

Accelerating the fatigue life characterization of composite sailplane structures - a novel approach

LP van der Walt

 [orcid.org 0000-0002-8122-8247](https://orcid.org/0000-0002-8122-8247)

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Supervisor: Prof AS Jonker

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Student number: 21095787

PREFACE AND ACKNOWLEDGEMENTS

My savior's fatigue

*The frailty of matter makes
books too many to read
and searching their depths make
deeper the fatigue to defeat.*

*Fatigue, oh fatigue! I hear the groan!
In study of thee, I've discovered my own.
So why shall I fight these
rules which I seek?
Alas! I must concede-
the flesh shall fatigue.*

*But I found a rule overruling
this rule: Fatigue of One flesh
gave matter meaning and
calls my spirit to confess:
"Fatigue, oh fatigue! Where is your sting?
To Christ all the glory, to Him my study I bring."*

To my wife

The completion of this study has proven to be the largest challenge of my life thus far. It has strained my whole being to the point of utter despair. Though I could continue indefinitely when asked to describe my trials, I would rather praise the faculties of my long-suffering wife. Anneke, you have encouraged me without end; waiting patiently even if it would have seemed that there would never be an end. You have been a refuge of motivation and you have done this for no personal gain. I praise you for your sacrifice.

To my friends and colleagues

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Abstract

Current accepted methods of composite sailplane life certification are based on full scale structural testing of the aircraft's wing. Two accepted test procedures exist, namely a static test to an elevated load level and a fatigue test simulating actual flight loads. Both these methods have their limitations. The static test procedure has a 12000-flight hour limitation on the certifiable life and the fatigue test procedure is very time consuming.

The service life requirement has steadily crept past the ability of the static test method and the only alternative is thus through expensive fatigue testing. Attempts in reducing the cost have been proposed from various angles, namely load spectrum truncation, empirical life predictions, load spectrum alteration and accelerated methods for fatigue curves determination.

Here an empirical method is proposed and evaluated. Based on the Palmgren-Miner assumption, it is proposed that the relationship in damage accumulation between a chosen constant amplitude fatigue test sequence and the representative flight load spectrum is linear. Consequently, this relationship can be determined with minimal empirical work and can be used to replace the time-consuming flight load spectrum with a more severe and less time-consuming constant amplitude loading sequence.

This hypothesis was evaluated on composite specimens, resembling representative structures at increased load levels. It was shown that a second order fit is more appropriate. Consequently, more empirical data is required to establish the relationship than was initially expected. Though not linear, equivalence between the two load-sequences can still be established and thus used to reduce the full-scale testing time. It is further proposed to conduct research at represented load levels, since it might reduce the slope of the second order fit in such a way that a linear approximation might be adequate.

Key words: *Residual fatigue cycles, Constant amplitude testing, Sailplane fatigue, Palmgren-Miner*

Opsomming

Huidige aanvaarde metodes vir die sertifisering van sweeftuie gebou uit saamgestelde materiale is gebaseer op die strukturele toets van 'n volskaalse vlerk monster. Daar is twee aanvaarde toetsprosedures, naamlik 'n statiese toets tot 'n verhoogde lasvlak en 'n vermoeidheidstoets wat die werklike vlugbelasting simuleer. Albei hierdie metodes het hul beperkings. Die statiese toetsprosedure het 'n beperking van 12000 uur op die sertifiseerbare lewensduur en die prosedure vir vermoeidheidstoetsing is uiters tydrowend.

Die vereiste vir lewensduur het geleidelik verby die vermoë van die statiese toetsmetode gekruip en die enigste alternatief is dus deur middel van tydsame vermoeidheidstoetse. Daar is al vanuit verskillende hoeke pogings aangewend om die koste te verlaag, naamlik belastingspektrum-afkorting, empiriese lewensvoorspellings, belastingspektrum-verandering en versnelde metodes vir die bepaling van vermoeidheidskrommes.

Hier word 'n empiriese metode voorgestel en geëvalueer. Op grond van die aanname van Palmgren-Miner, word voorgestel dat die verhouding in skade-akkumulasie tussen 'n gekose konstante-amplitude-vermoeidheidstoets en die verteenwoordigende vragbelasting-spektrum lineêr is. Gevolglik kan hierdie verwantskap met minimale empiriese werk bepaal word, en kan dit gebruik word om die tydrowende vlugbelastingspektrum te vervang met 'n swaarder en minder tydrowende konstante-amplitude-toets.

Hierdie hipotese is op saamgestelde materiaal monsters by verhoogde lasvlakke geëvalueer. Daar is aangetoon dat 'n tweede orde passing meer geskik is om die verhouding te beskryf. Gevolglik is meer empiriese gegewens nodig om die verhouding te bewerkstellig as wat aanvanklik verwag is. Alhoewel dit nie lineêr is nie, kan ekwivalensie tussen die twee toets metodes steeds vasgestel word en dus gebruik word om die volskaalse toetstyd te verminder. Daar word verder voorgestel om toekomstige navorsing toe te spits op eenvoudiger toets monsters en om meer verteenwoordigende vrag toestande te gebruik, aangesien dit moontlik sal toon dat 'n lineêre verhouding moontlik kan wees.

Sleutelwoorde: *Residuele vermoeidheid siklusse, Konstante amplitude toetse, Sweeftuig vermoeidheid, Palmgren-Miner*

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Abbreviations

BBA	Building block approach
CCF	Combined Cycles to failure
CFRP	Carbon fibre reinforced plastic
CLD	Constant life diagram
CS-22	EASA certification specification 22
D	Damage
DAM	Damage accumulation model
DLL	Design limit load
DUL	Design ultimate load
EASA	European Aviation Safety Association
ELP	Empirical life prediction
ELPP	Empirical life prediction process
ϵ ND	Strain life diagram
FCL	Fatigue compliance load
FAA	Federal Aviation Administration of the United States
FFL	Final failure load
FH	Flight hours
FCP	Fatigue certification process
FRP	Fibre reinforced plastic
FVF	Fibre volume fraction
GFRP	Glass fibre reinforced plastic
GSM	Grams per square meter
J	Factor of limit load
LBA	German civil aviation authority

LEF	Load enhancement factor
LF	Life factor
LL	Limit load
LPP	Life prediction process
NDI	Non-destructive inspection
PM	Palmgren-Miner
PVC	Polyvinylchloride
SACAA	South African Civil aviation authority
SND	Stress life diagram
UTS	Ultimate tensile strength
UL	Ultimate load

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CHAPTER 1 INTRODUCTION

The aim of this chapter is to provide a background on the subject of sailplane fatigue certification. A short history of sailplane materials and airworthiness requirements are discussed. Some historic sailplane fatigue tests, which leads to the current fatigue substantiation approaches and their challenges, are also presented. Finally, the reason for this study is explained by means of a problem statement and study objectives.

1.1. Background

The JS1 Revelation, designed and produced in South-Africa, is a high performance, type certified, fibre reinforced plastic (FRP) sailplane. The first prototype flew in 2006 and by 2016 more than 100 have been released into service. In the past ten years this glider has established itself as a worthy competitor against rival designs (Gliding international, 2017, p. 34).

Consequently, some of the gliders in the field have been in service just short of ten years and are rapidly accumulating flight hours (fh). These sailplanes' primary structures are only safe-life certified for 12000 flight hours (Straub, 2016, p. 14). Increasing the safe-life is beneficial for the owner, as well as the company, and for this reason research into safe-life extension methods has begun.

Investigations into the life certification of sailplanes are directly associated with the phenomena of material or structural degradation over time, due to alternating loads and environmental influences. This phenomenon is also known as material fatigue. Research into the behaviour of materials used on sailplanes is thus key to gain enough confidence in the structure to be able to assure the aviation authorities of the aircraft's safety and longevity for airworthiness.

Airworthiness requirements on sailplane life have logically flowed out of the idea of simulating actual service conditions on a physical structure. This method is called full-scale fatigue testing, but due to the scale and the testing time it leads to very expensive certification programs. Extensive research has been conducted to reduce the time and cost of these programs and many improvements have been made over the decades, but if the safe-life is to be extended at least one full-scale test is required by the authorities.

In this chapter the historic development of sailplane materials and fatigue airworthiness specifications is discussed. Some of the important historic full-scale fatigue tests and how the current fatigue certification requirements have flowed out of them are also shown.

1.2. History on sailplane fatigue

Material technology and development has always been a fundamental piece in the engineering design puzzle. Lighter and stronger materials (materials with better specific strength) empower engineers to achieve better designs. In many cases the development or discovery of new materials paved the way for the development of products which otherwise would have been impossible. This is especially true for the aircraft industry where materials with high specific strengths are paramount. In the sailplane industry material advances have caused enormous strides in the development of better and safer designs.

Considering the structural materials, sailplane development can be distinguished into three eras (Kensche, 2003, pp. 96-104). The first era began in 1891 with the gliders of Otto Lilienthal. His gliders made use of thin airfoils with braced wooden structures as can be seen in Figure 1-1.



Figure 1-1: Lilienthal glider
(Delta club 82, n.d.)

In the 1920's the second period started with the Vampyr which was designed with a plywood cantilever spar and a thick airfoil. Just like Lilienthal's gliders the structural material choice was wood, but plywood exploited the directional strength characteristics of wood and made the cantilever spar and thick airfoil a possibility.



Figure 1-2: Vampyr (flickr, n.d.)

The third era was initiated by the FS 24 Phönix in 1957. The Phönix was the first full FRP sailplane. Since this date FRP's have dominated the glider industry as the material of choice.



Figure 1-3: FS 24 Phönix.
(The Sailplane Class @ Penn State,
2002)

In 1971 the Kestrel 19 built by Slingsby was the first glider utilizing carbon fibre reinforced plastics (CFRP) (Kensche, 1985). This may be seen as a fourth era, because of the exceptional specific strength and –modulus of CFRP. This new material enabled the design of thinner airfoils, larger wing spans, higher aspect ratios, higher torsional stiffness of wings, improved aero-elasticity, better handling and what is of particular interest to this study: high design life due to exceptional fatigue characteristics (Kensche, 2003, p. 102).



Figure 1-4: Kestrel 19 (Wikipedia, 2016)

Each of the three eras of material development opened new doors to engineering design due to the improved mechanical properties of the materials. Unfortunately, aerospace engineers can only start using these materials in aircraft design once reliable design allowables for the materials have been established by the aviation authorities. Kensche (1985, p. 2) points to this fact, by showing that the SB10 which was built with CFRP, was designed with stress levels equal only to 50% of what is allowed today.

Practical and safe design allowables are determined through the collaborated effort of the aviation authorities, industry and academia. Each of these stakeholders play an important role in the process. Industry require less conservative design allowables to achieve better designs, whereas the authorities requires more conservative allowables to achieve safer designs. The research institutes supply the facilities and researchers to find the path between these two contradictory objectives. Finally, through this collaboration, airworthiness requirements are developed from research programs.

Though it seems logical that aircraft requires airworthiness specifications, sailplanes had no legislative requirements until 1927. The Rhön-Rossitten-Gesellschaft design rules were the first to be established and were followed by specifications such as DFS airworthiness requirements, JAR-22 and OSTIVAS (Kensche, 2003, p. 102). Today we have the *European Aviation Safety Association* (EASA) design specifications (CS-22), which are used by designers and certifiers.

The fatigue specification development of sailplane has also enjoyed considerable attention in the 20th century. This can be seen by the numerous articles published on OSTIV, a sailplane technical journal (OSTIV, 2017). This is due to rapid development of FRP's with higher design allowables (Kensche, 1995, p. 100) and the high rate of accumulation of flight hours (Esson & Patching, 1978, p. 10). Safe-life substantiation of FRP sailplanes thus became a serious field of study for manufacturers and authorities. These substantiation methods have focussed either on analytical or testing procedures of which the latter has enjoyed more attention due to the lack of enough empirical data and the intuitive confidence superiority of a physical test as opposed to a calculation.

In the 1960's the first full-scale fatigue test was done by Eugen Hänle on a Libelle H302 using a variable amplitude test program. During the same time the Akaflieg Darmstadt, in Germany, conducted a 1000 cycle constant amplitude (CA) fatigue test on a D 34d glider at minimum and maximum limit loads (Waibel, 2002, p. 56). Following this, several full-scale VA amplitude tests using the Franzmeyer spectrum¹ was conducted on sailplane wings (Waibel, 2002, p. 56).



Figure 1-5: D34
(Akaflieg Darmstadt, 2002-2004)

This testing method was founded on static tests and full-scale fatigue tests based on a theoretically derived block loading spectrum (Kensche, 1985, pp. 2-3). A Cirrus wing was tested at the Braunschweig University (Germany) using the Thielemann/Franzmeyer block

¹ See annexure 2 for an explanation of load spectrums

load spectrum and succeeded in showing compliance for a design life of 3000 fh (Pommerer, 2000, p. 2).

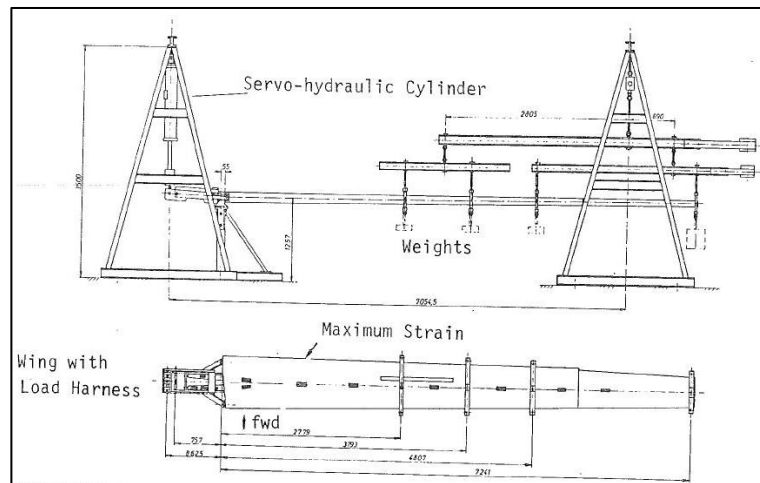


Figure 1-6: Nimbus 2 wing fatigue test.
(Kensche, 1981, p. 121)

In 1971 the Finnish glider PIK-20D had also required fatigue substantiation, because it was the first PIK-series utilising CFRP spar caps and it was found that the fatigue spectrum used on earlier models induced too low strain levels. The Nystrom theoretical load spectrum was used to test the sailplane in fatigue to 16000 simulated fh, corresponding to a certified life of 4000 fh (Nystrom, 1978, p. 38).

The Thielemann-Franzmeyer procedure was primarily developed for GFRP. The introduction of CFRP with higher design allowables, pressed the need for revised certification specifications and procedures (Kensche, 1985, p. 3). In response to this need the German Arbeitskreis Neue Fasertechniken (ANF) was established in 1977. This panel consisted of material suppliers, sailplane manufacturers, Akaflieds, research institutes as well as the local aviation authority (LBA).

The ANF conducted research on materials samples, spar beams and full-scale specimens, and was responsible for significant development on the topic of sailplane fatigue certification. A major contribution was the amendment of the certification specifications with the document: "Preliminary guidelines for the stress analysis of GFRP and CFRP sailplane and motorglider structures." In this work design allowables for GFRP was increased from 160 MPa to 250 MPa. Also, it was shown that compliance to certification specifications JAR 22.627 and JAR 22.619 can be achieved when a special additional safety factor of 1.15 is used during design and testing. This implies static tests at $1.725 \times \text{Limitload}$ (where 1.725 is calculated as 1.5×1.15) might be used to substantiate fatigue life (Kensche, 1985, p. 4).

A need to extend the 3000fh life, determined by the Cirrus test, to 6000 fh and to increase the design allowables for CFRP wings, stimulated the development of another full-scale test. In 1980-1981 a Nimbus 2C was tested by H. Kossira and C. Kensche. The wing showed compliance to a life cycle of 6000 fh and proved that CFRP can be operated at higher stresses (Kensche, 1981). CFRP design allowables was increased from 200MPa to 400MPa. Figure 1-6 shows the Nimbus 2 experimental setup making use of a servo hydraulic actuator and whiffletree to simulate a representative distributed load on the wing.

In Australia the Gliding Federation realized that the rate at which glider flight hours were accumulated in their country was higher than that of the rest of the world. This meant that Australian gliders would reach their service life much quicker than their European counterparts. This prompted an independent fatigue test program. Two Janus B wings were used especially for this purpose. One was a wing which was crash repaired while the other was brand new (Patching & Wood, 1991, pp. 100-104). The testing of these two wings continued to 35482 simulated fh, implying a substantiated safe-life of almost 12000fh when a life factor of 3 is used.

An important finding of these tests was that the structure showed characteristics of a damage tolerant structure. Damage growth was slow and could be examined by simple non-destructive methods (Waibel, 2002, p. 58).

A Polish researcher, Rodzewics (2000, p. 19), reports on the development of a load spectrum and a unique testing method of an entire sailplane, rather than only the wings as previously conducted. In his method he supports the wings and applies a load to the fuselage by means of an electromagnetic actuator rather than a servo-hydraulic actuator which is conventionally used. He shows that, through his method, 3000 fh compliance with a life factor of 3 shall take 172 days of continuous testing.

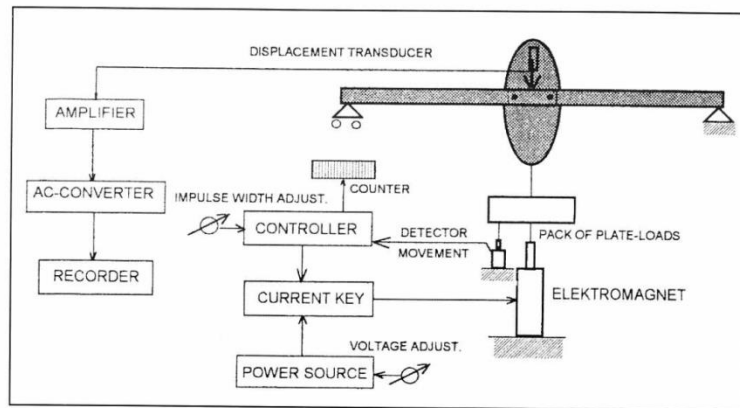


Figure 1-7: Full-scale experimental setup proposed by Rodzewicz (Rodzewicz, 2000, p. 19)

This brief historic overview has shown that numerous full-scale tests have been done in the 20th century. All these experiments have contributed to the understanding of fatigue in sailplanes, but two important milestones, which are still evident in the current sailplane fatigue philosophy, are:

- Ultimate strength testing to $j=1.725$ determined by the ANF.
- Carbon and glass FRP spar design allowables of 400Mpa and 250Mpa respectively determined through the ANF and Nimbus 2C tests.

