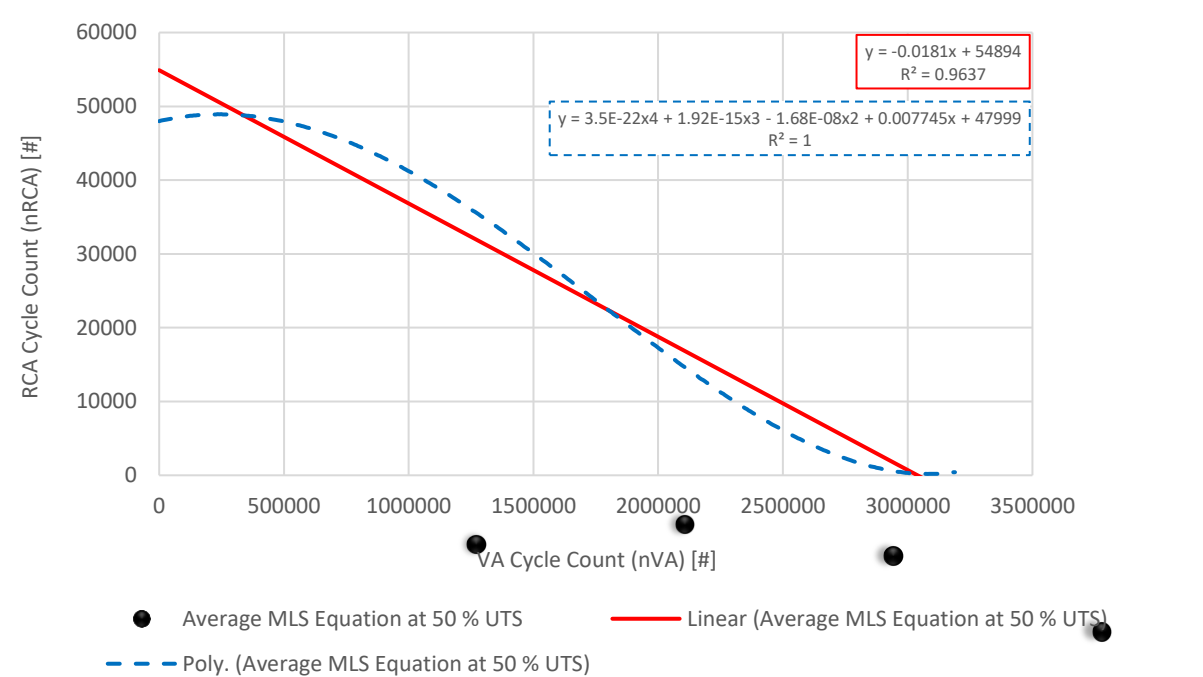


Figure 51: MLS Data Regression

The MLS polynomial function illustrated in Figure 51 and Equation 8 converted to the CA-VA equivalency diagram can be seen in Figure 52 for a load level applicable to sailplanes. The data conversion was done by rearranging the theoretical nRCA column of Table 23 to form theoretical nCA data points that can be paired with the nVA column of Table 23.

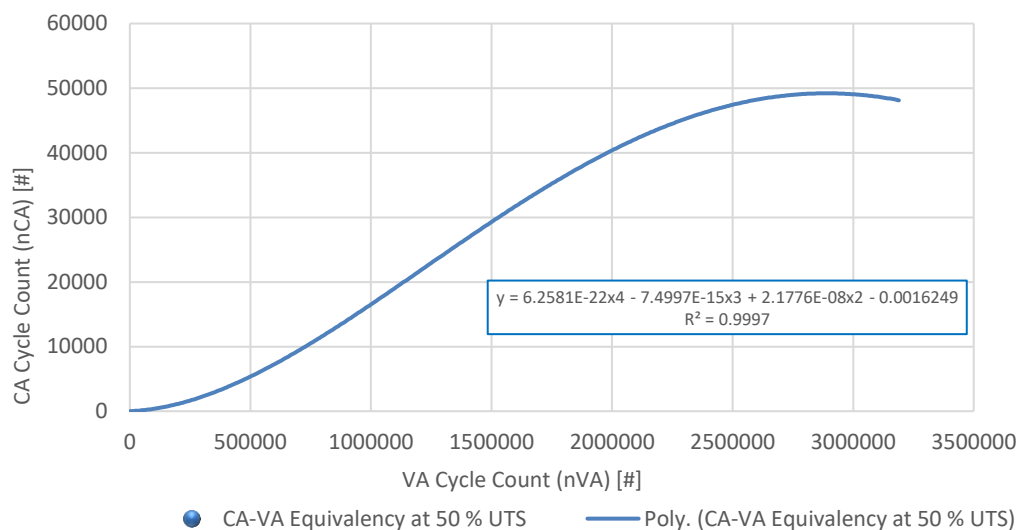


Figure 52: CA-VA Equivalency at 50 % UTS

The CA-VA equivalency formula of Figure 52 is written for the approximate amount of CA cycles from the amount of VA cycles in the form of Equation 9.

$$(nCA) = 0,0016249 * (nVA) + 2,1776 * 10^{-8} * (nVA)^2 - 7,4997 * 10^{-15} * (nVA)^3 + 6,2581 * 10^{-22} * (nVA)^4 \tag{9}$$

Data from Equation 9 can be seen in Table 24. It is noted from Table 24 that the maximum certifiable flight hours for the MLS coupon type specimen are 22800 hours with the specimen failing after 68400 actual hours of flight due to a safe life factor of 3.