1.3. Current approach to fatigue evaluation

The state of the art method required by EASA (2003) today, is still based on the ANF research of 1977 and the Nimbus 2C test of 1980. According to this method a life cycle of 12000fh is considered the safe life of a sailplane, if (Pommerer, 2000, p. 3):

- The primary structure is able to withstand a static limit load of $j=1.725$.
- It can be shown that standard design practices were used, based on previous substantiated structures.
- Inspection of all critical components is contained in the inspection program.
- Design stresses in the spar cap are kept below 400 [MPa] for CFRP or 250 [MPa] for GFRP.

This approach is very favourable for life cycle certification, since full-scale testing is not required and consequently the fatigue test budget is reduced dramatically. Unfortunately, the 12000 fh is the safe-life ceiling if this method is used and, as commented by Kensche (2002c, p. 32), there are already gliders in the field which have accumulated hours above this number.

Experience gained from tests conducted on sailplanes, light aircraft and wind turbine blades (which are similar to sailplane wings in design, construction and stress levels) also suggests

that the certified service life should be longer than 12000 fh (Kensche, 2002c, p. 32). It has also been shown according to Waibel (2002, p. 56) that wind turbine blades have service lives in excess of 25 years.

Though a general good feeling regarding the fatigue life of FRP structures and evidence pointing to a very high possible safe-life exist among the academia and industry stakeholders, the authorities are more conservative in their views. For this reason, as per LBA specification (Pommerer, 2000, p. 3), if a designer wishes to have a safe-life higher than 12000 fh a full-scale fatigue test with representative load spectrum and a life factor of 3 is the only option.

A full-scale fatigue test as requirement puts enormous financial strain on the designer or manufacturer, who wants to extend the safe-life past the 12000 fh ceiling, since full-scale tests run for very long periods, requires sophisticated, dedicated test facilities and personnel. Additional to these constraints, two major disadvantages exist:

1. Full-scale fatigue testing cannot prove FRP's inherent good fatigue properties. It merely extends the certified lifetime for another few flight hours. (Kensche, 2002c, p. 32) In other words, the true safe-life cannot be known unless the structure actually fails during fatigue testing.
2. The wings are usually destroyed by residual strength testing after cycling (Waibel, 2002, p. 56), meaning if further life substantiation is required, the test has to be repeated from the beginning with a new test specimen.

Due to the disadvantages of full-scale testing and the fact that safe-life extension is desired, a need for finding quicker and cheaper methods of life substantiation of sailplanes has become a study field in its own right. According to Stafiej (1993, p. 103) every possible method which can be employed to reduce the cost of full-scale testing must be explored. Even though these methods are not yet recognized by the authorities, it is no reason not to use or investigate them, because they help us to understand the problem of glider fatigue.

1.4. Problem statement

Full-scale fatigue testing of sailplane structures with a life factor of three is the only means of compliance for extending the current safe life ceiling of 12000 fh. Due to financial constraints, this method of life extension has become impractical for small to medium design and manufacturing facilities. For this reason, a need for finding a method of sailplane life extension which is less time consuming and more cost effective than full-scale testing arose.

Figure 1-8 illustrates this problem and possible solutions. The figure shows the financial cost of a fatigue certification program as a function of the certified life cycle. The horizontal dashed

line indicates the maximum financial capability of the certifier. The blue line shows how full-scale testing runs out of funding before the true safe-life is reached and the orange lines shows the possible alternative test methods which might be able to certify longer service lives for the same expense.

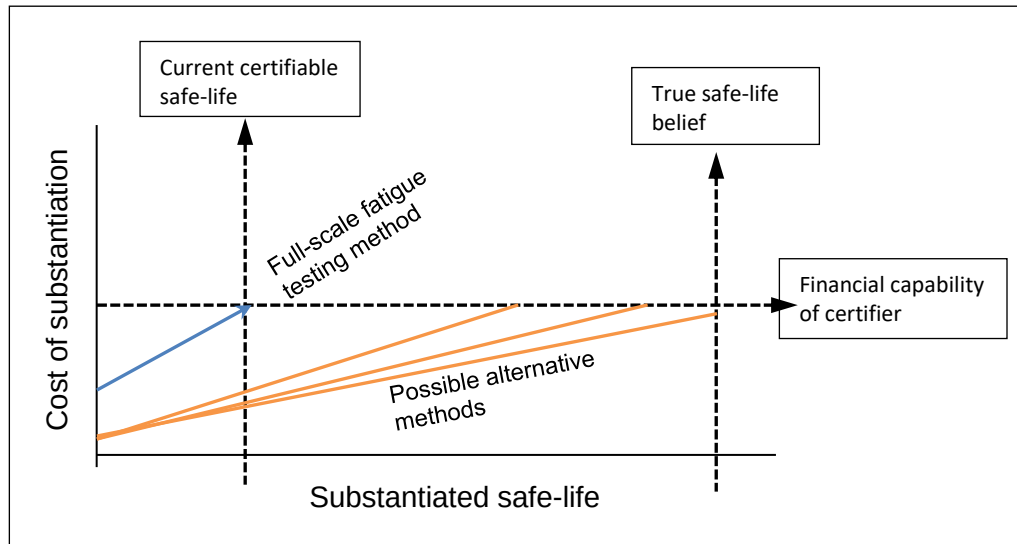


Figure 1-8: Illustration of problem statement

This study is directed to investigate possible alternatives for certifying a sailplane's fatigue life. The investigation shall focus on two areas in the current certification method which require improvement.

1. Finding methods able to prove a safe-life greater than the current limit of 12000 fh.
2. Finding methods which are less time consuming than traditional full-scale fatigue cycling with long representative flight load spectrums.

The investigation shall start with a survey of literature in which the aim is to accumulate research done on the fatigue certification of sailplane materials and structures. The methods employed by other spheres of the aerospace industry were also consulted to explore what methods is used elsewhere which could be applied to the sailplane industry. The expectation is that the literature survey shall then lead the argument to a logical conclusion of proposing an improvement in the fatigue certification process of sailplanes. This proposal shall then be evaluated. This might be an empirical, computational or any other way of verifying the proposal's applicability to sailplane certification.

CHAPTER 2 LITERATURE REVIEW

Chapter one has identified the problem as the high cost of sailplanes fatigue certification, especially for small factories. In this chapter the knowledge and insight of numerous academic and industry role players is consulted. Firstly, the current fatigue methodologies employed by the commercial, military and sailplane industries are discussed. This is followed by life prediction methods and the proposal of constant amplitude testing. Some concluding remarks are also given.

2.1. Fatigue specification, methods and philosophy

The introduction has briefly discussed the two methods through which fatigue compliance may be shown for sailplanes. This is either through load spectrum testing or ultimate strength testing conducted on a full-scale component. These two methods indicate that fatigue compliance relies very strongly on test methods and much less on fatigue design.

This notion is not entirely unique since the commercial aircraft industry also has a strong dependence on testing (Federal Aviation Administration, 1978). Firstly, simple coupon testing for material characterisation and establishing of design allowables and secondly, component and full-scale testing for validation of allowables and final fatigue compliance (Baker, et al., 2004, pp. 483-488).

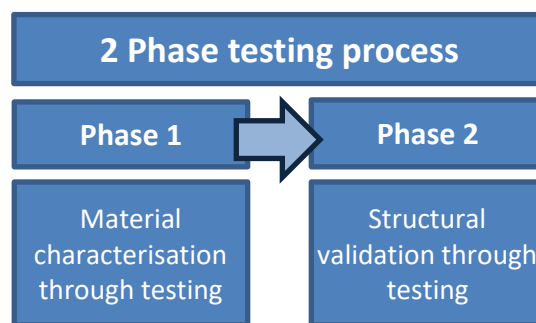


Figure 2-1: Two phase testing process

The FAA (2001, p. 11) also refers to a two phase process namely the design allowables tests and the component tests. An additional phase, which is not often explicitly mentioned in literature, is that of design. The structure which is to be validated in phase 2, naturally needs to be designed. This design is fundamentally rooted in the material characterisation phase and depending on the result of phase 2, might be iterative in nature (Department of Defence(b), 2002, p. 4.2). Hence it is not incorrect to add a phase between phase 1 and -2, which is the Design Phase. Figure 2-2 is a modification of Figure 2-1 showing the three phases rather than two.

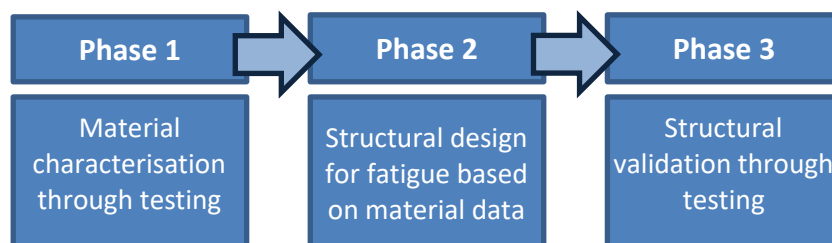


Figure 2-2: 3 Phase certification process

This three phase process lays the foundation for the continuation of all further discussions and is referred to as the Fatigue Certification Process (FCP). It is the basic process which is present in any structural certification program (Department of Defence(b), 2002, p. 4.2).

Different industries place different emphasis on each of the three phases. Through this survey of literature a comparison, based on the specific application of each of these phases, between the different industries shall be drawn to determine ways in which contributions can be made to the current sailplane methodology.

In this review of literature, firstly phase 2 is discussed which is followed by a discussion on phase 1 and -3. Both these sub-headings are concerned with the current industry practices. The last chapter shall survey the research efforts of the past 20 years which attempted to make new contributions to improving the industry methodology and authority requirements regarding sailplane fatigue.

2.1.1. Fatigue design methodologies (Phase 2 of the FCP)

Niu (Niu, 1990, p. 538) states that it is broadly recognised that an accurate solution to the fatigue problem of an aircraft is a practical impossibility, but that there exist various methods which could be employed to reduce the risk of fatigue failure. Applying these methods during the design phase (Figure 2-2) is considered as designing against fatigue and shall be referred to as fatigue design methods.

In order to design against fatigue, it is required to identify possible modes of failure of the structure (Niu, 1990, pp. 539-540). Once the modes of failure are understood, clear design requirements, –methods and -loads can be decided upon for each failure mode. Niu identifies four basic failure modes namely:

1. Static ultimate load failure- in which the airframe is designed to withstand ultimate design loads for 3 seconds without failure.
2. Fatigue life undamaged airframe- meaning the structure is designed to be free of damage for a certain time. In other words, resist the initiation of notable fatigue damage. This method is often referred to as the safe-life approach (See heading 2.1.1.2).
3. Fatigue life of damaged airframe, meaning the structure is designed to be safe for a certain time even though damage is present in structure. This is often referred to as the fail-safe approach (See heading 2.1.1.1).
4. Static residual strength of the structure in damaged state, meaning the structure is designed in such a way to always have enough strength reserve during its operational life. This is also part of the fail-safe approach (See heading 2.1.1.1).

From these four failure modes one may deduce a fatigue design philosophy. Points one and two are design criteria which ensure resistance to the initiation of fatigue damage while points three and four resist the fatigue growth of a structure which is already damaged (Niu, 1990, p.

539). A structure should be designed in such a way as to resist the initiation and growth of fatigue.

Two complimentary design methods which are used to achieve the goal set out by the above philosophy is the fail-safe and safe-life methods which are both required to design structurally safe airframes. These two methods are discussed next.

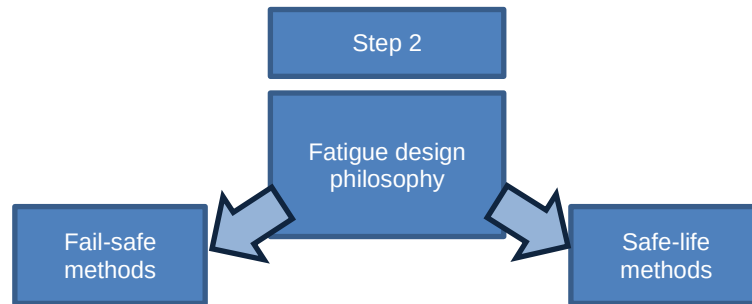


Figure 2-3: Design philosophies

2.1.1.1. Fail-Safe or Damage Tolerant design

A fail-safe design may be defined as a design where the complete or partial failure of a primary structural element shall not result in the catastrophic failure of the aircraft and that return to ground is possible (Niu, 1990, p. 554). According to Baker et al. (2004, p. 480) there are two methods to ensure that this goal is achieved:

- *Alternate load path approach*: This is a redundant structure, meaning that when failure occurs, it is localized and the remaining structure has the ability to carry the load without catastrophic failure (Megson, 2007, p. 404).
- *Slow crack growth approach*: In this approach the crack growth rate is known. The crack shall be monitored through inspection and preventative measures shall be taken before it grows to a critical size which can cause failure. This approach may be used in single load path applications without any redundancy.

A limitation to the application of the fail-safe method does exist. Authorities define a minimum threshold known as the fail-safe strength below which the residual strength of the structure may not degrade. Usually this is specified to be 80% of the limit load (Niu, 1990, p. 555). To ensure that a structure does not degrade past the fail-safe limits, periodic inspection to track damage accumulation is at the heart of this method. Figure 2-4 illustrates the method.

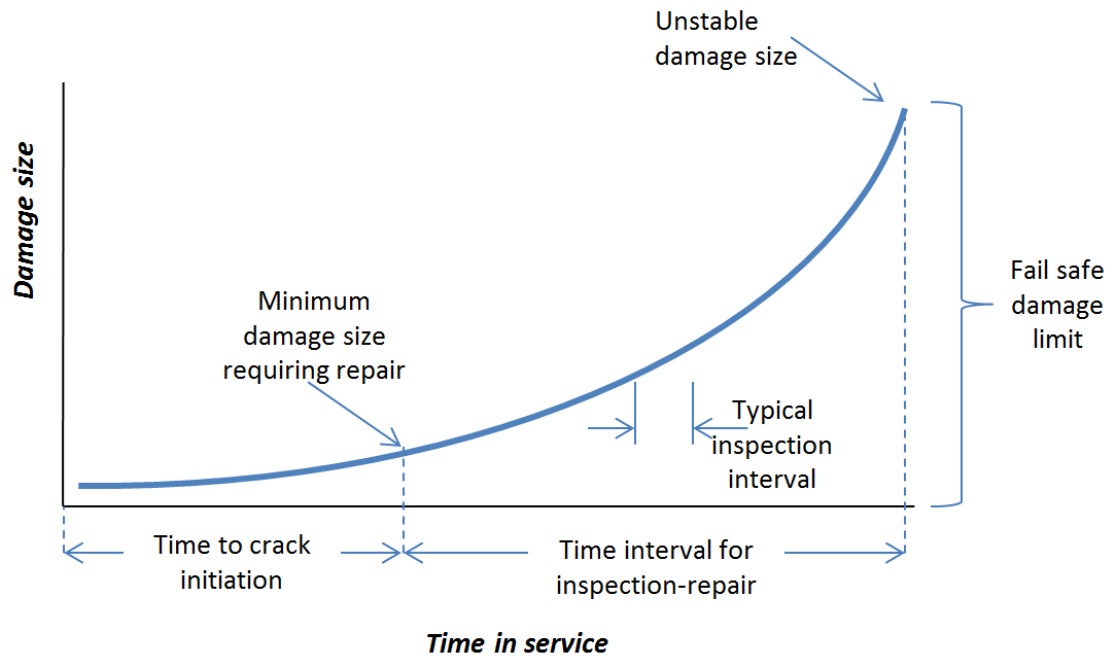


Figure 2-4: Fail-Safe design method (Niu, 1990, p. 556)

2.1.1.2. Safe life design

A part or structure designed using the safe-life approach has a known minimum life cycle throughout which it shall not fail (Megson, 2007, p. 404). The minimum life cycle is then used to define replacement intervals of aircraft parts. A part which has laboured its entire safe-life cycle must be replaced, regardless of the fact that damage can be observed or not. The design life of these components is determined through fatigue analysis and -testing of materials and components to ensure high reliability on the life prediction of a component. A safety factor on the life is also required (Niu, 1990, p. 555). Such a safety factor is often referred to as a scatter factor and is explained in Equation (1)

$$Fatigue\ Life = (Design\ Life) \cdot (Scatter\ Factor) \quad (1)$$

Figure 2-5 illustrates both the safe-life and fail-safe methodologies graphically. In this illustration the blue dashed line indicates the material strength degradation as a function of time in service. The fail-safe and safe-life intervals are shown at the bottom. The column's headers indicate the type of damage present during a specific time interval while the bottom title indicates the reason for failure, should failure occur in this interval.

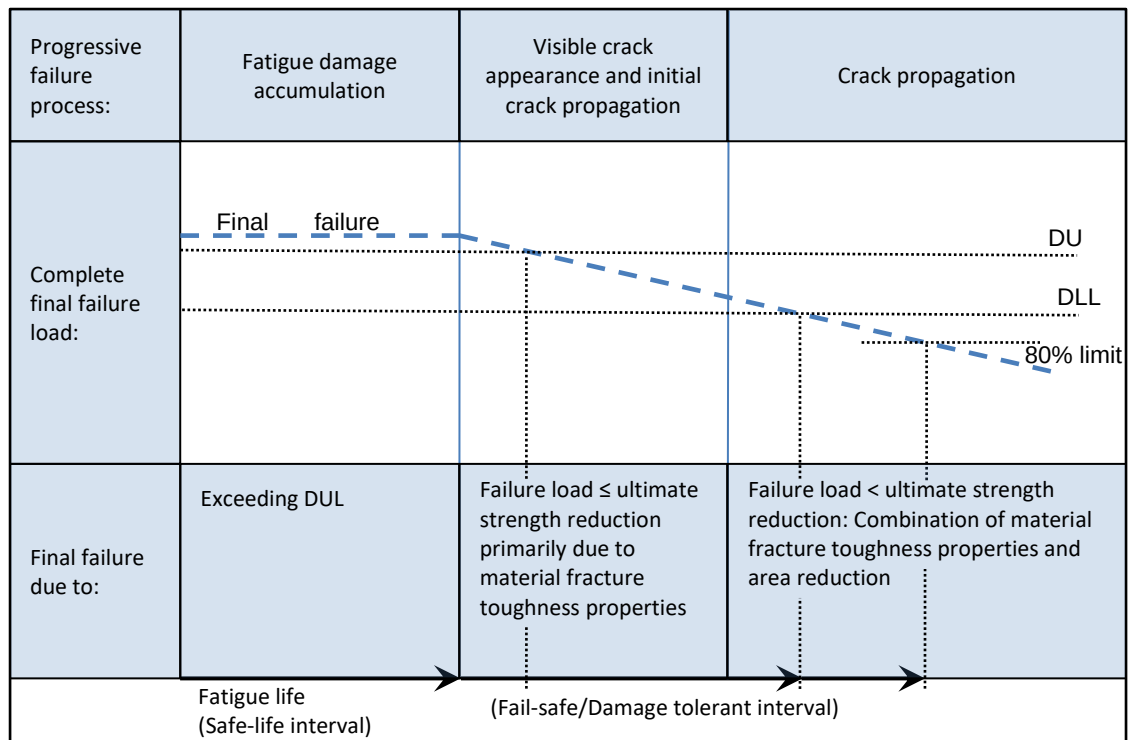


Figure 2-5: Safe-life & fail-safe methodologies (Niu, 1990, p. 541)

2.1.1.3. Sailplane fatigue design philosophy

Is not easy to find the sailplane fatigue design philosophy articulated explicitly as found for the more general aircraft industries. It might be easier to deduce the philosophy from the various certification requirements which have been compiled. The most current accepted specifications are not found in a single legislative document but spread across a vast resource base which is not always very attainable. For the sake of the argument three resources were found which gave the most consistent and mutually supportive information. These documents are:

- Standards for Structural Substantiation of Sailplane and Powered Sailplane Components Consisting of Glass- or Carbon FRP.
- Certification Specifications for sailplanes and powered sailplanes CS-22.
- Minutes of LBA/DGAC meeting dedicated to 12000 hours gliders fatigue life-November 14, 2000.

The important design requirements which could be extracted from these documents were:

- CS22.303 which specifies a 1.5 safety factor as the baseline unless specified differently (European Aviation Safety Agency, 2009, pp. 1-C-1).
- CS22.627 which requires the avoidance of stress concentrations as far as it is practicable (European Aviation Safety Agency, 2009, pp. 1-D-2).
- CS22.603 (a) which requires that there exist confidence in the selected materials of safety critical components, either based on experience or testing (European Aviation Safety Agency, 2009, pp. 1-D-1).
- CS22.305 (a) which states that static limit loads may not permanently deform the structure (European Aviation Safety Agency, 2009, pp. 1-C-1).
- CS22.305 (b) which specifies that the structure should be able to withstand ultimate load for at least 3 seconds (European Aviation Safety Agency, 2009, pp. 1-C-1).
- If a full-scale fatigue test is not done, the following additional design requirements should be met (Pommerer, 2000, p. 4).
 - Stress levels, in critical parts of the wing spar, must be kept below allowable stresses given in the LBA Standards (400 N/mm^2 for CFRP and 250 N/mm^2 for GFRP).
 - Wing structure should be able to withstand up to $1.5 \cdot 1.15 \cdot \text{Limit loads} = 1.725$ for 3 seconds.
 - The design of the sailplane wing should be similar to that of a sailplane wing which has been previously substantiated through full-scale testing.

- Structural substantiation by analysis will be accepted only if it has been shown for the selected design by experience and knowledge based on tests that the calculation method used provides reliable results (Luftfahrt-Bundesamt, 1991, p. 6).

Considering these design requirements, some light is cast on the philosophy behind sailplane fatigue design. The majority of the specifications have requirements with regard to the static strength of the structure; like static stress limitations and special static safety factors. There is no explicit requirement on fatigue design except the broad guideline on stress concentrations. It seems that if all static load requirements are met, a sailplane complies, based on past experience where these specifications proved to be satisfactory for fatigue.

The philosophy for sailplane fatigue design can thus be graphically shown as in Figure 2-6. The static load requirements which are designed for are denoted LL, UL, FCL, FFL referring to limit load, ultimate load, fatigue compliance load and final failure load, respectively. Complying with these static requirements are considered adequate design against fatigue and ensures a certified product life of 12000 fh (Pommerer, 2000).

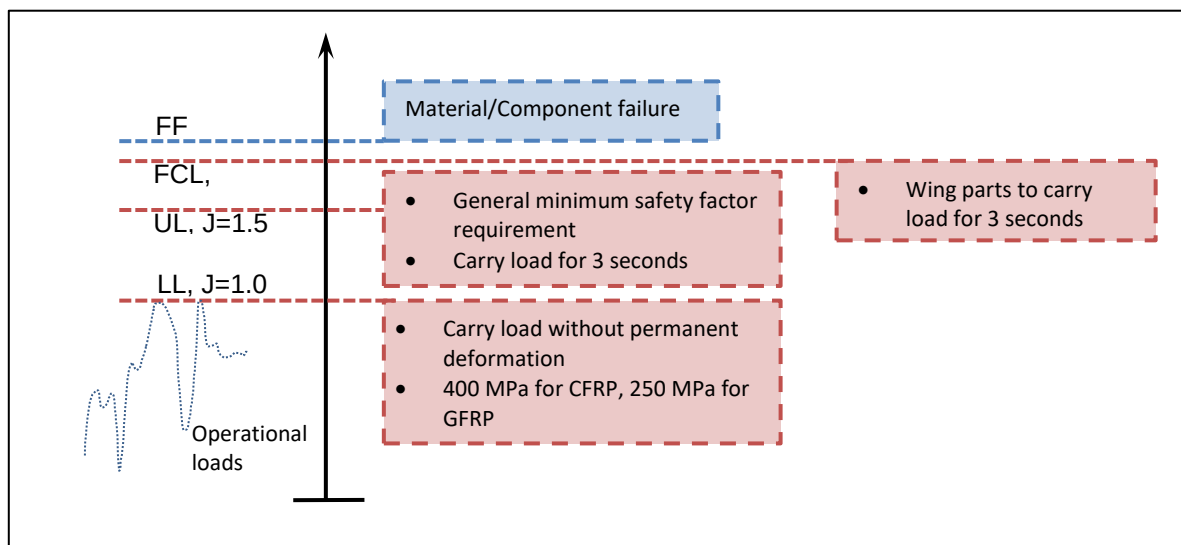


Figure 2-6: Design philosophy

Following this philosophy, the damage accumulation rate and the true safe-life are not predicted and for this reason remains an unknown throughout the glider's life cycle. The safe life of 12000 fh is arbitrarily chosen based on informed experience. This might be a conservative safe-life but due to improvement in contemporary material systems, the method might be critiqued on being too conservative and thus limiting the designer.

The fatigue design methodologies of the sailplane industry and commercial aircraft industry can now be compared. Based on the possible failure modes as shown earlier by Nui (Heading 2.1.1) it becomes clear that an explicit design philosophy against the initiation or the

propagation of fatigue damage in the sailplane airframe does not exist. Compliance to modified excessive static requirements fulfils the fatigue requirement as well. Table 2-1 illustrates the comparison.

Table 2-1 Design philosophy comparison

Failure mode	Commercial and military industry	Sailplane Industry
Static failure	✓ Static design	✓ Static design with special modifying factors
Fatigue initiation	✓ Safe-life	✗
Fatigue propagation	✓ Fail-safe	✗

Some comments on Fail-Safe and Safe-Life for sailplane

Though the safe-life and fail-safe approaches are not currently standard in the Sailplanes industry, a handful of researchers have made some comments on their application. The conclusions are currently still somewhat contradictory, which indicate that this is a topic requiring extra research.

The LBA (Luftfahrt-Bundesamt, 1991) makes provision for a safe-life analysis in their specification, if confidence in the method can be proven. Payne (1987, p. 50) has shown through specimen testing that the safe-life approach may have merit, but due to sudden failure behaviour the fail-safe approach has less merit.

Waibel (2002, p. 58) on the other hand commented that the Australian fatigue test conducted on a Janus wing showed characteristics of a damage tolerant structure- which is in accordance with the fail-safe approach. Also, Trappe (2008, p. 56) comments in one study on a sailplane's structure the importance of further study into fail-safe methods for life estimation of FRP aircraft.

An example of employing both the safe-life and fail-safe approach on a sailplane has been demonstrated in the fatigue analysis of a PIK- 20D sailplane (Soinne, 2015, p. 73). Steel components on this sailplane were safe-life limited at 65 000 fh, after which they must be replaced. Safety of the rest of the structure is ensured through a special fatigue inspection every 5000 fh starting at 10000 fh. This is in accordance to the safety by inspection of the fail-safe methodology.

In a study of Kensche (2002c) on composite beams similar in design to a sailplane spar, a strong case was made that the life of these structural elements is for all practical reasons, infinite and safety through inspection might be an adequate means of complying for continuous airworthiness.

2.1.1.4. Concluding remarks on sailplane fatigue design

It is interesting to find the design approach difference between the commercial and the sailplane industry. The main difference is the sailplanes industry's reluctance to use the safe-life or fail-safe approach. The reasons for this are not clear from literature, but one may venture to speculate some reasons.

The safe-life and fail-safe approaches were both developed in an era when aircraft were predominantly manufactured using metals as the structural material of choice. The fatigue failure mechanisms for metals are known and reliable models for predicting the behaviour have been developed. Sailplanes designers have chosen composites as material of choice long before the other industries.

Fatigue failure mechanisms in composites are manifold and modelling the behaviour is exceedingly difficult compared to metals (Vassilopoulos, n.d., p. 3). Applying the safe-life and fail-safe principals to composite materials could be seen as breaking new ground, for which the sailplane industry has limited resources. For this reason using safe-life or fail-safe might remain impractical unless inexpensive methods of damage monitoring and establishing material fatigue behaviour can be developed.

Another reason for not innovating on the subject of fatigue design for sailplanes might be out of pragmatic reasoning. The current method is established, reliable, cost effective and practical. It might not be the best method and might have a few limitations, but for the time being it works.

As shown in the introduction of this study, and the objective of the study, the major limitation is the cost of extending the safe-life past the 12000 fh ceiling. Good safe-life or fail-safe analysis methods could assist in resolving these limitations in the following ways:

- If safe-life design methods can be developed and show reliable results, some structural validation tests (phase 3) might become totally unnecessary thus eliminating the most costly part of the FCP.
- If inspection methods can be developed which can identify the progress of fatigue damage, then fatigue safety might be shown through inspection and not be limited at all by a safe-life.

2.1.2. Fatigue testing methodologies (Phase 1 and phase 3 of the FCP)

In this chapter the testing practices used in the FCP shall be discussed. Both phase 1 and phase 3 of the FCP is concerned with physical testing of either material or components and for this reason both these phases shall be objects of this review. Phase 1 is primarily concerned with establishing design data which is required for phase 2 and phase 3 is concerned with the validation of the design which was done in phase 2.

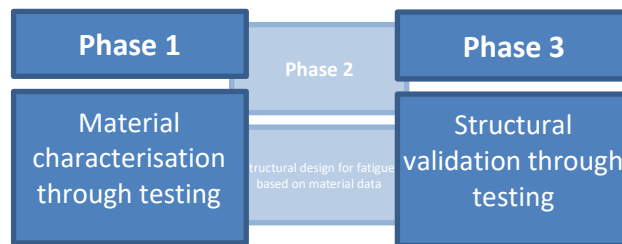


Figure 2-7: Phase certification process

The engineering process of characterisation of the material and structural behaviour of an aircraft can be seen as a constant trade-off between analysis and testing (Department of Defence(b), 2002, pp. 4-1). Analysis is cheaper, but requires empirical design values and validation while testing is inherently trusted, but usually more expensive. In this trade-off, reliability and cost should be in equilibrium. In the course of the 20th century the aircraft industry has laboured hard to devise some clever methods of achieving this equilibrium. These methods shall be discussed here and compared to the sailplane industry.

2.1.2.1. Building Block Approach to structural substantiation

Limiting the Building Block Approach (BBA) to the category of testing methods shall be an injustice. The BBA is rather a holistic approach to the entire FCP in order to substantiate structures' mechanical performance with a high degree of reliability at the least amount of financial cost. It is listed in this chapter because of its high dependence on physical experiment.

The BBA is a synergistic method, combining analysis and testing. Analysis is used to guide the tests and the tests used to validate the predictions. This approach reduces technical risk and optimizes expenditure, by still meeting the material and structural reliability requirements. (Department of Defence(b), 2002, pp. 4-1)

The BBA entails a large test program, investigating test specimens of increasing complexity, starting with simple material coupons and ending in a full-scale aircraft. Each level of complexity builds on the knowledge gained from the previous less complex stages. In other words, the design and testing practices used for the more complex specimens are developed from the less complex specimens' results. By focussing the investigation on the less complex

and cheaper specimens while reducing the number of specimens in each succeeding complexity level, considerable cost savings is achieved. (Department of Defence(b), 2002, pp. 4.1-4.2).

The BBA can be explained by means of a pyramid illustration as seen in Figure 2-8. The base consists of many low complexity specimens, generic in nature with the goal of determining the standard design allowables, which can be documented in a database for generic applications. This is a typical phase 1 process of the FCP. The higher levels are more non-generic, with fewer specimens and concerned with design validation and consequently can be classified as phase 3 processes of the FCP (Baker, et al., 2004, pp. 481-487). Table 2-2 elaborates further on the illustration of Figure 2-8.

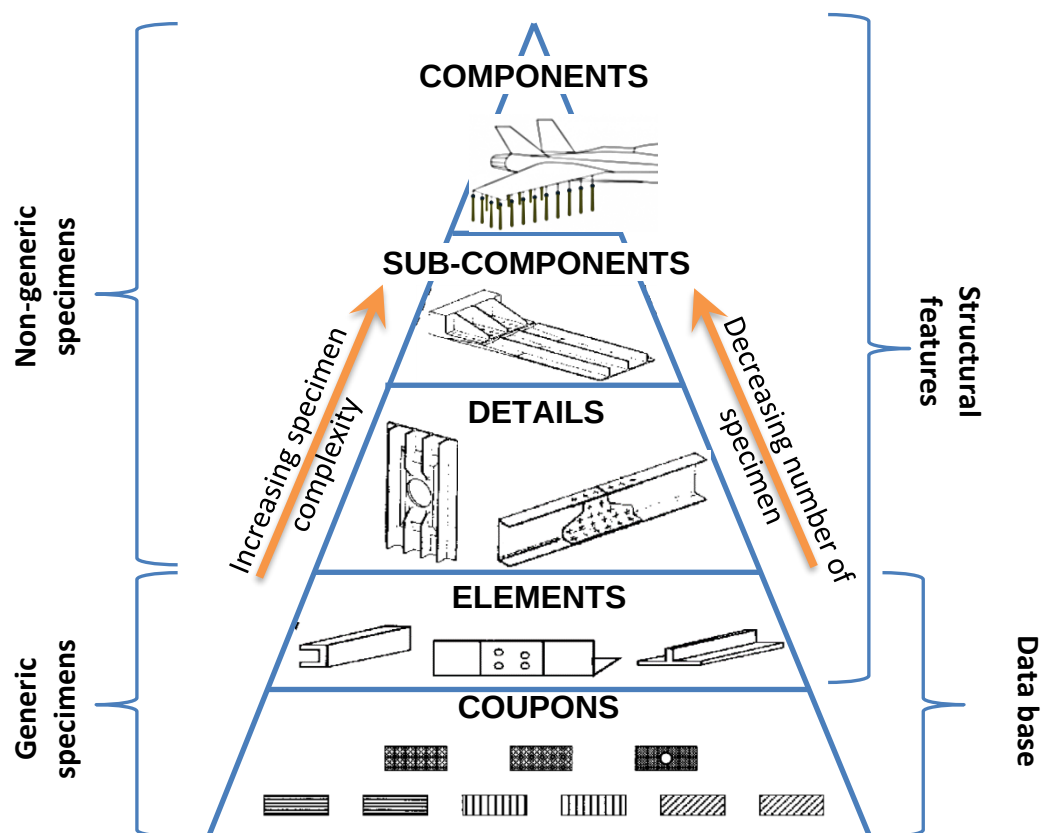


Figure 2-8: Building block approach.
(Department of defence(a), 2002, pp. 2-2) (Interface advanced force measurement, 2016)

Table 2-2: Building block approach to airworthiness tests
(Baker, et al., 2004, p. 481) and (Mayer, 1996, p. 7)

Level	Building block	Typical number of specimens	Typical size	Objective	Relative cost
1	Material coupons	400	100mm	• Generate generic data base • Material design allowables	14%
2	Structural elements	50	200mm	• Generate a generic feature database • Represent all potential failure modes • Check calculation rules	
3	Structural details	10	1m	• Generate non-generic design values • Check damage tolerance	3%
4	Sub-components	4	3m	• Check for unexpected failure modes • Compare with detail and element tests	13%
5	Component	2	15m	• Check strain levels against failure strains • Demonstrate airworthiness compliance	70%
	Entire aircraft	1	30m	• Check strain levels • Demonstrate airworthiness compliance	

Furthermore, the BBA is an iterative design and validation approach in which uncertainty in the design practices is filtered out systematically as the process progresses through the levels. An investigation may not continue to the next level until adequate design reliability has been achieved in the current complexity level. Figure 2-9 illustrates this internal iterative process from one complexity level to the next (Department of Defence(b), 2002, p. 4.2). Also, note the three phases of the FCP found inside the building block work flow.