Table 24: Certifiable Flight Hours Correlation

Sailplane Certifiable Flight Hours	nVA	Approximate Equivalent nCA
0	0	0
2000	278326	1981,195
4000	556651	6418,528
6000	834977	12477,04
8000	1113302	19411,83
10000	1391628	26568,21
12000	1669954	33381,55
14000	1948280	39377,38
16000	2226606	44171,35
18000	2504932	47469,25
20000	2783258	49066,99
22800	3189674	48131,57

4.5. Validation

4.5.1 Residual Strength Method

The CA Residual Strenght test results for a 50 % of the UTS can be seen in Table 25. It can be noted from Table 25 that the tested specimen failed at around 10,3 kN at 35000 CA cycles and around 9,4 kN at 43000 CA cycles.

Table 25: CA Residual Strength Results at 50 % UTS

nCA	Sample	Gauge Width mm	Gauge Length mm	Thickness mm	Tensile Load kN
35000	23	17,92	15,58	1,7	10,199
35000	24	17,98	15,7	1,8	10,285
35000	25	18,09	15,57	1,88	10,829
39000	26	17,88	15,66	1,76	9,562
39000	27	18	15,5	1,78	10,288
39000	28	17,92	16,33	1,78	9,685
43000	29	18	15,84	1,78	9,01
43000	30	17,9	16	1,86	8,696
43000	31	18	15,8	2,05	9,594
43000	32	17,66	15,92	1,74	9,515

The CA Residual Strength data trend of Table 25 can be seen in Figure 53 to be linear from 35000 CA cycles to 43000 CA cycles. It can however be noted from the graph that the tensile strength drastically reduces in value at around 43000 CA cycles before failure occurs at 48605 CA cycles. The tensile load average for 0 nCA cycles were obtained from the tensile test results at 50 % UTS and the maximum nCA cycle count average was determined from the CA fatigue tests at 50 % UTS.

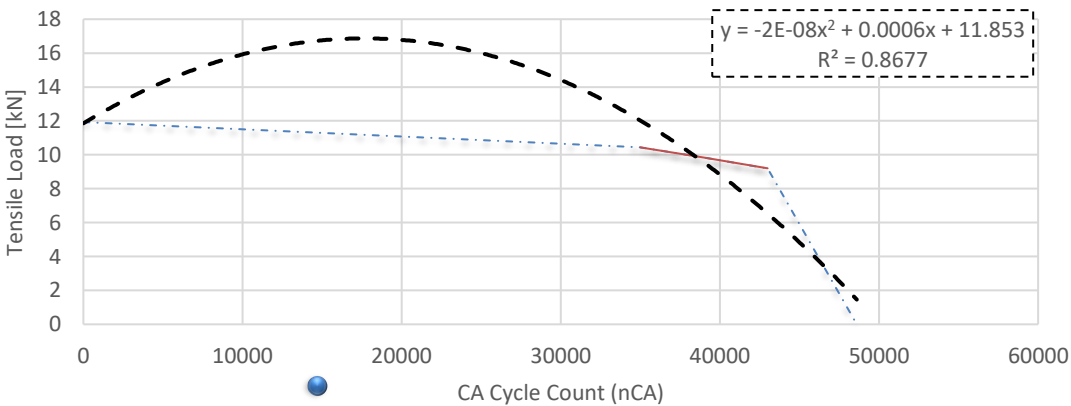


Figure 53: CA Residual Strength Test Results at 50 % UTS

The VA Residual Strenght test results at 50 % UTS can be seen in Table 26. It can be noted from Table 26 that the tested specimen failed at around10,7 kN at 278326 VA cycles and around9,3 kN at 834977 VA cycles. The tensile load average for 0 nVA cycles were obtained from the tensile test results at 50 % UTS and the maximum nVA cycle count average was determined from the VA fatigue tests at 50 % UTS.

Table 26: VA Residual Strength Results at 50 % UTS

nVA	Sample	Gauge Width mm	Gauge Length mm	Thickness mm	Tensile Load kN
278326	33	17,9	15,72	1,76	10,799
278326	34	18	15,72	1,74	10,972
278326	35	17,72	15,7	1,74	10,262
556651	36	17,88	15,66	1,74	10,176
834977	37	17,98	15,9	1,76	9,238
834977	38	18	14,5	1,74	9,416

It can be noted from Figure 54 that most of the data points are in the first quadrant which does not help the data prediction accuracy.