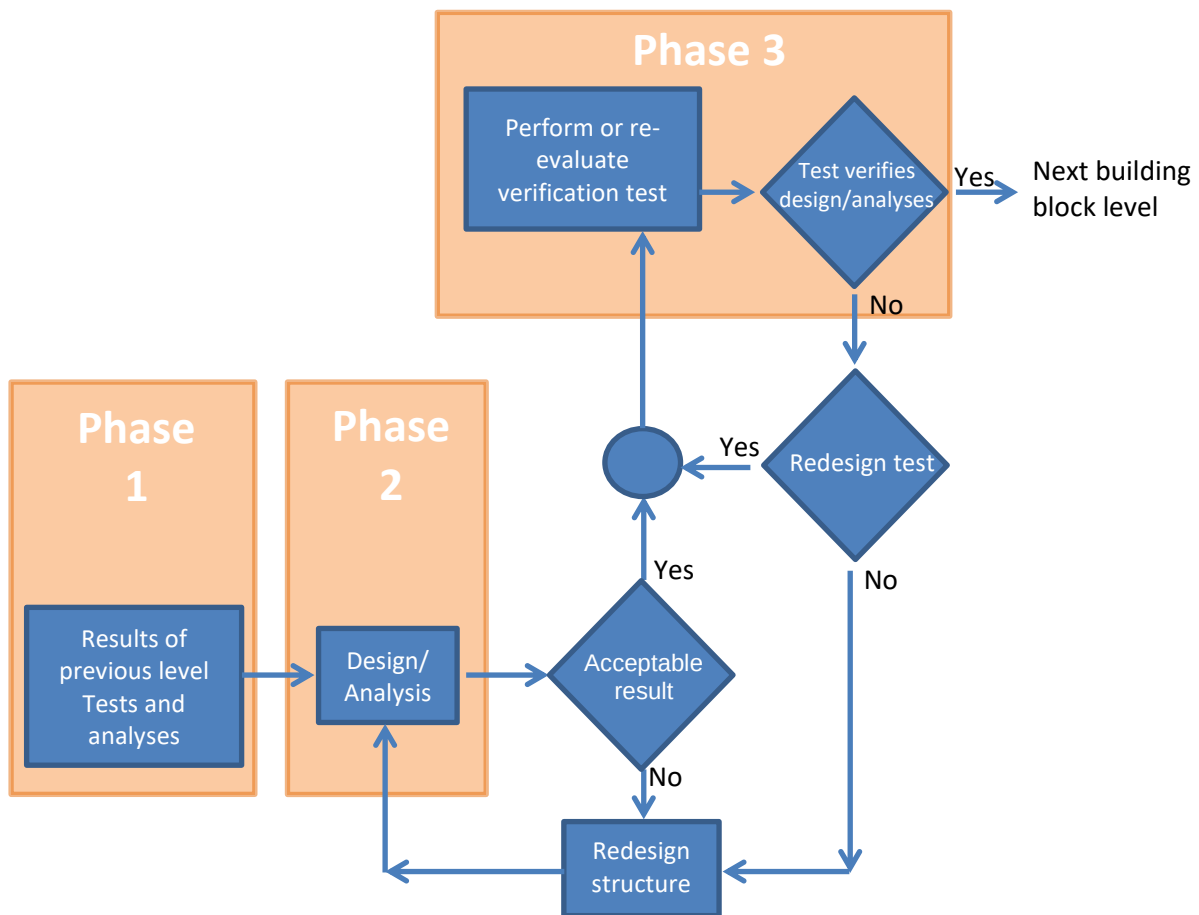


Figure 2-9: Iterative process in the building block approach.
(Department of Defence(b), 2002, p. 4.2)

As can be seen in Table 2-2, though the BBA reduces the amount of high level complex specimen, the cost distribution still leans strongly to the high level tests (Phase 3 of the FCP). According to Mayer (1996, p. 7) the high-level test may account for 70% of the entire structural substantiation cost even though the BBA is used. This shows that attempts in reducing the cost of this phase is definitely worth pursuing. In the next chapters, methods of achieving this are discussed.

2.1.2.2. Maintaining reliability during the BBA

The BBA has been successful in reducing the amount of test specimens required for validation without compromising on the reliability. This has traditionally been done by means of the Life Factor (LF, also referred to as a scatter factor) approach in which a test specimen is tested for a larger amount of design life cycles in order to qualify the design life (Waruna & Tomblin, 2012, p. 4).

The required LF is calculated based on the data obtained from a lower level data set. The LF is defined as the ratio between the mean life to the life at a percentile of the designer's choice (Tomblin & Seneviratne, 2011, p. 22). Most often this percentile is 90th or 95th which is referred to as the B- and A-Basis² values respectively (Baker, et al., 2004) in the aircraft industry. The Life Factor can be understood by graphical explanation (Figure 2-10).

As an example, Figure 2-10 shows typical strength reduction data for the lower level building block tests as a function of design life times. A component under consideration for validation, in a higher level, based on this data, shall achieve B-Basis reliability for a single design life if:

- The maximum design fatigue load is applied.
- The specimen is able to resist failure for a time equal to at least the mean failure life time (in this case 5 design lives) of the lower level data.

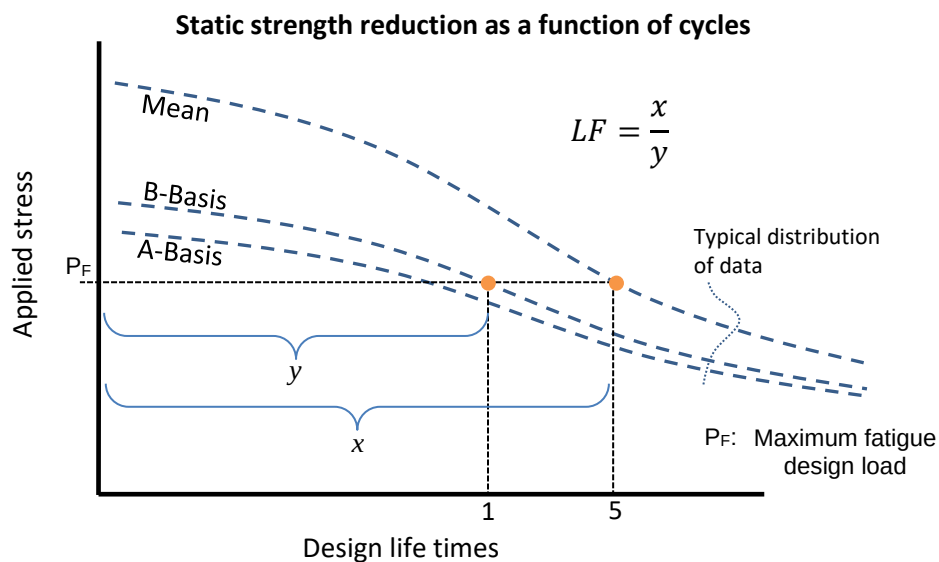


Figure 2-10: Life Factor
(Whitehead, et al., 1986, p. 43)

Data obtained during fatigue testing of coupon-, component- or full-scale structures manufactured of composites, exhibit much larger scatter than that of metals. This fact is

² A- and B-Basis allowables are discussed in annexure 7

illustrated in Figure 2-11 where a comparison of the probability density functions of the tensile strength of composite and metal specimens is shown. Tomblin (2011, p. 7) shows a case where a metal structure required a LF of 2 and an equivalent composite structure required a LF of 13 for equivalent B-Basis reliability. The large scatter observed in composites had such a big influence on the life factor, that the application thereof had become almost impractical.

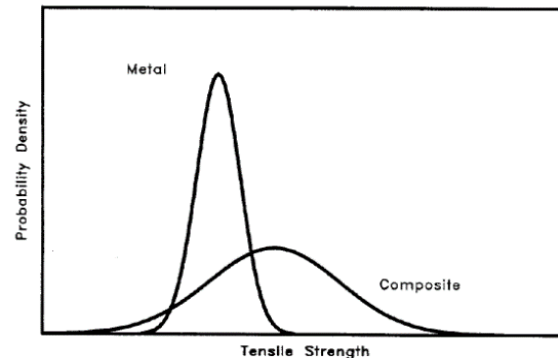


Figure 2-11: Probability density function comparison between composites and metals (Department of Defence(b), 2002, p. 9.3)

A solution for this problem was found in the Load Enhancement Factor (LEF) approach. In this method, rather than increasing the test cycles, the load which is applied to the test specimen during the fatigue test is increased in such a way as to decrease the test duration and maintain the same reliability in the results as in the LF approach (Waruna & Tomblin, 2012, p. 5).

The LEF can be defined as the ratio of the maximum fatigue test load to the maximum fatigue design load and is explained graphically in Figure 2-12. If a similar reliability is required for the LEF as in the LF the maximum fatigue test load, P_T , is then equal to the mean static strength at 1 design life.

In some cases, the LEF have had a too severe an effect and caused failure modes which are not representative. In such cases two methods are used to overcome this problem.

- A combination of LEF and LF is employed. This has the effect of reducing both the LEF and the LF (Whitehead, et al., 1986, p. 48).
- Clipping of the high cycles of the fatigue load spectrum. In this method the high amplitude load cycles which may contribute to undesirable failure modes is omitted from load enhancement. A LF approach is used on these cycles while the rest of the spectrum still makes use of the LEF (Waruna & Tomblin, 2012, p. 10).

Another method, other than the LEF, which is used to reduce the testing time, is named spectrum truncation (Kossira, 1997). Spectrum truncation is based on the assumption that low amplitude cycles do not contribute to the damage accumulation, but that the fatigue life is only a function of the higher amplitude cycles. Following this assumption, all cycles below a certain threshold are omitted from the spectrum, consequently reducing the amount of cycles in the test program and reducing test time.

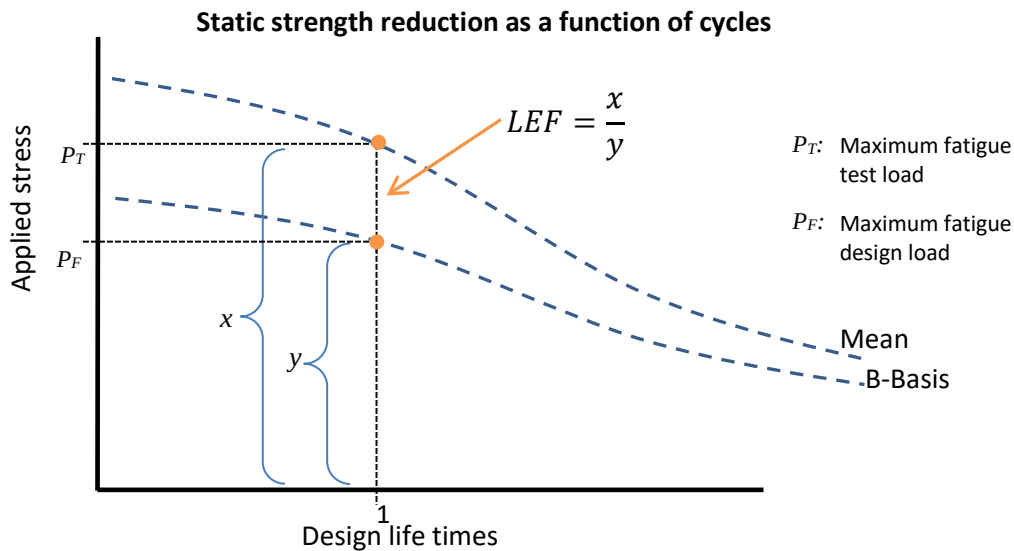


Figure 2-12: Load Enhancement factor.
(Whitehead, et al., 1986, p. 43)

This threshold is most often based on the endurance limit (Waruna & Tomblin, 2012, p. 13). Composite materials exhibit very high endurance limits, which means that they are relatively insensitive to low amplitude cycles. The threshold/endurance limit should be determined through experiments, but various studies have shown that with composites a truncation level of as high as 70% of the test limit load can be achieved without compromising reliability (Department of Defence(b), 2002, pp. 7-57). Figure 2-13 shows a graphical representation of a spectrum. The number of occurrences is plotted as function of load level. The vertical dashed line indicates the loads which are truncated from the spectrum.

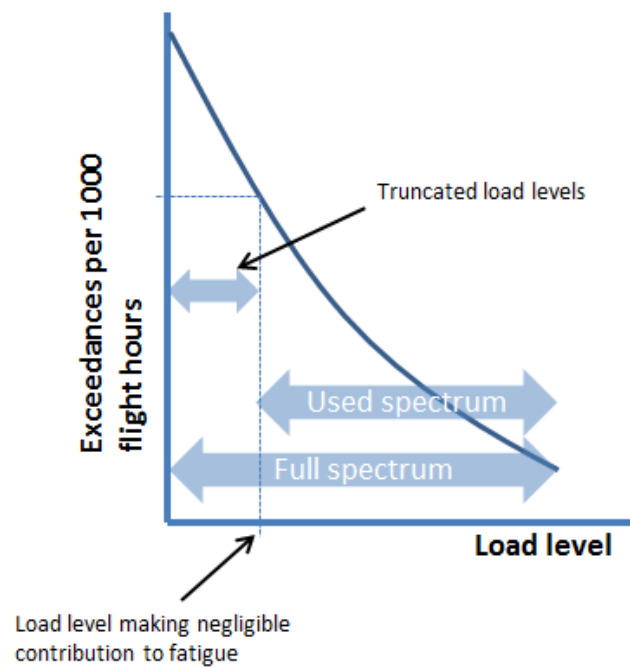


Figure 2-13: Spectrum truncation
(Waruna & Tomblin, 2012, p. 13)

In other cases, a more aggressive approach to reduce test time and expenses has been taken. This approach is named the ultimate strength approach. The ultimate strength approach achieves adequate reliability in fatigue life by increasing the design ultimate load (DUL) (Whitehead, et al., 1986, p. 53) with a certain factor. This factor is calculated from the results of lower level fatigue tests. A successful static test to this increased DUL is seen as an accepted means of fatigue compliance (Department of Defence(b), 2002, p. 7.57).

2.1.2.3. Sailplane fatigue testing philosophy

As seen in the previous section, the military and commercial industry have already developed ingenious methods to reduce testing time, risk and cost of large fatigue testing programs. These methods include BBA, life factor, load factor, ultimate strength testing and spectrum truncation. To gain some important insight into sailplane fatigue philosophy it is helpful to compare these methods to the methods used by the sailplane industry.

The methods of fatigue testing used by the sailplane industry can be found within the specifications of the three documents (European Aviation Safety Agency, 2009), (Pommerer, 2000) and (Luftfahrt-Bundesamt, 1991). The specifications are summarised according to phase 1 and phase 3 of the FCP which are both concerned with testing of material and structures.

Phase 1 specifications

- CS 22.603 (a) which requires that there exists confidence in the selected materials of safety critical components and that testing is an approved means of establishing this confidence (European Aviation Safety Agency, 2009, pp. 1-D-1).
- All materials used on the sailplane are to be tested and the LBA should approve them (Luftfahrt-Bundesamt, 1991, p. 1).

Phase 3 specifications

- Full-scale testing is required to substantiate the life cycle. As mentioned earlier the certifier may choose one of two options namely either static ultimate test or spectrum fatigue test.
- If the spectrum fatigue test is chosen, the following additional requirements apply:
 - A life factor of at least 3 is required.
 - After completion of the fatigue test, the component is again to be loaded statically at a temperature of up to 54°C to a load up to limit load ($j=1.0$) and held for 3 seconds (Luftfahrt-Bundesamt, 1991). Investigation for deformations is to be conducted after this test.
 - Load up to ultimate load ($j=1.5$).
- If the ultimate static load option is chosen the following additional specifications apply (Pommerer, 2000, p. 4):
 - Stress levels, in critical parts of the wing spar, must be kept below allowable stresses given in the LBA Standards (400 N/mm² for CFRP and 250 N/mm² for GFRP).

- Wing structure should be able to withstand loads up to $1.5 \cdot 1.15 \cdot \text{Limit loads} = 1.725$.
- The design of the sailplane wing should be similar to that of a sailplane wing which has been substantiated previously.

Following from these testing specifications the phase 3 testing philosophy for sailplanes can be deduced. Figure 2-14 illustrates this philosophy graphically. Note that both options of fatigue testing are shown. In this figure the stress level is given as a function time. Option 1 shows the spectrum test with a life factor of three followed by the static test procedure to LL and UL. Option 2 shows the static test procedure to LL, UL and FCL.

First impressions might lead one to think that the two options are not related to one another at all, but close investigation reveals the contrary. Though option two might be thought of as "over designing" due to the larger safety factor requirement of >1.725 , this larger safety factor is indirectly also built into option one, since option one must take reduction in load carrying capacity into account. The breaking load line with a negative slope shows this reduction as a function of applied load cycles in the option 1 picture.

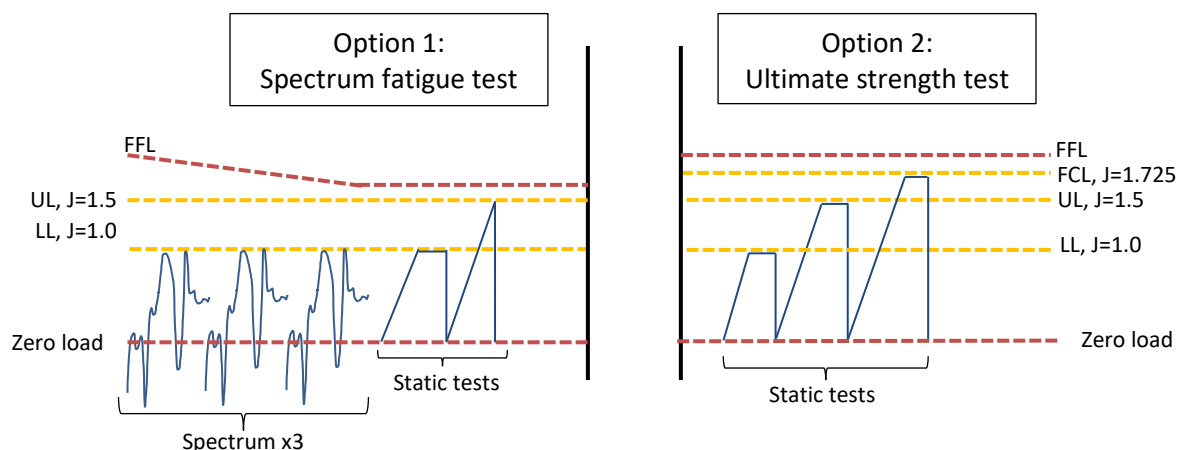


Figure 2-14: Sailplane fatigue testing philosophy (stress level is given as a function time)

To the best of our knowledge, the rationale or philosophy behind the sailplane testing approaches is not generally available and for this reason it is difficult to understand or discuss. When they are compared to the other methods employed by industry some interesting points can be highlighted:

- **Life Factor of 3:** According to the building block approach discussed earlier, a one specimen full-scale fatigue test cannot yield results with high enough confidence without extensive support of less complex test specimens or a very large life factor. Literature has shown examples of life factors for a one specimen composite fatigue

test, which are much larger than the sailplane requirement (Department of Defence(b), 2002, p. 7.54).

Following this, it is interesting that a single full-scale test of a composite sailplane with a relatively small life factor of 3 is considered an adequate means of fatigue compliance in the sailplane industry. This, most probably, has its origins in the FAA recommended life factor for small aircraft of metal construction (FAA, 1973, p. 8).

One example was found of larger life- factors and load enhancement used on a sailplane, but this was not done under Finnish National Transport Agency and not the LBA and EASA legislation (Soinne, 2015).

- The application of the BBA, LF and LEF in the sailplane industry could not be explicitly found. The ANF tests of the 1970's and the investigation on the Nimbus 2 included specimens of different complexities (Kensche, 1985, p. 4), (Kensche, 1995, p. 69) which thus hint to the application of BBA.
- The 0.15 additional factor to achieve $J=1.725$, which is used as the static test load to show fatigue compliance seems to resemble the ultimate strength approach described by Whitehead et.al (1986, p. 53) and discussed earlier. Whether this factor was determined by lower level empirical data as Whitehead et.al prescribes, remains uncertain. As understood from Kensche (1985, p. 4), the factor is merely a modifying factor to prove that regions of stress concentration was avoided as required by JAR 22.627 (Anon., 2001, p. D3).
- Truncation of the spectrum to reduce testing time is a method also used by the sailplane industry. This can be seen in the revised KoSMOS II spectrum (Kossira, n.d.), which reduced the number of cycles from 1.7×10^6 to 5×10^5 cycles. This was achieved through omission of all cycles with amplitudes 7.14% lower than the maximum amplitude (Kossira, n.d.). Also Rodzewics (2009, p. 455) proposed a special method of truncation for application in sailplane certification in which some loads are omitted based on their contribution to the total damage of the part.

2.1.2.4. Concluding remarks on Sailplane fatigue testing

It is interesting to find the testing approach differences between the commercial and the sailplane industry. The main difference is that the BBA method is not used by the sailplane industry.

The concept of a life-factor is present in the Sailplane FCP but is always assumed to be 3 while the life-factor approach in the commercial aircraft industry is based on a statistical analysis of the lower level empirical data of the building blocks.

The ultimate static loading of a full-scale specimen is used to show fatigue compliance. This also is a concept coming from the commercial industry, but the application yet again is different. When applied to the commercial aircraft the static load is calculated through a statistical analysis of lower level building block data while for sailplanes it is a fixed value for any sailplane, namely $1.725 \cdot LL$.

The reason why the sailplane requirements are not as elaborate as the other industries' is not explicitly found in literature. A guess might be that the complete BBA approach was used when the Nimbus II experiments were done and that the results of these experiments is now applicable to all sailplanes which are similar in design to the Nimbus II. Following the certification requirement that: *"Standard design practices are used"* (Pommerer, 2000, p. 3)

Commercial and military industry has applied the BBA with success, but it is a process which is much more complex than the process currently followed by the sailplane industry. Making a carbon copy of the BBA for the Sailplane industry shall not help in finding a solution for cost effective life extension. It shall most probably increase the cost, but using the elements thereof to establish a customised BBA for the Sailplane FCP could be advantageous from a cost and reliability point of view.

2.1.3. Concluding remarks on sailplane fatigue philosophy

The fatigue philosophy for sailplanes was shown to rest on two principles namely design and testing. It was shown that there currently exist two processes through which the sailplane safe-life can be proved for fatigue compliance: A full-scale ultimate strength test or a full-scale variable amplitude fatigue test. Figure 2-15 displays these two options and the shows where each element fits into the FCP.

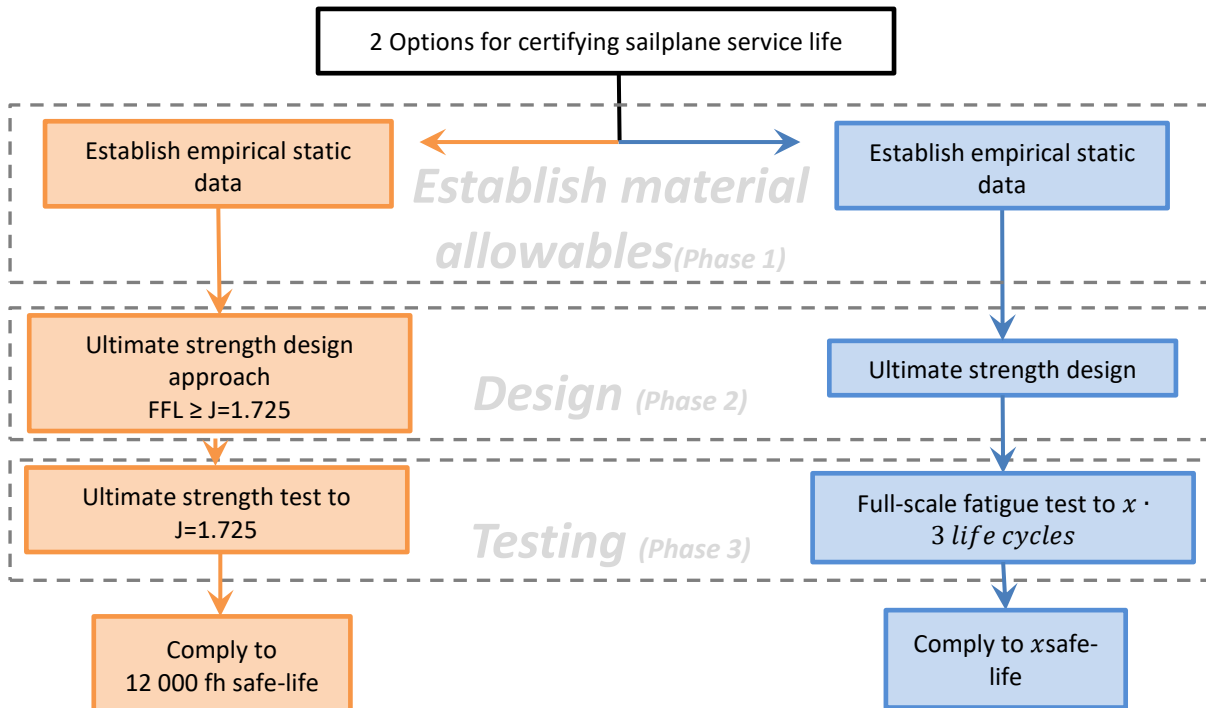


Figure 2-15: Sailplane fatigue workflow

Both these options were shown to be restrictive. Option 1 restricts the possible certifiable safe-life at 12000 fh and Option 2 restricts the certifiable life due to a certification budget which cannot be exceeded. Hence, the research question: Can alternative methods be found? Furthermore, after comparison to other FCP's, it was found that there is some difference between the sailplane and other industries' approach to fatigue certification.

Regarding testing, though some elements of the sailplane FCP shows similarity with that of other industries, the foundation is not clear from literature. This includes the application of ultimate strength approach to $J=1.725$ and application of a Life Factor of 3. According to Whitehead et al. (1986) both these methods should be based on considerable fatigue data from lower level building blocks, but in standard sailplane practice a fatigue database is not required at all.

Consequently, no fatigue design of the sailplane structure is possible, because no data is available to guide any design method or model. Following this, sailplane design is based on static loading alone. Other industries use design practices like safe-life and fail-safe which could also be beneficial for the sailplane industry, but it is currently not used at all.

This deviation from the practices used in other industries could raise suspicion, but since the practices are accepted and proposed by the LBA it is assumed to be founded well on scientific method, though evidence of this could not be found. Consequently, these two processes definitely have merit for praise.

Fatigue certification of an aircraft is a tedious and expensive process, especially if it is attempted to start at coupon level and end with the full-scale specimen. Small industry does not have the financial means to fund such certification programs for each new model of aircraft even if the BBA is used. Light aircraft focusing on the private market shall simply become too expensive. For this problem there are two options; either do not certify your aircraft's service life or find an alternative means of certification.

Finding an alternative means of life certification is exactly what the ANF and LBA have achieved through the Cirrus and Nimbus tests and they should be commended for this. They established an ultimate strength test ($j=1.725$) capable of certifying a considerable safe-life and which is applicable to numerous sailplanes. They were also able to approve a very low life-factor of 3 when testing using a fatigue spectrum.

Were it not for these achievements each sailplane manufacturer would have been required to determine their own ultimate strength load or life-factor through numerous costly fatigue experiments. The general acceptance of these two specifications is a great kindness to the entire industry and is already a great leap in the direction of reducing the fatigue certification cost.

Unfortunately, these two developments are already 20 years old and no further improvements have yet been accepted by the authorities. It has been shown that improvements do not necessarily require innovation, since methods already exist in other industries which are not yet used by the sailplane industry. On the other hand, these methods were developed for a large-scale industry with more financial means. For this reason, innovation to obtain simple and effective fatigue design and testing methods for the smaller industry is also of great importance.

However, it is not implied that no attempts in alternative methods have been researched, but rather that none of the proposed methods has yet gained traction with the authorities.

Innovation on academic grounds is not good enough, the innovation must filter through to the rock bed of legislation or else the application thereof shall never be realised. In the succeeding chapter some of the research attempts in finding fatigue design and testing methods for industrial application shall be reviewed.

2.2. Alternative fatigue methods for sailplanes

After the establishment of the 12000 fh ceiling specifications, a handful of studies into the fatigue of sailplanes have been undertaken. The goal of these studies was to find new certification methods which will enable extending the service life past the 12000 fh ceiling which is cheaper than the current methods.

The initial focus was to eliminate the costliest phase (Phase 3) in the FCP through establishment of life prediction methods which is rooted in phase 1 and phase 2 of the FCP. Relatively good practices were developed, but thus far the authorities could not be convinced to trust a life prediction alone. Since phase 3 could not simply be discarded even though good life predictions were shown, the focus was naturally adjusted to find methods of reducing the cost of phase 3.

This heading is sub divided into two sections. Firstly, the discussion regarding the life prediction attempts and then a discussion on finding more effective methods for phase 3 of the FCP.

2.2.1. Life prediction

Simulation of the fatigue behaviour of a material or component eliminates the iterative process of building and testing of prototypes and thus has the potential of reducing the cost of life substantiation (Vassilopoulos, 2010, p. 13).

Validation of the fatigue life of a metal structure or component can often be done analytically and does not always require testing. When in the composite field this is another case entirely. A great difficulty which today's researchers are still faced with is to develop accurate mechanistic methods to model the damage mechanisms in FRP (Passipoularidis & Brondsted, 2009, p. 6).

Ideally, as in typical metal fatigue evaluation models, one would prefer a practical engineering model which requires minimal material characterization to simulate fatigue damage. Due to correlation discrepancies between macroscopic fatigue performance and micro-structural damage the majority of fatigue models available for FRP's are not mechanistic in nature, but rather based upon the phenomenological³ approach or on macroscopic/empirical approaches (Passipoularidis & Brondsted, 2009, p. 6). The phenomenological approach is based on actual damage measurements while empirical methods, which will be discussed in more detail, are only based on cycles to failure.

³ Find a short discussion on the phenomenological approach in annexure 8

The names of these two prediction methods come from the methods used to quantify the damage, but the model quantifying the damage is only one element of the entire Life Prediction Process (LPP). Nijssen (Nijssen, 2007, p. i) points out that a life prediction consists of four different elements namely:

- A. Cycle counting: Used to build load spectra from a measured load time history.
- B. S-N curve derivation⁴: Used to relate stress or strain to number of cycles to failure.
- C. CLD derivation⁵: Used to supply information of cycles to failure for different stress ratios⁶.
- D. Damage model: Phenomenological or empirical model quantifying damage based on the counted cycles and the CA fatigue curves.

Figure 2-16 illustrates this as well as the logical flow of the LPP.

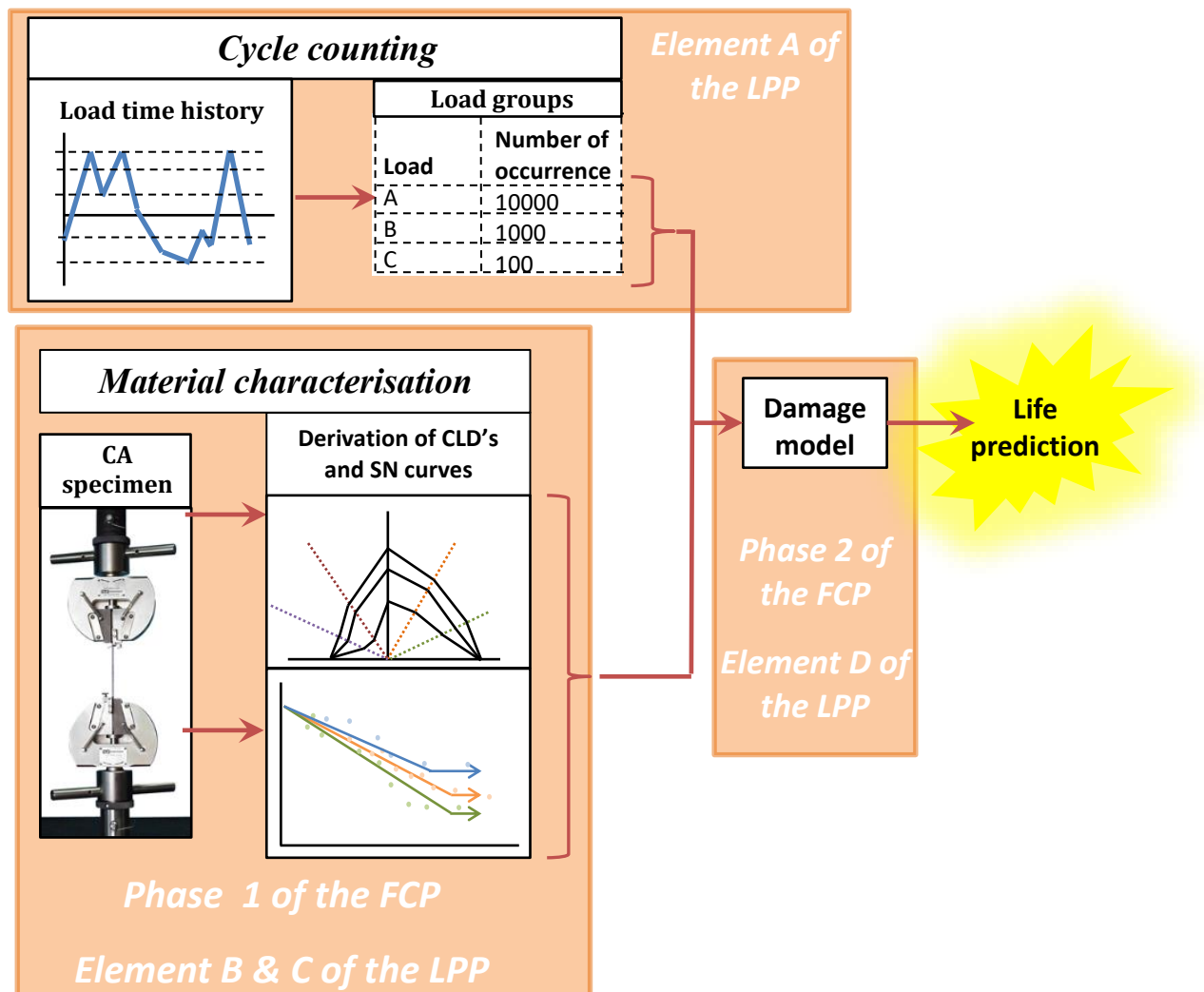


Figure 2-16: Life prediction process (LPP)

⁴ See annexure 5 for a discussion on SN curves and the derivation thereof

⁵ See annexure 5 for a discussion on CLDs and the derivation thereof

⁶ See annexure 5 for a discussion on stress ratio

Passipoularidis (Passipoularidis & Brondsted, 2009, p. 6) defines the empirical approach as models using a damage parameter as an indicator of the amount of damage accumulation until complete failure. This method does not supply any information on the nature of the damage or the effect on material performance (Nijssen, 2007, p. 9).

The most common of these models is the linear Palmgren-Miner (PM) rule which assumes linear accumulation of damage and predicts failure as soon as the damage parameter, referred to as “*D*”, is equal to unity as can be seen by Equation (2).

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

Equation (2) sums the percentage damage accumulated per loading block, with *k* being the amount of loading blocks, each loading block representing a group of constant amplitude cycles at a certain load level, *n* the amount of cycles present in that specific block and *N* the amount of cycles required for failure at that specific load level. This is illustrated in Figure 2-17.