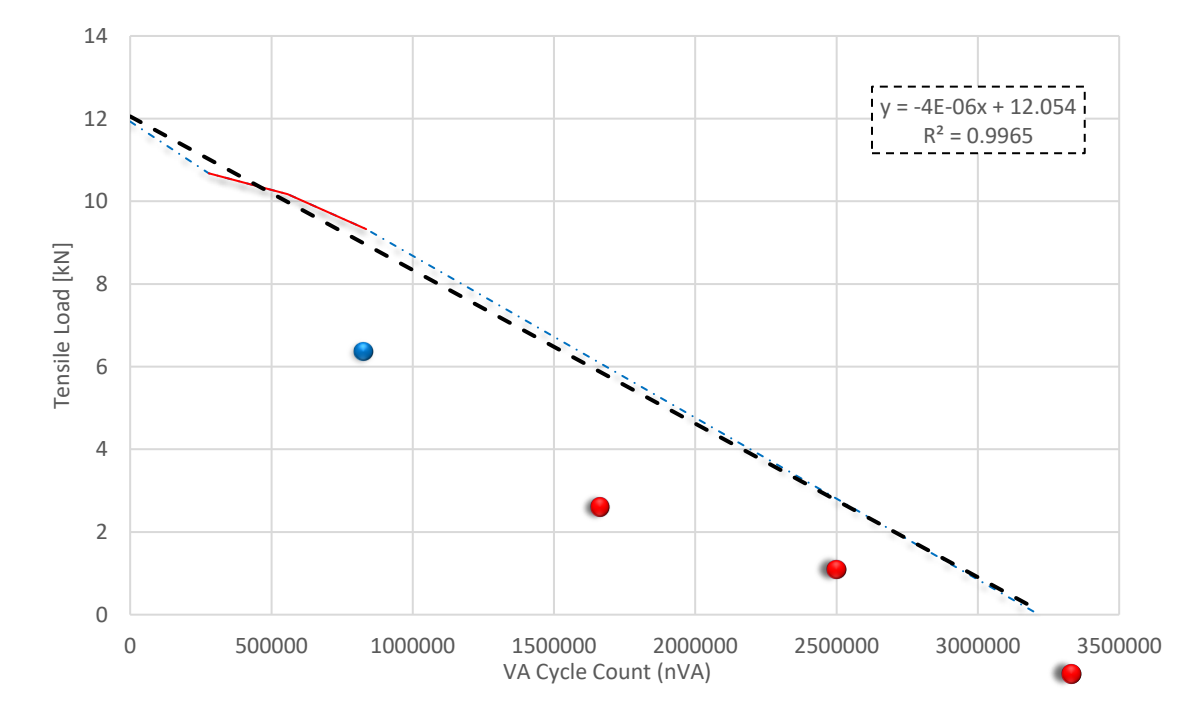


Figure 54: VA Residual Strength Test Results at 50 % UTS

The correlation between the Residual Strength for CA and VA from Figure 53 and Figure 54 can be determined through normalization of the different datasets to values between zero and unity. The correlation between CA and VA cycles at 50 % of the UTS for the Residual Stiffness method can be seen in Figure 55.

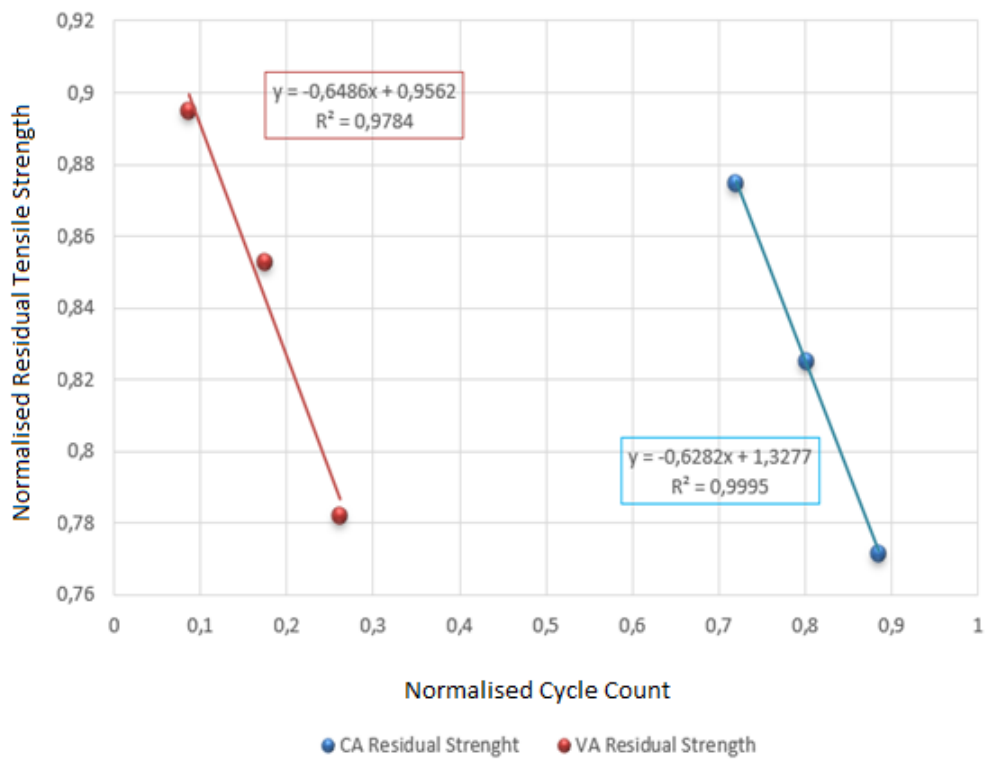


Figure 55: CA-VA Residual Strength Comparison

In Figure 55 the datasets are wide apart with the regression slope having almost the same value, which suggest that the damage of the VA cycle tests occurred at the beginning of the VA Residual Strength test sequence. A parallel correlation for the Residual Strength method between CA and VA cycles at 50 % of the UTS can thus be assumed.

The validation and verification of the MLS methodology was performed by plotting the Residual Strength and MLS results at 50 % UTS on the same graph in Figure 56.