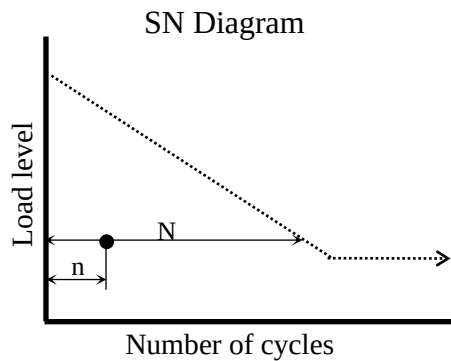


Figure 2-17: Palmgren-Miner method

Regardless of the fact that this PM model is extremely simple, it unfortunately has a few limitations when used for composite materials:

- Nijssen (Nijssen, 2007, p. i) points out that the damage parameter cannot physically quantify the damage accumulation, but only indicate whether failure will occur or not. When the damage parameter is 0.5 it does not necessarily mean that 50% of the life cycle is complete.
- Passipoularidis (Passipoularidis & Brondsted, 2009, p. 18) concludes that the PM model when used in block or spectrum applications may either yield conservative or over optimistic estimations.
- The model does not take into account the effects caused by the sequence of load application (Nijssen, 2007, p. i).

- The model is far from being mechanistic since it can be applied to any material because it is not taking micro damage mechanisms into account (Harris, 1999, p. 136).
- True damage accumulation is not always linear and the interpolation method may be inaccurate. For these reasons there exists an uncertainty in the life estimation when using this method (Hayman, et al., 2008, p. 348).
- This method makes the non-conservative assumption that the strength of the coupon is representative of the full-scale composite structure, while in practice many manufacturing defects and imperfections may be introduced into the component (Hayman, et al., 2008).
- The amount of cycles to failure, N , is found from empirical data, meaning that extensive material testing is required before this method can be applied.

Some of these limitations may be overcome by implementing more complex DAM's which are readily available. However, Rodzewics (Rodzewicz, 2010) points out that the Palmgren-Miner method is by far the most popular. He states that because all the different models yield approximate results there is no point in using a more sophisticated model. For this reason many research efforts have not looked further into better DAMs, but rather attempted to understand the PM method better. These attempts shall now be discussed.

2.2.2. Empirical approach in the literature

As with many research topics in the sailplane industry, the empirical LPP (ELPP) has enjoyed relatively few research efforts as compared to the wind turbine or other related industries. However, the research is clear on the successes, the obstacles faced and the future path to be taken. The research which shall be discussed may be roughly grouped in three categories namely:

- sensitivity of the different elements of the ELPP
- the verification of the ELPP
- using the ELPP to identify fatigue critical components.

2.2.2.1. Sensitivity of the ELPP elements

Echtermeyer (Echtermeyer, et al., 1996) reports on a study done on life prediction of GFRP based on linear damage accumulation. Echtermeyer proposed an empirical method of estimating a modifying factor (γ) for adjusting the damage sum. This means that failure does not occur at $D = 1.0$ but rather at $D = 1.0 \cdot \gamma$.

Experimental results of certain materials showed good correlation with the predictions, while other materials did not. The material which showed poor correlation had a damage accumulation sum which varied between 0.2 and 1.6. Though this shows a large variance, Echtermeyer (1996, p. 912) points out that it might be acceptable from an engineering point of view, since some metals have also shown such big scatter using linear damage accumulation. From the study it can be seen that γ is different for different materials, meaning the damage sum cannot always be simply assumed to be 1.0 as the theory states.

Kensche (Kensche, 2002a) reinforced this finding through an investigation of three different material and layup coupons, but showed that the damage sum has a high dependability on SN diagram estimation as well. The prediction was based on the 95% confidence, 95th percentile SN diagram as well as the mean fatigue diagram. The 95%/95% prediction showed considerable better results than the values estimated from the mean curve.

Table 2-3 shows the experimental damage accumulation sum. $D < 1$ indicating failure earlier than predicted while $D > 1$ indicates that the initial prediction was conservative. The table emphasizes that the damage parameter is a function of material selection as well as SN curve estimation. Following the results, it is proposed to use a damage accumulation sum of $D = 0.1$ when experimental data is too little to make a reliable life prediction.

Table 2-3: Kensche's experiments damage sum
(2002a, p. 50)

		PM damage accumulation sum D	
Survivability		MEAN	95%/95%
Material	Material 1	2.27	0.90
	Material 2	0.25	1.02
	Material 3	1.21	1.39

Krause et al. (Krause, et al., n.d.) conducted a study specifically to measure the influence that each element in the ELPP has on the prediction process. The study focussed on the influence of:

- different rainflow counting algorithms (Methods of classifying load amplitudes and counting the number of load cycles in a specific classification within a load time history or load spectrum.)
- different SN-data fits.
- different CLD formulations.

Four different rainflow counting methods were compared by normalizing the life prediction using the first method as reference. It is shown in Figure 2-18 that at low cycle fatigue, life predictions may differ with a factor of 1.5, just by using a different rainflow method.

The influence of different CLD formulations was determined by comparing a CLD built out of two SN curves with that of a CLD built out of only one (Also called the Goodman formulation). It was shown that the life prediction for the 2-SN curve CLD predicted only one third the life of the Goodman formulation. From this observance a case is made that an increase in the number of SN curves will increase the CLD resolution, increase reliability in interpolation and consequently yield more conservative results.

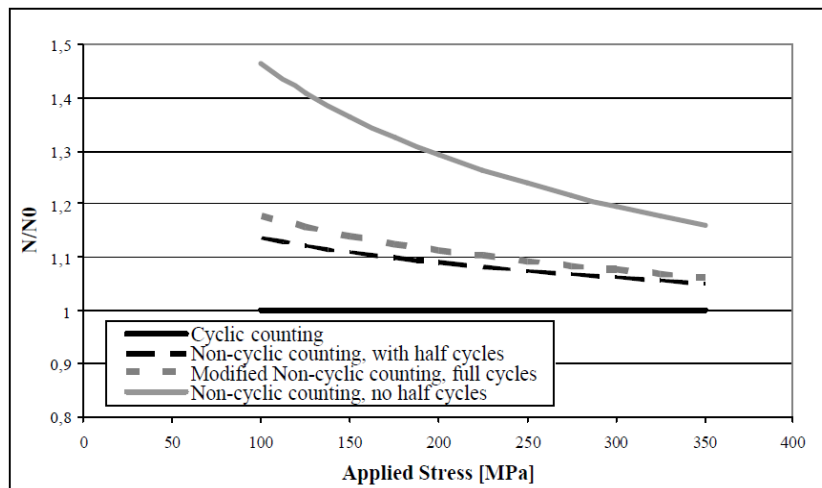


Figure 2-18: Rainflow counting methods' influence on predicted life (Krause, et al., n.d., p. 11)

It was shown that the rainflow counting algorithm and CLD play a decisive role in the life prediction. The influence of these two parameters' variance is much higher than using another DAM or different SN data fits. For this reason, it is very important to declare which algorithms or methods are used during a life prediction (Krause, et al., n.d., p. 11).

Rodzewics proposed an ELPP with a unique perspective on the formulation of the fatigue curves. He identifies that phase 1 of the FCP is an experimental intensive step which requires cost reduction and acceleration and for this reason derived a method of estimating the CLD based on the empirical determination of only four points. He shows that the derived CLD might yield quite different life predictions compared to a CLD derived from more complete data, but when taking FAA life factors into account the life prediction is within the ball park (Rodzewicz, 2009). Figure 2-19 shows his comparative damage summation between the traditional and the proposed CLD formulation, indicated for two different load spectra.

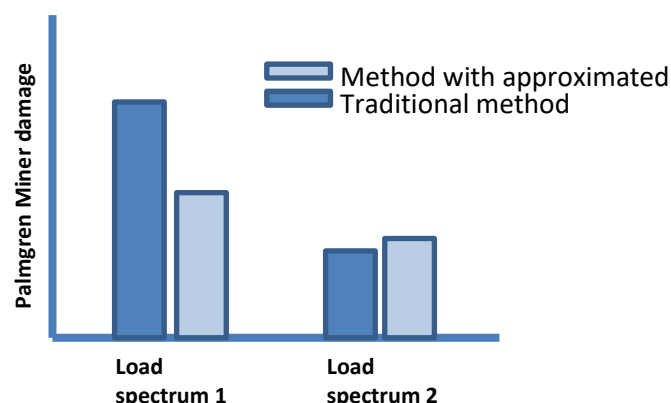


Figure 2-19: Total damage accumulation comparison when using traditional CLD and Rodzewicz's four point derived CLD (Rodzewicz, 2009, p. 11)

2.2.2.2. Verification of the ELPP for sailplanes

Kensche (Kensche, 1995) discusses the possibility of sailplane life prediction based on coupon data and the PM method. Relatively good correlations with service load testing on coupons were achieved. Service loading yielded conservative (they lasted longer than predicted) damage accumulation parameters: $D = 1.14$ and $D = 1.03$ for GL-UP and GL-EP respectively.

This illustrated that the PM method might be adequate for application in sailplane life estimates. The method was then used to predict the life of a glider spar cap. He calculated a safe-life of 5×10^6 flight hours with a design strain of 0.7%, which is equivalent to infinite life.

Kensche (Kensche, 1995, p. 75) points out that due to the complex loading in the wing structure a life prediction cannot be done based on the spar cap alone. The shear web, the bond between spar cap and shear web, areas of steep decrease or increase of stiffness and local instabilities should also be analysed.

In one documented case the ELPP was evaluated by verifying its applicability when the test specimen is scaled up. Rather than using coupons to verify, a beam of typical sailplane type design was considered and the predicted life cycle was about 100% higher than achieved in the test, but because both the achieved and predicted life cycle was equal to infinite life this difference seemed trivial (Kensche, 2002c).

2.2.2.3. Identification of fatigue critical components

Following Kensche's comments on the importance of analysing other structural elements than the spar cap he conducted a study on shear loaded coupons (Kensche, 2002b). Tubular specimens (Figure 2-20) were manufactured with $\pm 45^\circ$ plain weave. These tubes were torsionally loaded to simulate the shear stress acting on the 45° layers in the shear web of a sailplane spar.

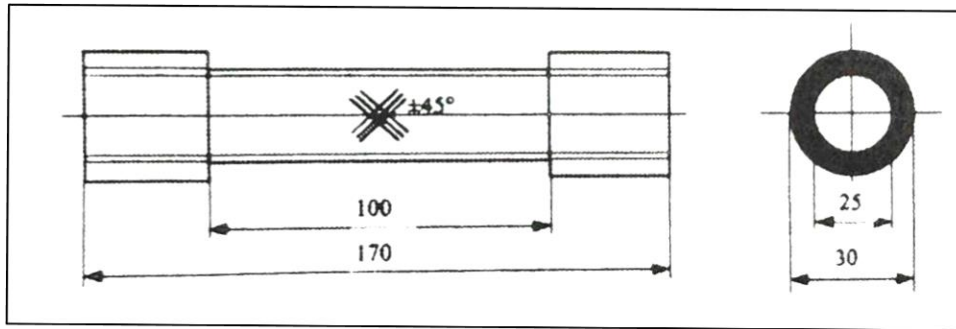


Figure 2-20: Tubular specimens' geometry (Kensche, 2002b, p. 51)

CA fatigue curves were derived and a life prediction was done based on a VA load sequence. A life of 0.45×10^6 was calculated for the shear coupon. Using a similar method of life prediction for a spar cap material it yielded a life estimate of 123×10^6 fh (Kensche, 2002b, pp. 53-54). These values were not validated through VA loading tests as with previous studies, but only predicted since the previous studies already validated the method.

Through these results, Kensche made the case that the spar cap of a sailplane is not fatigue critical (Kensche, 2002b, p. 53) and that future studies on glider spars should be focussed on the shear web because it has significantly lower predicted life than the spar cap (Kensche, 2002b, p. 54) according to the PM method.

Soinne (2015) reports on a fatigue evaluation done on a PIK-20D glider. Various structural details of the sailplane were investigated using the PM method. Failure was assumed at $D=0.1$ and a life factor of 8 (as proposed by FAA (FAA, 1973, p. 8)) was used. The KoSMOS spectrum and representative SN diagrams were used. The spar cap indicated a safe-life of 4.8×10^{20} fh while the shear web showed a safe-life of 9.38×10^5 fh. The results support the predictions of Kenche that showed that the shear-web is more critical than the spar cap.

2.2.2.4. Concluding remarks on ELPP

Krause, Echtermeyer and Kensche showed that the PM life prediction is very sensitive to the methods employed in each step of the ELLP. Kensche shows that good accuracy is possible when the ideal methods are chosen, but this is only substantiated on coupon level. He also showed that upscaling caused very large discrepancies between prediction based on coupon data and the large component test.

Three arguments for consenting with inaccurate predictions were found. Kensche argued: because predictions and the large component test both proved a safe-life equivalent to infinite life in sailplane terms, the inaccuracy becomes of little importance. Rodzewics argues that as long as the life prediction is within the large scatter allowance of an 8 Life Factor proposed by the FAA it is acceptable. Echtermeyer argues that large scatter in life prediction has also been observed for metals as well, and that it might be acceptable from an engineering point of view.

Unfortunately, though a prediction might be shown either to be accurate or that accuracy is insignificant, the authorities have not yet arrived at a point of accepting life prediction alone. This notion is not without merit. Payne (1987, p. 50) has shown in early studies that life estimations using Palmgren-Miner might yield very marginal predictions for GFRP structures. For such reasons validation (Phase 3) of the prediction remains a requirement (Waibel, 2002, p. 58).

Improving the authority's confidence in life prediction might be possible through continuous research efforts on the PM method, but for now the fact remains that life prediction does not assist in reducing the strenuous full-scale test budget. It actually increases it, because it requires a more extensive preliminary material database to formulate the fatigue curves (Phase 1 of FCP). On the positive side it broadens our knowledge on the fatigue behaviour and enables the designer to exploit the material capability even further. It was also shown by Kensche and Soinne that if the ELPP cannot be used for safe-life estimation, it can at least be used to identify fatigue critical elements of the structure.

It was shown that life prediction focus on phase 1 and phase 2 of the FCP and was developed with the hope of eliminating phase 3, which is currently the most expensive phase. This has unfortunately not been achieved yet and for this reason some research into accelerating phase 3 was the logical next step. Three researchers namely Kensche (2002c), Trappe et al. (2008) and Rodzewicz (2009) have attempted this. The research by these authors shall be discussed in more detail in the next paragraph.

2.2.3. Constant amplitude testing

Constant amplitude fatigue testing is a method of decreasing the test time of a fatigue test by increasing the loading amplitude. Similar to the load factor approach except the spectrum is not modified by a load factor, but replaced with an entire new test sequence which causes accelerated damage accumulation.

In the sailplane industry this approach has been documented three times. It was used in preliminary investigations by Kensche (2002c), Trappe et al. (2008) and Rodzewicz (2009) and not for actual certification purposes. Other researchers like Waibel (2002), Echtermeyer et al. (1996) have named the method as a possible alternative to VA testing, but have not published any studies on the subject for sailplanes.

In the wind turbine industry, following from the load enhancement approach, the low amplitude cycles was continuously intensified to reduce testing time. Consequently all the loads in the VA spectrum migrated towards a single load intensity which started to resemble a CA test (Heijdra, et al., 2013). Today many large wind turbine fatigue tests are done by means of CA test programs.

The idea behind the testing method is to replace the VA test which takes an extremely long time with a quick CA test. The major problem with this idea is to relate the two test methods to one another or relate the CA test to actual flight hours.

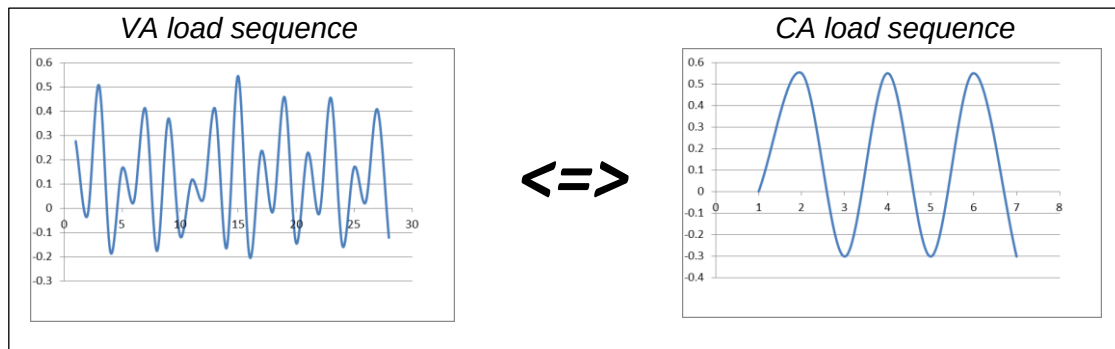


Figure 2-21: VA <=> CA

Kensche (Kensche, 2002c) conducted a study in which he used the PM method to relate constant amplitude cycles to flight hours. This is done by comparing the damage accumulation of the CA test with that of the VA test on the same test specimen or material. Figure 2-22 shows the experimental setup of this investigation.

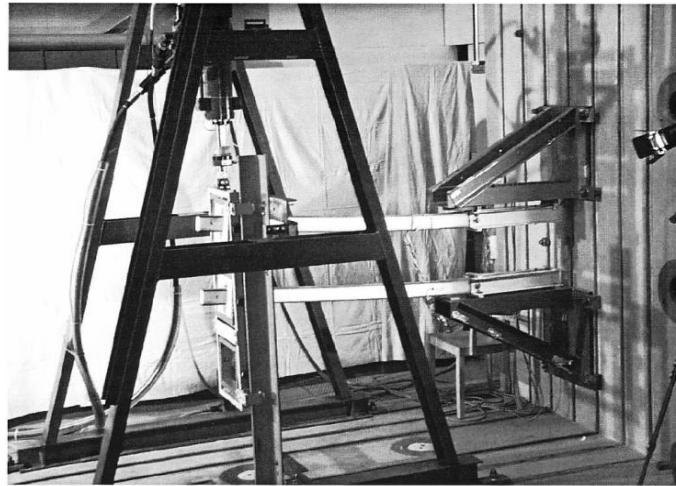


Figure 2-22: Spar test rig used for constant amplitude testing (Kensche, 2002c, p. 36)

The experiment was conducted on large scale structural elements of typical sailplane construction. Two spars were CA tested at a reference strain of 35%. The first signs of damage were in the adhesive between the spar cap and shear web after 1×10^4 cycles. According to PM and material coupon data this is equivalent to 3.9×10^5 flight hours which may be seen as infinite life. Also, two spars were tested by means of the VA KoSMOS spectrum. These spars were tested to a simulated life cycle of 72×10^3 FH and show no reduction in stiffness or residual strength, thus indicating that they may indeed have infinite life.