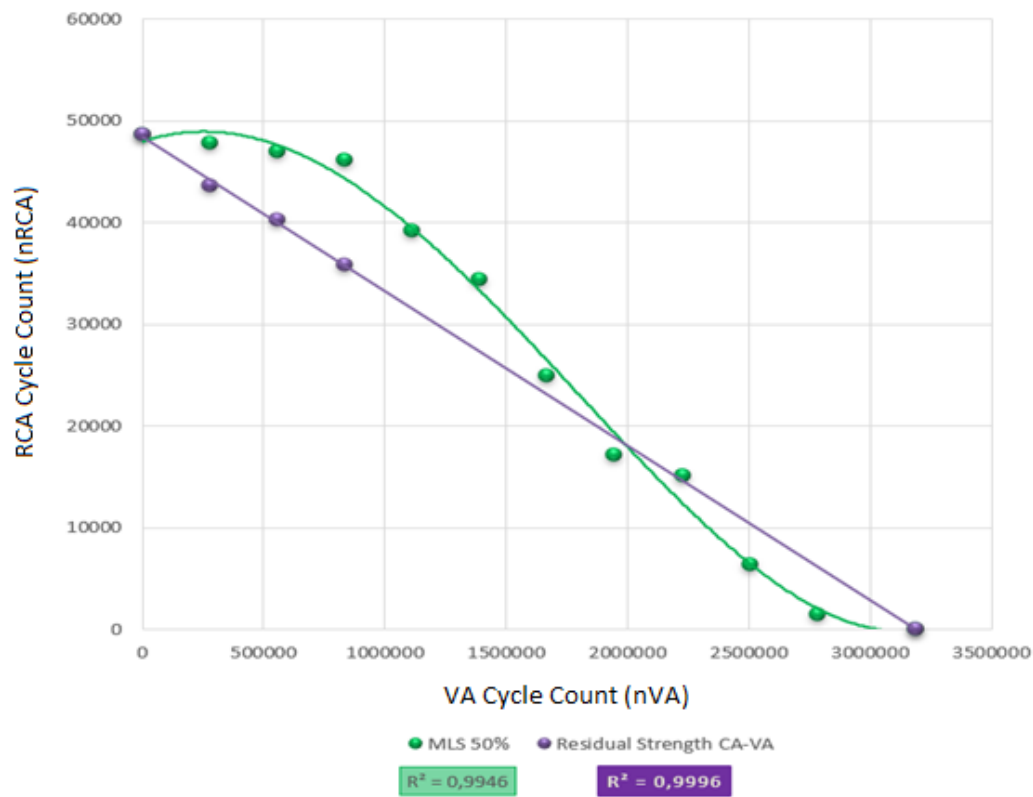


Figure 56: Residual Strength - MLS Comparison at 50 % UTS

The Residual Strength slope in Figure 56 is linear from the RCA axis towards the VA axis, which shows that the damage state of the specimen tested with the different loads cycle types are equivalent as demonstrated by Kensche (2002(a)). The empirical MLS method, however, results in a 4th degree polynomial function which is more accurate due to the multiple data points tested and fewer assumptions made compared to the Residual Strength method.

4.5.2 Residual Stiffness Method

The residual stiffness test results for a 50 [%] UTS CA load can be seen in Table 27. It can be noted from Table 27 that the tested specimen elongated 0,056 [mm] after 27000 CA cycles and 0,199 [mm] after 39000 CA cycles. The displacement increments between the nCA cycles reduces exponentially with the increase of nCA cycles which could eventually indicate the fatigue limit through further testing.

Table 27: CA Residual Stiffness Results at 50 % UTS

nCA	Sample	Axial Load Applied kN	Displacement mm
27000	39	4	0,056
30000	40	4	0,138
33000	41	4	0,162
36000	42	4	0,182
39000	43	4	0,199

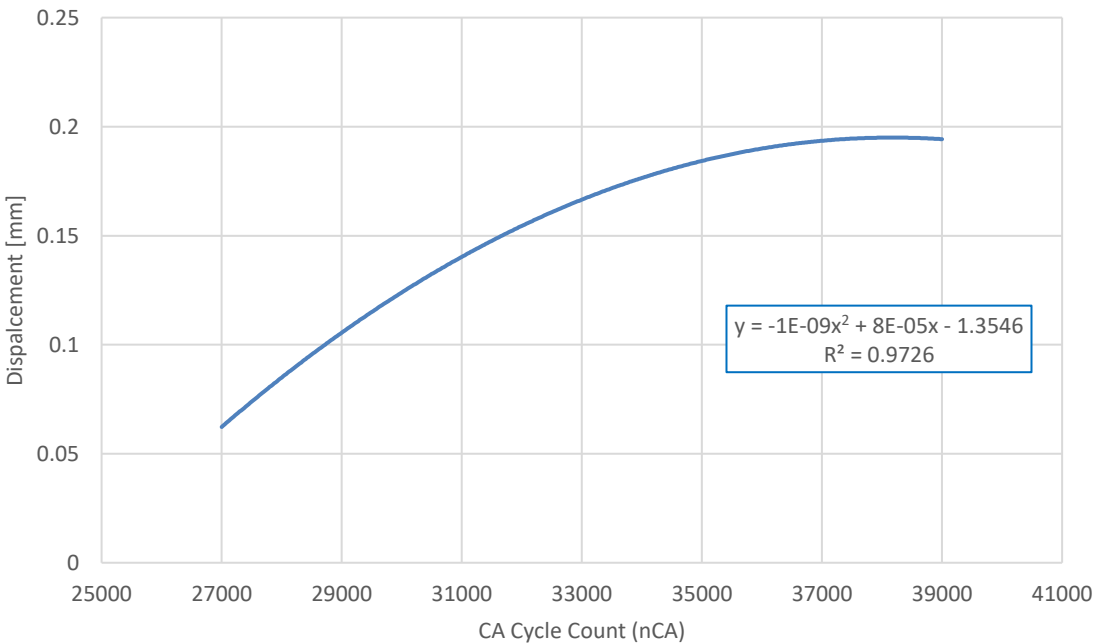


Figure 57: CA Residual Stiffness Test Results at 50 % UTS

The Residual Stiffness data trend of Table 27 can be seen in Figure 57 to be quadratic from 27000 CA cycles to 39000 CA cycles with specimen displacement reaching a maximum of 0,199 mm at 39000 CA cycles.

The VA Residual Stiffness test results for a UTS of 50 % can be seen in Table 28. It can be noted from Table 28 that the tested specimen elongated 0,132 mm after 278326 VA cycles and 0,202 mm after 834976,5 VA cycles.

Table 28: Residual Stiffness 50 [%] UTS VA Test Results

nVA	Sample	Axial Load Applied kN	Displacement mm
278326	44	4	0,132
556651	45	4	0,156
834976,5	46	4	0,202

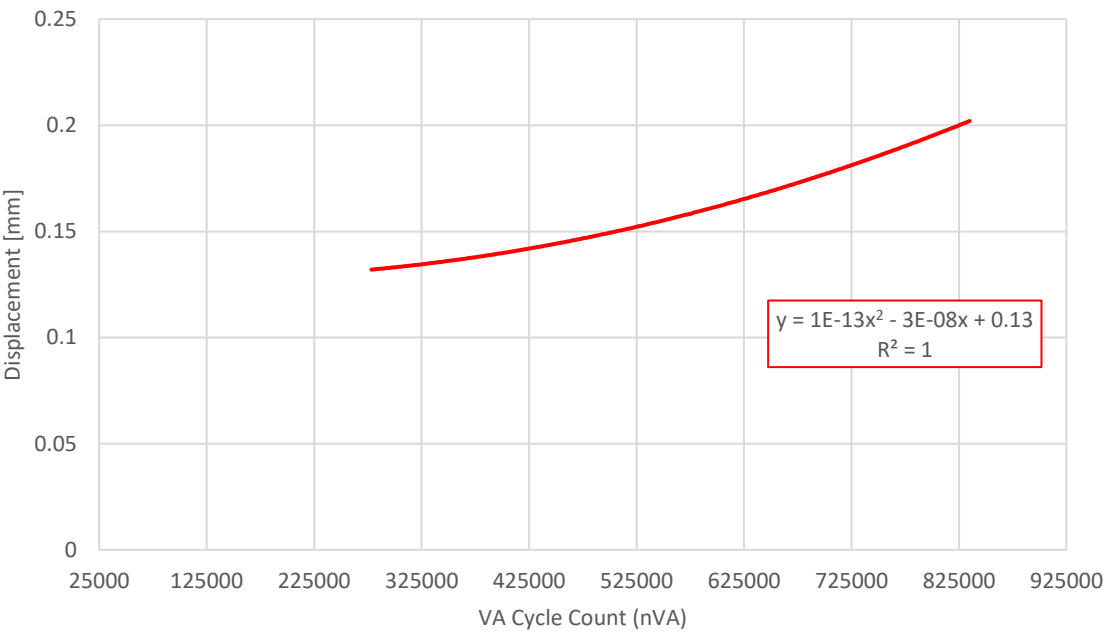


Figure 58: Residual Stiffness VA Test Results

The Residual Stiffness data trend of Table 28 can be seen in Figure 58 to be exponential from 278326 VA cycles to 834976 VA cycles with specimen displacement reaching a maximum of 0,201 mm at 834976 VA cycles.

The correlation between the Residual Stiffness for CA and VA from Figure 57 and Figure 58 can be determined through normalization of the different datasets to values between zero and unity. The correlation between CA and VA cycles at 50 %UTS for the Residual Stiffness method can be seen in Figure 59.