Trappe et al. (2008, pp. 52-58) conducted a study in which a CA loading program and a VA loading program was compared. The test specimen was a representative wing sub-section (Figure 2-23). The goal of the study was threefold:

1. Determine what typical damage occurs in FRP glider structures loaded under fatigue.
2. Determine whether the kind of damage behavior is similar between a CA test and a VA test.
3. Determine whether the beginning of micro cracking indicates the beginning of total degradation.

The one step test was conducted at limit load with a maximum frequency of 0.08 Hz. Maximum strain was 8% and first signs of damage were observed in the wing skin after 180 load cycles. High shear loading of the skin caused the sandwich core material to fail which lead to buckling of the skin. After repairing the failure and reducing the maximum strain to 6.4% similar damage occurred after 120 load cycles.

The wing section which was loaded with the KoSMOS 2 spectrum showed the same damage mode after 36000 simulated flight hours at a maximum strain of 6.4%. After fatigue testing both wing sections was able to withstand the limit load in bending and torsion in a static test.

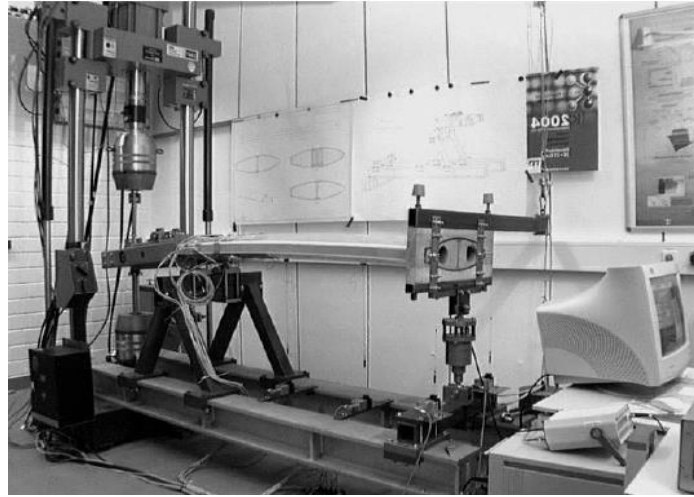


Figure 2-23: Fatigue test setup of a representative glider sub structure (Trappe, et al., 2008, p. 56)

The torsional load was removed and fatigue testing was continued in pure bending. The one-step test was tested to 20×10^3 load cycles while the spectrum test was continued to 42×10^3 simulated flight hours. Unfortunately, no additional comments on the findings of the pure bending tests were made. An observation which is in favour of CA tests, is that the CA test showed identical damage modes as in the VA test. Further it could not be established that micro cracking is a good indication of total laminate degradation.

Rodzewicz (2009, p. 457) used a CA method to investigate a high bearing loaded region in a sailplane spar structure. GFRP and CFRP specimens were tested. He investigated the residual strength of specimen as a function of cycles and amplitude and found a definite fatigue threshold, that when exceeded, caused dramatic residual strength reduction. By means of ultra sound NDI he also determined that for this specific type of specimen the reduction in strength corresponds to severe delamination and established this as the damage and failure criteria.

Rodzewicz (2009, p. 455) also proposed a method of reducing the time and cost of phase 3 of the FCP which is similar to the CA method and is called the equivalent load cycle approach. He proposes reducing the amount of cycles in the spectrum based on the PM damage which each cycle is responsible for. Firstly, eliminating all cycles which cause negligible damage and secondly, finding a single loading amplitude and number of cycles which will cause an equal amount of damage as the sum of all the remaining cycles. Figure 2-24 shows that by following this method the load cycles can be reduced from $1,7 \times 10^6$ to 4882 load cycles for this specific spectrum, but unfortunately any empirical verification of this method has not yet been published.

	Load spectrum – full	Load spectrum reduced
Number of half-cycles	1,753,072	4,882
Equivalent load increment $ \Delta LL $	13	23
Value of damage coefficient D for 1,000 h of operation	0.00389	0.00387

Figure 2-24: Rodzewicz's equivalent load cycles (2009, p. 455)

2.2.3.1. Concluding remarks on CA testing

CA full-scale fatigue testing for certification purposes in the sailplane industry is still in an explorative phase and very little research has been done on the subject. From the literature discussed above one may conclude the following:

- The use of PM in association with the CA test seems like a probable approach for future certification attempts (Kensche, 2002c, p. 42). This follows from the finding where the CA cycles achieved, resembles infinite life according to PM.
- Failure modes are similar when the CA and VA loading programs are compared (Trappe, et al., 2008, pp. 52-58) .
- CA testing shows promise in accelerated fatigue investigation of critical areas (Rodzewicz, 2009, p. 457).

2.3. Conclusion from literature

The literature survey was divided into two sections. Firstly, the current methods of the sailplane FCP and secondly, the new proposals regarding the FCP.

The first section concluded that the two current methods of sailplane fatigue certification are very favourable to the certifier. It was also acknowledged that improvement in the FCP is required if safe-life past 12000 fh is required and that the commercial and military industries have developed methods which could be applied to the sailplane industry, though it will need some customisation.

In the second section the *Empirical Life Prediction* (ELP) and the CA test was discussed as alternative methods for the FCP. The ELP is an innovation in phase 1 and phase 2 of the FCP. Current sailplane design is based on static design alone, but these studies have proven the application of a fatigue design method. Unfortunately, it is at the cost of a more complex phase 1 and without achieving the elimination of phase 3. The CA test approach to full-scale fatigue testing is an attempt to reduce the cost of phase 3 of the FCP and was shown to hold much promise for achieving this goal.

The literature review has thus shown the current accepted methods and possible future methods which could be used for sailplane certification. These methods can be summarised in table format. Table 2-4 can be understood as follows: Each row represents a possible method of sailplane fatigue certification of which the first two are the current accepted methods. The three main columns with headings represent the three phases of the FCP. Each possible method has at least one part of the FCP which is unique.

Table 2-4: Possible Sailplane FCPs

	<i>Phase 1</i>	<i>Phase 2</i>	<i>Phase 3</i>	
No.	Material data acquisition	Design	Substantiation of design	Comments
1	Static coupon tests to determine static design values	Static design with reserve safety factor ($J=1.725$)	Static test to $J=1.725$ at 54°C to validate a safe-life of 12000 fh.	First of two options currently accepted by authorities
2	Static coupon tests to determine static design values	Static design with reserve safety factor	VA spectrum test ($LF=3$) to determine and consequently prove the safe-life	Second of two options currently accepted by authorities
3	CA coupon tests to establish SN and CL diagrams	Empirical life prediction based on VA spectrum and CA data	VA spectrum test to validate life prediction	Kensche has demonstrated this method (2002a)

	<i>Phase 1</i>	<i>Phase 2</i>	<i>Phase 3</i>	
No.	Material data acquisition	Design	Substantiation of design	Comments
4	CA coupon tests to establish SN and CL diagrams	Empirical life prediction based on: VA spectrum and CA data	CA test to validate life prediction (CA cycles related to flight hours through empirical damage calculation)	Kensche has demonstrated this method (2002c)
5	CA coupon tests to establish residual strength or –stiffness data	Phenomenological life prediction based on VA spectrum and coupon data	Na	The method has not enjoyed attention in the sailplane industry
6	CA coupon tests to establish SN and CL diagrams Accelerated 4 point method	Empirical life prediction based on: VA spectrum and CA data	VA test with truncated spectrum and equivalent load cycles calculated from PM equivalent damage	Proposed by Rodzewicz (2009)
7	CA and VA test of various complexity test specimen.	Safe-life or fail-safe design approach	Experimental validation of various complexity test specimen	The building block process

Each of these options is evaluated as possible options for achieving the research objective namely: Finding a cheaper method than the currently accepted method of certifying a sailplane design life greater than 12000 fh.

Method 3: Though the research has contributed to the knowledge of sailplane fatigue design it is actually increasing the budget, because it is adding much more complexity to phase 1 and phase 2 while not changing phase 3. If the Life prediction could be trusted with high confidence it might be possible to eliminate phase 3, but since the authorities are adamant on physical testing this method does not seem to hold water as a cost reducing method.

Method 4: This method adds complexity to phases 1 and 2, but reduces the cost of phase 3 dramatically. For this reason there might be some merit in this method as a possibility for reducing the cost of extended life certification.

Method 5: No examples of this method applied to sailplanes are found in literature, but it is almost equivalent to Option 3 and thus the same comments apply. Not a likely candidate.

Method 6: This method adds complexity to phases 1 and 2, but much less than option 3 and 4. It also employs a unique method of reducing the time of phase 3. For this reason this method is also seen as a possible candidate for reducing the cost of extended life certification.

Method 7: This method is based on the BBA method, which consists of various design and testing phases which amounts to much more work than the current accepted methods. Though there are reliability benefits, the compromise in cost does not justify its use.

Other than the two accepted methods of sailplane fatigue certification five extra methods was found in literature of which only two seem to have potential in answering the research question. These two are method 4 as proposed by Kensche and method 6 as proposed by Rodzewics.

Different than the currently accepted FCP these two methods rely strongly on all three phases. This makes the method much better from a design point of view because it enables the designer to establish structural design life. The currently accepted methods, because it is based on static design, do not have this capability and consequently a design life does not exist.

On the other hand, these new methods do increase the overall complexity of the FCP, but due to some clever innovations the culmination of costs will hopefully be cheaper than a traditional full-scale fatigue test. These innovations are:

- An accelerated empirical method of determining fatigue design curves yielding poorer but acceptable reliability (Rodzewicz, 2009).
- Reducing the amount of cycles in the VA loading program by replacing high frequency cycles with lower frequency cycles which cause similar damage (Rodzewicz, 2009).
- Replacing the VA loading program with a CA loading program by means of equivalent damage calculation (Kensche, 2002).
- Reducing the amount of cycles in the VA loading program through a truncation method (Rodzewicz, 2009).

CHAPTER 3 METHODOLOGY

In the introduction of this study the goal of finding and validating an alternative method of life extension was established. Literature has shown that the constant amplitude test may be a viable alternative to spectrum testing. In this chapter a method of finding the equivalent number constant amplitude cycles is proposed. An empirical method of verifying the applicability of this method is also suggested.

3.1. Rationale for achieving research objective

The literature review has given an overview of general fatigue certification procedures and more specific procedures applicable to the sailplane industry. The FCP consisting out of three phases was derived from these findings. It was shown that improvements in each phase of this process may have an influence on the cost and simplicity of life certification, but that phase 3 may have the largest influence and that current research efforts are focussed on this phase.

The work of Kensche and Rodzewics on accelerated phase 1 and phase 3 methods was identified as methods worth pursuing. Both suggest the replacement of the VA loading program with equivalent high amplitude cycles to accelerate the damage accumulation during phase 3 test specimen validation. Both suggest that a single amplitude loading program, named the Constant Amplitude (CA) Test, be used.

These authors relate the CA test to actual service life by comparing the equivalent PM damage caused by a VA load program of which the corresponding service life is known. This method is dependent on a large quantity of empirical data (Phase 1), because the PM comparison (Phase 2) requires accurate fatigue design curves. Regarding the research objective this dependence on phase 1 and phase 2 is not ideal; even though phase 3 is accelerated and shall consequently result in cost saving, an increase in phase 1 and phase 2 efforts shall have the opposite effect.

For this reason these two phases are identified as areas which require improvement. Furthermore, since the CA test could be seen as the most severe application of a LEF (Heijdra, et al., 2013) additional improvement in phase 3 could not be imagined. Accelerating or completely eliminating phase 1 in the FCP for CA testing is identified as an objective worth pursuing as an attempt in achieving the governing research objective of accelerating the entire FCP for sailplanes. A method for accelerating phase one shall be discussed systematically under the next sub heading.

3.2. Proposal and hypothesis

A new empirical method of determining the relationship between a VA and CA sequence without the use of SN-, CL diagrams and a PM calculation is proposed. In this method the number of CA cycles equivalent to a certain VA sequence is determined through experiments on coupon and structural element level. Once the CA equivalent cycles are established with adequate confidence, a full-scale specimen can be tested with this amount of CA cycles for fatigue substantiation.

The method is grounded in the PM assumption of linear damage accumulation irrespective of the loading sequence. Consequently, it was assumed that if two different types of load

sequences both cause failure of two respective but identical specimens these two load sequences could be seen as equivalent. Since damage accumulation is linear the relationship between these two load sequences is then established.

In other words: Specimen 1 is cycled with load sequence A until failure. Specimen 2, which is identical to specimen 1, is cycled with load sequence B until failure. In both cases the number of cycles is counted. Since both have had the same effect on the specimen, namely failure, they are deemed equivalent as shown in Equation (3).

$$xA = yB \quad (3)$$

If PM is valid then the relationship between the two load sequence types is established and could be presented graphically as in Figure 3-1 and written as

$$A = \frac{y}{x} \cdot B \quad (4)$$

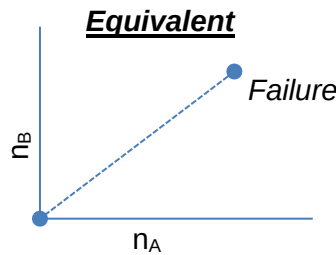


Figure 3-1: Equivalent cycles graph

Through this method the relationship between two load sequences could be determined by finding only one point on the graph in Figure 3-1. Compared to the numerous data points required to draw a CLD this is a considerable improvement to the traditional method of relating two load sequences using equivalent Palmgren-Miner damage calculation.

One disadvantage is that cycling to failure could still be quite a long process, but this can be overcome with a slightly different approach. The relationship can also be determined without testing both load sequences to failure.

The process is as follows:

Choose the load sequence which shall cause failure in the least amount of time (load sequence B) and run a specimen till failure. yB is now determined. Cycle a new specimen with sequence A until a predetermined amount of cycles (p). Stop the test and start applying load sequence B until failure (failure occurs in q number of cycles). Both tests achieved equal results on separate specimens, namely failure and on these grounds are named equivalent.

$$\text{Test 1} = \text{Test 2}$$

$$yB = pA + qB$$

$$B - qB = pA$$

$$(y - q) \cdot B = pA$$

$$A = \frac{(y - q)}{p} \cdot B \quad (5)$$

And since PM is assumed valid, Equation (5) is equivalent to Equation (4)

The experimental procedure for finding this relationship can also be illustrated graphically in Figure 3-2. On this graph the straight line with negative slope is the failure line. The y-axis shows the number of cycles of sequence type B and the x-axis the sequence type A. As soon as the summation of any combination of the two load sequences cross the failure line, failure should occur.

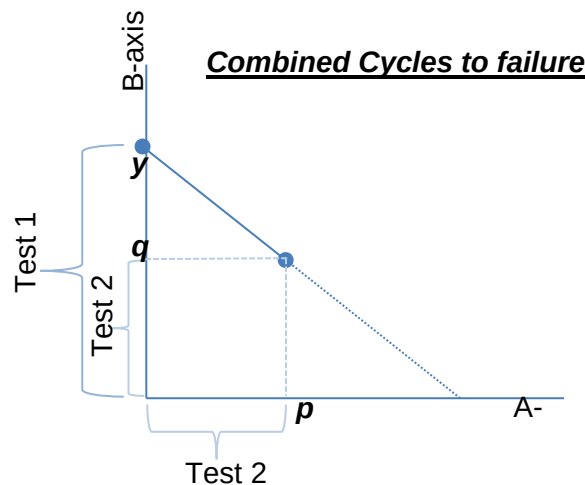


Figure 3-2: Combined cycle to failure graph

Unfortunately PM is not always valid, meaning the linear relationship shown in Equation (5) and Figure 3-2 might not truly exist. For this reason it was decided to evaluate the validity of PM as a probing investigation into the feasibility of this method. The hypothesis which required evaluation was thus: When plotting the “combined cycles to failure graph” the PM assumption is valid for engineering application and consequently the failure line can be derived from any two points as shown in Figure 3-2.

In order to evaluate this method it was decided to find various points on the Combined Cycles to Failure graph (Figure 3-2) by means of experiment. This would then give a clear indication if the hypothesis is acceptable or not.

3.3. An analogy as verification of proposal

The rationale and hypothesis could also be understood if observed through the perspective of an analogy. In this analogy it is assumed that different types of sequence have the same effect on damage as different type of substances has on filling a certain volume. In this analogy visualize the parameters as follow:

Table 3-1: Analogy of water and oil. Allegorical equivalent

	Unit of measure		Allegorical equivalent	Unit of measure
CA sequence	[No of cycles]	→	Water	[kg]
VA sequence	[No of cycles]	→	Oil	[kg]
Specimen damage capacity	[Fraction of 1]	→	Volumetric capacity of a container	[kg/m ³]

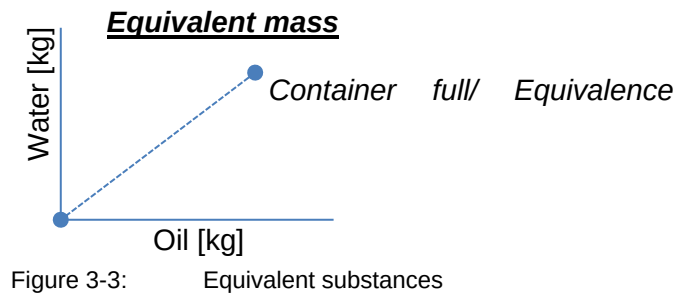
Assume the following:

- Just as equivalent cycles is required at various stages in the damage progress, equivalent mass of the two substances is required at various stages in volume progress until full capacity.
- Volume, similar to damage, is not directly measurable.
- Knowing the volume of a substance is only possible at full container capacity. Any volume smaller than full capacity cannot be known. This is similar to damage. The amount of damage can only be known at failure (D=1) and prior to any cycle application (D=0).
- The total volume of the container is known, just as damage at failure (full capacity) is known.
- Measurement of mass is possible, just as the number of cycles.