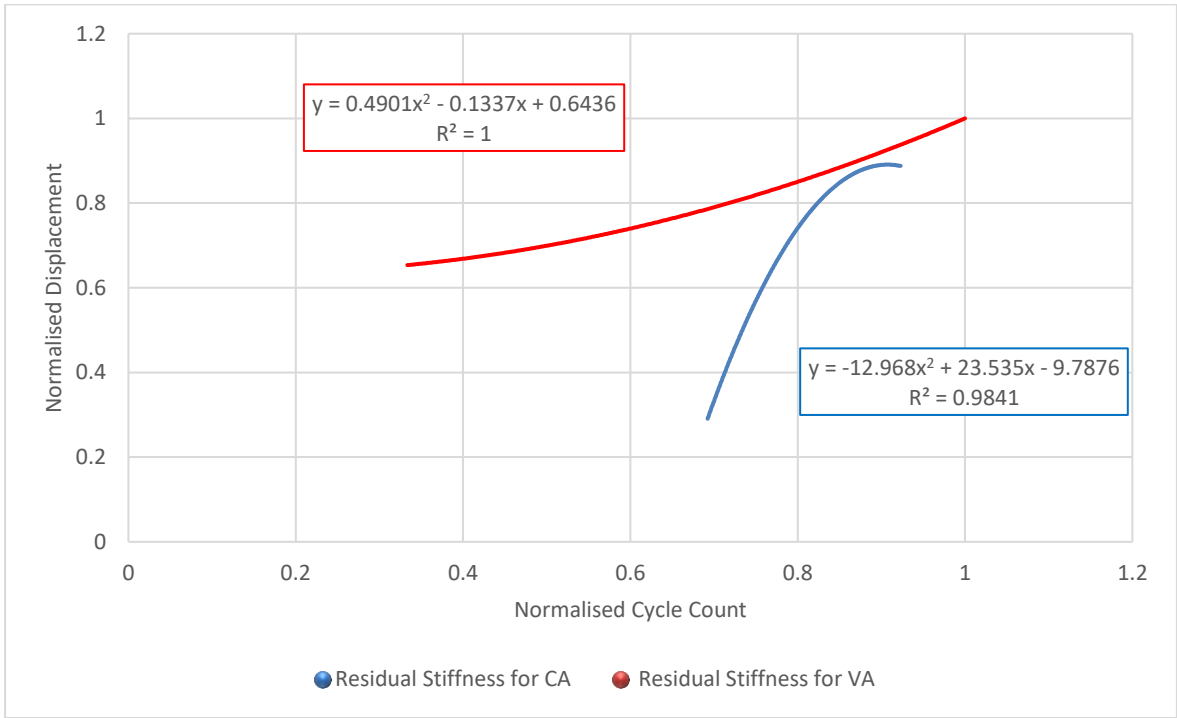


Figure 59: CA-VA Residual Stiffness Comparison

In Figure 59 the datasets are within 7 % of each other with the regression slopes having almost a similar value, which suggest that the damage of the VA cycle tests occurred at the beginning almost at the same equivalent CA cycle count.

The validation and verification of the MLS methodology was performed by plotting the Residual Stiffness and MLS results on the same graph as illustrated in Figure 60.

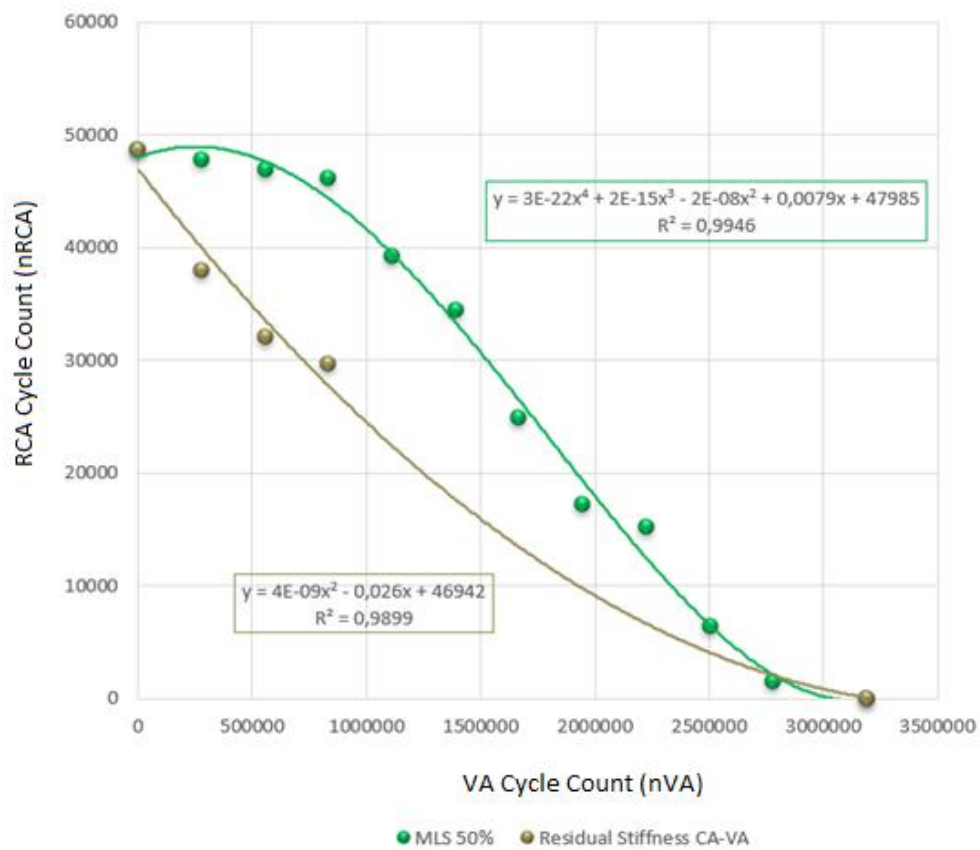


Figure 60: Residual Stiffness - MLS Comparison

The Residual Stiffness slope in Figure 60 is a 2nd degree polynomial function from the RCA axis towards the VA axis. The empirical MLS method results in a 4th degree polynomial function. The residual stiffness method indicates failure earlier than the MLS method due to the damage accumulation being non-linear, resulting in the residual stiffness method being over conservative.

4.6. Summary

In this sub-chapter the results obtained are summarised for the conclusion in Chapter 5.

4.6.1 Static Test

The static test method was straight forward and the specimen was proven to be safe against buckling under the compressive loads tested in this study. The ultimate tensile load of the coupon type specimen was determined from the test results to be 11,92 kN.

4.6.2 CA Test

The CA test methodology yielded an average failure count of 48605 CA cycles for the coupon type specimen.

4.6.3 VA Test

The VA test methodology yielded an average failure count of 3189674 VA for the coupon type specimen that translates to 22920 certifiable flight hours.

4.6.4 MLS Test

The MLS test results yielded a 4th degree polynomial function, Equation 8, between RCA and VA load sequence types for all three UTS load percentages tested. It was verified that the MLS equation was non-linear for all the tested flight load parameters including 75 % UTS. The non-linear function obtained is against the Van der Walt (2020) hypothesis that a linear correlation exists between CA and VA at flight applicable loads. The reason for the non-linear correlation is that CA cycles is a refined form compared to VA cycles, with VA cycles containing fatigue noise that does not necessarily result in damage of a specimen.

4.6.5 Validation

The Residual Stiffness and Residual Strength methods used for validation illustrated that the damage state for VA is difficult to determine since the damage build-up of the structure occurs randomly during cycling. The Residual Stress for CA, however, resulted in good results due to the damage build-up being more predictable.

The MLS methodology validation is thus more reliant on literature regarding the current certification methods used for sailplanes, compared to the Residual Strength and Stiffness methods. When comparing the Residual Stiffness and Strength methods results to the MLS method's results, an approximation can be made that the MLS methodology is valid and that further testing according to the BBA for certification of full sailplane structures are required.

5. CONCLUSION AND RECOMMENDATIONS

5.1. Introduction

The aim of this study was to evaluate the empirical Multiple Load Sequence (MLS) methodology proposed by Van der Walt in 2020, for Fibre Reinforced Polymer (FRP) material fatigue life prediction and sailplane service life extension past the 12000 Fh boundary, as defined by current European sailplane legislation requirements. Several research objectives were stated in Chapter 1, which if answered would determine whether the problem statement is satisfied through a result of the relationship between the Constant Amplitude (CA) and Variable Amplitude (VA) fatigue cycle methods determined by empirical experimentation.

5.2. Conclusion

The hypothesis of this study stated that the data results obtained from the MLS methodology for CA and VA fatigue would converge linear at loads applicable to sailplane flight loads, i.e., 50 % of the Ultimate Tensile Strength (UTS).

The hypothesis was proven untrue with the MLS test results of 75 % UTS being equivalent to that of 50 % UTS when the data was normalised. In 2020, Van der Walt determined the correlation between CA and VA loading to be non-linear, which is the same as the evaluation result.

According to the experimental results, the relationship between the CA and VA load sequences are that a correlation of a 4th degree polynomial function exists for the different load sequences, which makes it complex to interchange the load sequence types. The MLS methodology, however, remains valid through use of the 4th degree polynomial function, and the Building Block Approach (BBA) for service life extension of sailplanes past the 12000 Fh boundary.

The conclusion is that although the MLS methodology equation was found to be non-linear, it delivered a similar result compared to the validation and verification methods. There is henceforth great merit in CA testing instead of the conventional VA testing for accelerated certification of sailplanes after an adequate BBA database is established according to sailplane certification legislation requirements.

5.3. Recommendations

5.3.1 Specimen Recommendations

The coupon specimen manufacturing used for evaluation purposes of the methodology was found to be difficult, due to the fine tolerances required to reduce data scatter for evaluation of the MLS methodology. It is proposed that for the next specimen type larger tolerances as per sailplane manufacturing standard should be used.

It is recommended that the next specimen type should be similar to the shear web of sailplanes consisting out of wood, brass, high density Polyvinyl chloride (PVC) foam, carbon fibre reinforced polymer, and carbon fibre roving. An elemental specimen like the one proposed would allow for accurate data on BBA elemental level.

5.3.2 Validation Method Recommendation

The two methods used to validate the MLS methodology requires further research on the damage state of a specimen subjected to the different loading cycle types. The data gained from such a study would allow greater accuracy for exchanging the cycle types and would prove the validity of the Palmgren-Miner (PM) linear damage accumulation method for CA fatigue load spectrums possible used for sailplane certification in the nearby future.

5.3.3 MLS Method Recommendation

The MLS Method requires further testing on a Glass Fibre Reinforced Polymer (GFRP) material elemental level BBA specimen. It is furthermore recommended that strain gauges should be bonded to the specimen to obtain the poison ratio and stresses present in the specimen during testing. The strain gauge result combined with microscope observation would allow insight to the damage of the specimen at different stages of the specimen's fatigue life.

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APPENDIX A: MTS LANDMARK 100KN CALIBRATION CERTIFICATE

IMP Calibration Services (Pty) Ltd

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VAT Reg No: 4210273431

ON SITE CALIBRATION CERTIFICATE OF A UNIVERSAL TESTING MACHINE SANAS ACCREDITED LABORATORY FOR FORCE - NO: 816

CALIBRATION CERTIFICATE NUMBER: NHQ-70401 A

This certificate is a correct record of the measurements made. Copyright of this certificate is owned by the issuing laboratory. This certificate may not be reproduced except in full, without written approval of IMP Calibration Services (Pty) Ltd

COMPANY:	NORTH WEST UNIVERSITY -
CONTACT PERSON:	POTCHEFSTROOM
PHYSICAL ADDRESS:	MR. SAREL NAUDE
LOCATION OF CALIBRATION:	STEVE BIKO STREET, POTCHEFSTROOM
DATE CALIBRATED:	AT ABOVE ADDRESS
CALIBRATED BY:	29-08-2018
ITEM CALIBRATED:	S. CILLIERS
MANUFACTURER:	DYNAMIC TESTER
MODEL NUMBER:	MTS
UNIVERSAL TESTER SERIAL NUMBER:	LANDMARK
LOAD CELL / TRANSDUCER SERIAL NUMBER:	10359599
CAPACITY OF MACHINE:	400803
RESOLUTION OF MACHINE:	100kN
RANGE CALIBRATED:	0.01 kN
STANDARD USED:	0 TO 90kN
TRACEABILITY NUMBER:	100kN LOAD CELL (115696/3152)
PROCEDURE FOLLOWED:	IHQ-66434 A
	FP1301

The accuracy's of the standards used are, by way of regular inter-comparison, traceable to National Standards, within the limitations of the services available. The values in the certificate are correct at the time of calibration. Subsequently the accuracy will depend on such factors as the care exercised in handling as well as use of instruments, equipment and the frequency of use. IMP Calibration Services (Pty) Ltd cannot be held responsible for values changing, and are therefore not liable except for the recalibration of the equipment where we accept responsibility.

No long term stability checks were carried out. This certificate automatically expires if the instrument is repaired or if the seal is broken.

Page 1 of 3

DIRECTOR: MJS MATTHYSEN (NDCA)

CALIBRATION CERTIFICATE NUMBER:

NHQ-70401 A

Results:

Readings before adjustments

	UUT	STD
	kN	kN
1	0.00	0.00
2	50.00	50.54
3	90.00	90.74

TEMPERATURE	START	18,1 °C	END	18,2 °C
-------------	-------	---------	-----	---------

Reproduceability

Step	UUT	STD			AVERAGE	CORRECTION	U/C (%)
	kN	kN			kN	kN	
1	0.00	0.00	0.00	0.00	0.00	0.00	
2	10.00	10.01	10.01	10.01	10.01	0.01	± 0.3
3	20.00	20.02	20.02	20.02	20.02	0.02	± 0.3
4	30.00	30.04	30.06	30.05	30.05	0.05	± 0.3
5	40.00	40.04	40.08	40.06	40.06	0.06	± 0.3
6	50.00	50.02	50.05	50.03	50.03	0.03	± 0.3
7	60.00	60.03	60.04	60.02	60.03	0.03	± 0.3
8	70.00	70.00	70.02	70.01	70.01	0.01	± 0.3
9	80.00	80.01	80.03	80.02	80.02	0.02	± 0.3
10	90.00	90.05	90.05	90.05	90.05	0.05	± 0.3
Zero Reading		0.00	0.00	0.00			

Remarks:

- 1.) The reported expanded uncertainty is based on a standard uncertainty multiplied by a coverage factor $k = 2$ providing a level of confidence of approximately 95%, the uncertainty of measurement has been estimated in accordance with the principles defined in the GUM, Guide to Measurement, ISO, Geneva, 1993.
- 2.) The calibration was performed in tension.
- 3.) The machine was adjusted.
- 4.) The STD Load Cell was rotated.

CALIBRATION CERTIFICATE NUMBER:

NHQ-70401 A

Checked by:

S. C. LEE
Name

Technician
Designation

[Signature]
Signature

Authorized by:

M. B. N. A. C.
Name

Technical Signatory
Designation

[Signature]
Signature

Date issued:

05/09/2018

*****END*****

APPENDIX B: “EXCEL ENVELOPE SPECTRUM CREATOR”

[..\Downloads\Excel Envelope Spectrum Creator.xlsm](#)

Per (G/G)	25%																																		
TEST	atom		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32	
	aAR		A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	R	S	T	U	V	W	X	Y	Z	#	.	!	.	!	.	
	FULL SPECTRUM		10	16	84	31	847	2454	4363	5491	7193	10304	20466	32204	76383	18497	28801	4004	28782	74307	8423	8337	8741	1023	1036	250	463	694	857	37					
	420		3	0	4	21	83	442	624	1081	1573	1700	3293	5322	8881	16441	17825	6729	3621	6888	1677	2055	4408	448	2086	2072	723	365	174	50	10	0	1	0	13877
DUPLICATE TEST		0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
RANDOMIZE																																			
DUPLICATE TEST																																			
RECALCULATE																																			
FILLER																																			
CONCENTRATE																																			

APPENDIX C: SELECTED FRP MATERIAL PROPERTIES

Table 29: Selected Material Properties

Property	Value/Type	Unit
Material Type	Glass Fibre	60/40
Orientation	Bidirectional	90 degree
Weave Pattern	Plain Weave	N/A
Youngs Modulus (E)	72	GPa
Poison ratio (v)	0.23	-
Thickness (t)	0.15	mm
Weight (GSM)	202	g/m ²

APPENDIX D: MTS LANDMARK SPECIFICATIONS

Table 30: MTS Landmark® 370.10 Servo Hydraulic Specifications

Specification	Value
Dynamic Force Capacity	100 kN
Actuator Dynamic Stroke	150 mm
Min vertical test space	70 mm
Max vertical test space	1213 mm
Machine Stiffness	467 x 10 ⁶ N/m

APPENDIX E: KOSMOS II SECTION EXAMPLE

The following example of a section in the cycling spectrum is analysed to describe the KoSMOS II flight loading cycle classified as a VA cycle for compiled ASCII characters DZ. The ASCII snippet begins with a sine wave at node point (0 degrees) and then increases to load class D ($J= 0,944$) @ 90 degrees which is the crest and then decreases to node point (180 degrees) and then further decreases to ($R\text{-ratio} \cdot D$ ($J= 0,944$)) @ 270 degrees which is the trough and then increases to node point (360 degrees).

If character D is followed by character Z, the sine wave begins at node point (360 degrees), then increases to (load class Z ($J= 0,275$) $\cdot R\text{-ratio}$) @ 450 degrees which is the crest, then decreases to node point (540 degrees), then further decreases to Z ($J= -0,275$) @ 630 degrees which is the trough and then increase to node point (720 degrees).