Method 1

If the container is filled with water and oil respectively and in both cases the mass is measured then equivalent mass for a certain volume was established. For this case we know that the mass and volume is directly proportional (Equation (6)) and thus if the volume is halved, mass is halved as well. Thus the equivalence between the two substances is linear in nature and can be displayed similarly to Figure 3-1 in Figure 3-3

$$V = \frac{m}{\rho} \quad (6)$$



In the actual case the problem exists that going to full capacity with the VA sequence might be impractical due to long testing time. Similarly, it could be said that we do not have enough oil to fill the container to full capacity and since volume can only be known at full capacity another method is required.

Method 2

A method of combining the two substances, so that the cumulative volume is full capacity, is proposed. In this proposal two points are required. Firstly, the mass of water required to fill the container to full capacity (Point A) is measured. Secondly combining the oil and water required to fill the container and measuring the mass of both substances (Point B). The results can now be displayed in table or graphical form (Table 3-2).

Table 3-2: Analogy of water and oil. Combination of two substances

Point	Water (y-coordinate) [kg]	Oil (x-coordinate) [kg]
A	5	0
B	4.5	1

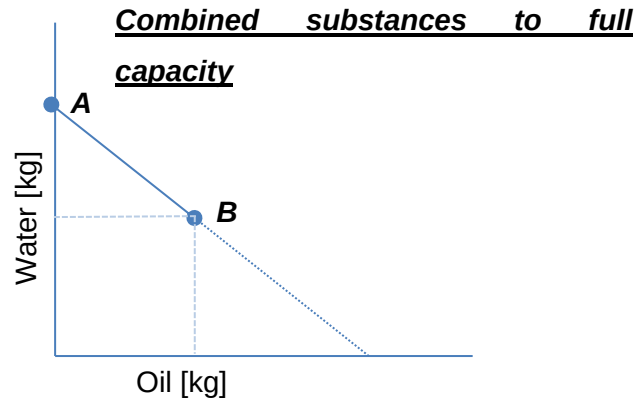


Figure 3-4: Combined substances plot

To find the equivalent mass from this experiment is trivial. Simply ask the question: What is the difference between point A and B? Or in other words, considering point B, how much water was replaced by the oil? This is found by subtracting the amount of water of point B from the amount of water in point A. Mathematically it is written as follows

$$A_{water} - B_{water} = B_{oil} \quad (7)$$

$$(5 - 4.5)_{water} = (1)_{oil}$$

$$\begin{aligned} &0.5 \text{ kg water} \\ &= 1 \text{ kg oil in terms of volume} \end{aligned}$$

We know for the case of combining volume that the equivalence is linear, but in the case of damage we do not. For this study it is proposed to evaluate if this linearity exists by varying the combined substance method. In other words, repeat the second method with different masses of oil (See Table 3-3). By doing this the linearity of the equivalence can be evaluated.

Table 3-3: Combined substances method

Point	Water (y-coordinate) [kg]	Oil (x-coordinate) [kg]
A	5	0
B	??	??
C	??	??
D	4.5	1

3.4. Experiment

In order to evaluate the hypothesis, it is required to plot a Combined Cycles to failure graph as shown in Figure 3-5. This was done by establishing three groups of data in the experimental space. If a linear regression through the data could be shown as an adequate fit, the hypothesis would have had grounds to be accepted.

The three points were found by applying three different types of load sequences to the three respective groups of equivalent specimens. The first group was cycled using a VA sequence namely the sailplane standard KoSMOS II spectrum. Reference force was chosen as $0.75 \cdot U_F$. The second group was cycled with a CA sequence in which the amplitude was equal to the reference force. The third group was cycled using the combined approach. The VA sequence was run to 50% of the number of cycles, which caused failure in the first group data; after which it was run to failure using the CA sequence. This procedure is also referred to as the multiple load sequence (MLS) procedure.

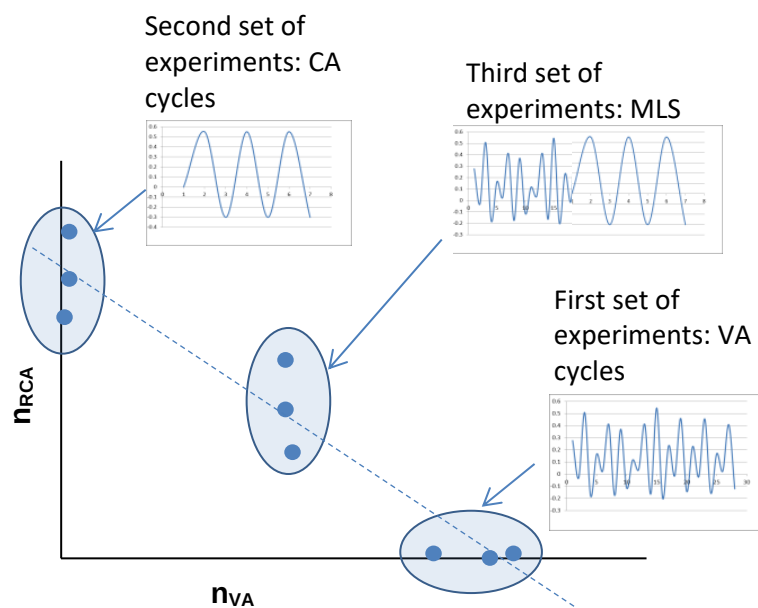


Figure 3-5: Three data sets on the combined cycles to failure graph

The VA sequence was the sailplane standard KoSMOS II spectrum. The amplitude of the CA loading pattern was equal to the largest cycle in the KoSMOS II spectrum. The stress ratio, frequency and wave form are identical in all three the loading sequences.

- Stress ratio = -0.55 which is the loading ratio of the VA spectrum KoSMOSII.

- Frequency = 6 Hz which is the maximum frequency that did not cause excess temperature rise and was not too fast for the servo hydraulic compensation to yield a good response.
- Wave form was chosen to be sinusoidal because it is the most representative of actual loading (Vassilopoulos, 2010, p. 8).
- Maximum amplitude = 75% Ultimate static loading capacity.

No special requirement was decided upon for the ambient conditions during testing and for this reason temperature and humidity was not controlled nor measured.

3.4.1. Test specimen

The hypothesis may be considered using any one of many different specimen designs, but because the study is directed to the sailplane industry it was decided to keep the sample representative of the application.

Literature has shown that the shearweb of a FRP sailplane spar is much more susceptible to fatigue damage than the spar cap. (Kensche, 2002b, p. 54). It is a better assumption to choose the shearweb, rather than the spar cap, as the component which determines the fatigue life.

This finding prompted an investigation into the exact stress state, which the shear web experiences and how to mimic this stress state in a simple test specimen. The test specimen had to be simple in design, so that fabrication repeats would not introduce too much variability. An additional constraint was that the specimen had to be tested in a single actuator servo hydraulic test bench, since it was the only available apparatus which could apply the VA loading pattern. This meant that the load application had to be in line with actuator displacement so that no moment could be transferred to the clamps.

The complexity of the stress state in the shearweb⁷, drove the decision to design a miniature beam, similar in design features to an actual glider spar and loaded in its centre with two simple supports on either side. This was then a very close representation of the stress state found in a glider spar and the loading configuration would also make it possible for a single actuator servo hydraulic machine to load the specimen.

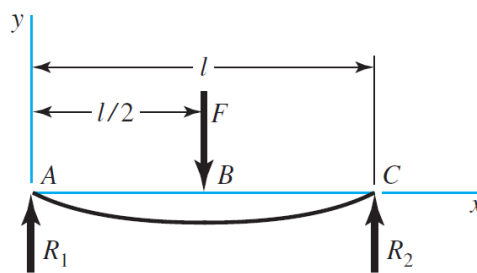


Figure 3-6: Simple supported beam, centre load
(Budynas & Nisbett, 2008, p. 1023)

The test loading sequence would induce tension and compression, and for this reason the points A, B and C had to be pin connections, meaning the specimen would have three holes as load introduction points. The load introduction points are usually critical for bearing failure and for this reason the beam required some local reinforcements in these regions. Due to the

⁷ See annexure 1 for discussion on a sailplane spar stresses

bending of the beam there would be relative rotation between the beam and the pin connections and for this reason the interface required brass bushes.

The nature of this type of loading generates a constant shear force between A and B and between B and C, both these forces being equal to one another, but opposite in direction. See Figure 3-7. Since the shear loading on the shear web is investigated, these two regions were designed to be the most critical to the specimen.

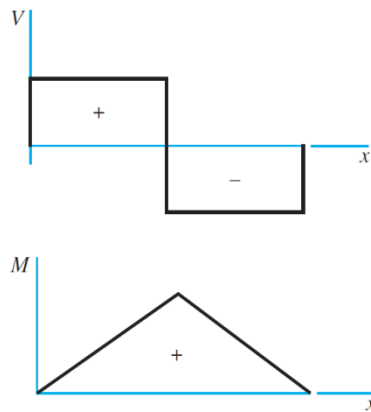


Figure 3-7: Shear force and bending moment distribution of specimen
(Budynas & Nisbett, 2008, p. 1023)

The bending moment, as shown in Figure 3-7, which is its largest at the centre, should not cause the failure and for this reason an over designed spar cap manufactured of a stiffer material than shear web was required.

3.4.1.1. Design specifications

Following from the discussion above the design specification can be summarised as follows.

Geometric specifications

1. Test specimen must be representative of a glider's primary wing structure. In other words, the spar must consist of carbon roving spar caps and a bi-directional glass fibre shear web.
2. Load introduction points are to be reinforced to ensure failure does not occur in these positions.

Loading specification

1. Limit load of the gauge region must be chosen on the basis of fatigue experiments conducted on the 0/90 coupons (Boshoff, 2014), which is a very similar laminate, as the one used in the shear web. It was decided to choose the limit load, which corresponds to a shearweb stress of 204 MPa, since such a stress shall cause rapid damage accumulation.
2. Spar cap of the specimen must be designed to have a high safety factor to ensure failure cannot occur there. It was decided that the LBA proposed ultimate stress of 400 MPa should be used.
3. Stress in all other lamina and bonds are to be kept to a minimum with a safety factor of at least 2.
4. Buckling of the spar caps must be resisted either intrinsically by the specimen design, or by means of a jig.

Fatigue and damage accumulation specifications

1. The gauge region must be designed to ensure relatively quick damage accumulation. 1×10^3 limit load cycles were decided to be a practical amount of cycles. Based on fatigue coupon data this corresponds to a limit load in which the stress in the material was 204 MPa.
2. A certain region of the shear web called the gauge region of the specimen must be the critical loaded region to ensure that damage accumulation mainly takes place there.

Material specifications

1. Shear web must be manufactured of a plain weave glass fibre fabric to assist in accelerated damage growth.

2. Spar cap requires unidirectional carbon fibre.

3.4.1.2. Design procedure

Based on the aforementioned specifications a conceptual geometry as shown in Figure 3-8 was proposed: A plate type specimen with spar caps spaced apart and sandwiched by glass fibre on either side

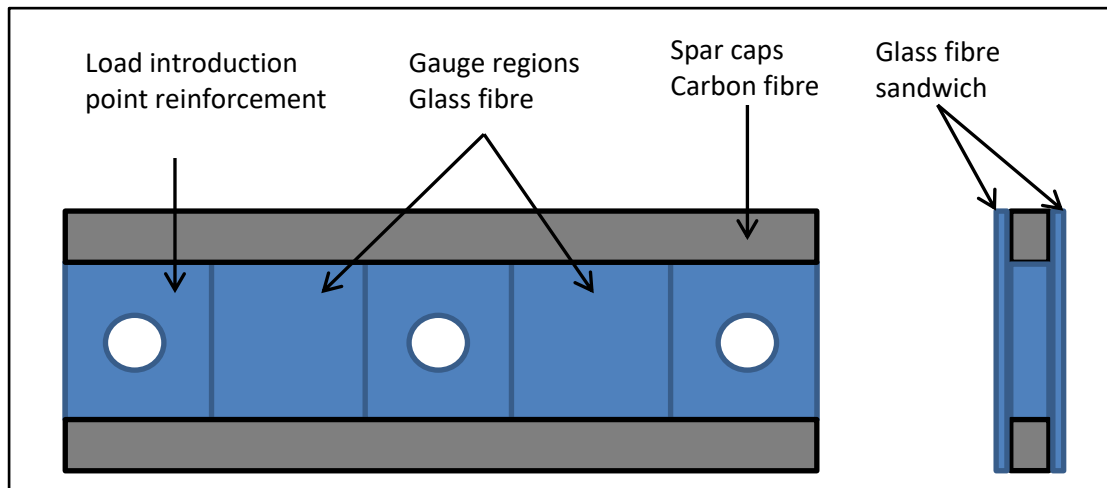


Figure 3-8: Concept of test specimen

The limit load (defined as the maximum load applied to the specimen during the fatigue program) should be chosen in such a way as to ensure failure of the specimen does not take excessively long. The chosen limit load was based on observations from previous fatigue experiments conducted on simple tensile specimens. Though these tensile specimens only experience uni-axial loading and not bi-axial loading as the shear web material does, it was the only point of reference available and thus considered adequate. The data indicated that if the limit load is chosen as 75% of the ultimate load, failure should occur within a reasonable time frame (Boshoff, 2014).

The design of the specimen was achieved by means of a two-step process, which could be iterative if required. The first step was to design the specimen according to the initial specifications using NX Nastran finite element analysis. Due to the fact that the CAD and FEA environment is directly connected and that the FEA consists out of 2D shell elements only, virtual design iterations have very short cycle times. This made manual optimisation easy enough.

The second step in the process was to manufacture a specimen according to the FEA design and conduct a static test on the specimen to verify the design. If variation from the design existed, it would be investigated and the specimen would be redesigned with calibrated FEA parameters.

The FEA assumptions were as follows (Also refer to Figure 3-9):

- A simple 2D shell mesh with QUAD elements 1.5mm in size was used.
- Point A and C had only y-rotation degrees of freedom.
- Point B was free in y rotation and z translation.
- The force was applied at point B.
- Point A, B and C were connected to the mesh by means of rigid elements as can be seen in the Figure 3-9.

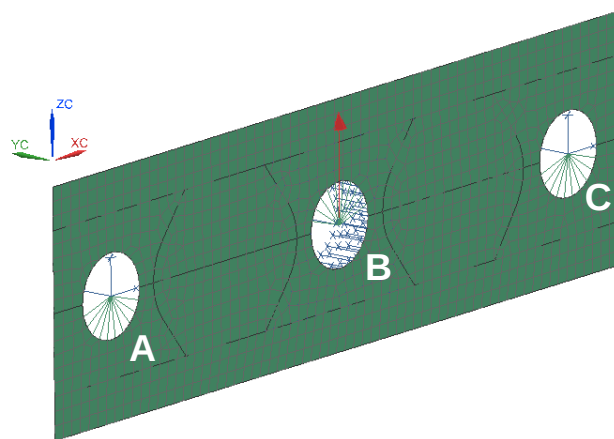


Figure 3-9: FEM assumptions

The first specimen was designed with the following specification.

Table 3-4: First specimen specification

Specimen thickness	3.6mm
Shear web	2 layers, 195gsm, E –glass, plain weave placed at 45° on either side
	2 plies of 1.5mm foam to space the shear web
Spar caps	3mm thick carbon UD laminate (ATG composite 343gsm 6K)
Spar cap width	8mm
Load introduction reinforcement	3mm thick glass plates
Shear web design FVF	37%
Spar cap design FVF	45%
Ultimate load (Load when shear web failure was expected)	4600N

Verification of the design yields very poor correlation. The specimen was loaded to double the expected load of failure; after which further loading was abandoned because the pins of the

jig would have been over loaded. Investigation of this variation yielded the following observations

- The fibre volume fraction (FVF) achieved during manufacturing was much higher than expected. This had a considerable effect on the strength of the spar cap. The design FVF was 0.45 and the achieved FVF was 0.6 for the spar cap. This computed to a 30% increase in fibre volume, which increased the strength considerably.
- It was observed that even if the adjustment in FVF was made, the stress in the top spar cap was still calculated higher than the expected stress at spar cap failure. From this observation it was decided that excessive spar cap stress is acceptable, because it shall not amount to specimen failure and the shear web is still the most critical component.

When the FVF was adjusted the theoretical and experimental values correlated well with the first design. The second specimen was designed with the calibrated method and consequently yielded the specimen design which was used for the experiments.

Table 3-5: Second specimen specification

Specimen thickness	3.3mm
Shear web	1 layer 195gsm, E-glass, plain weave, at 45° on either side
Gauge area	2 plies of 1.5mm foam to space the shear web
Spar caps	3mm thick. 1.5mm PVC foam sandwiched between two UD carbon laminates (ATG composite 343gsm 6K)
Spar cap width	5mm
Load introduction reinforcement	3mm thick GFRP plates
Shear web design FVF	51%
Spar cap design FVF	60%
Ultimate load (Load when shear web failure was expected)	4700N
Shear web ultimate strain	0.5%
Limit load	3525N
Shear web limit strain	0.375%

The finite element analysis result of the second specimen is shown in Figure 3-10. The colour spectrum indicates the failure indices according to the maximum strain failure criterion. It can be observed that the failure is most probable to take place in the gauge area. The compression spar cap is also highly loaded but it was shown through experiment that this high stress does not amount to failure and that the actual stress capacity for the spar cap is higher than predicted.

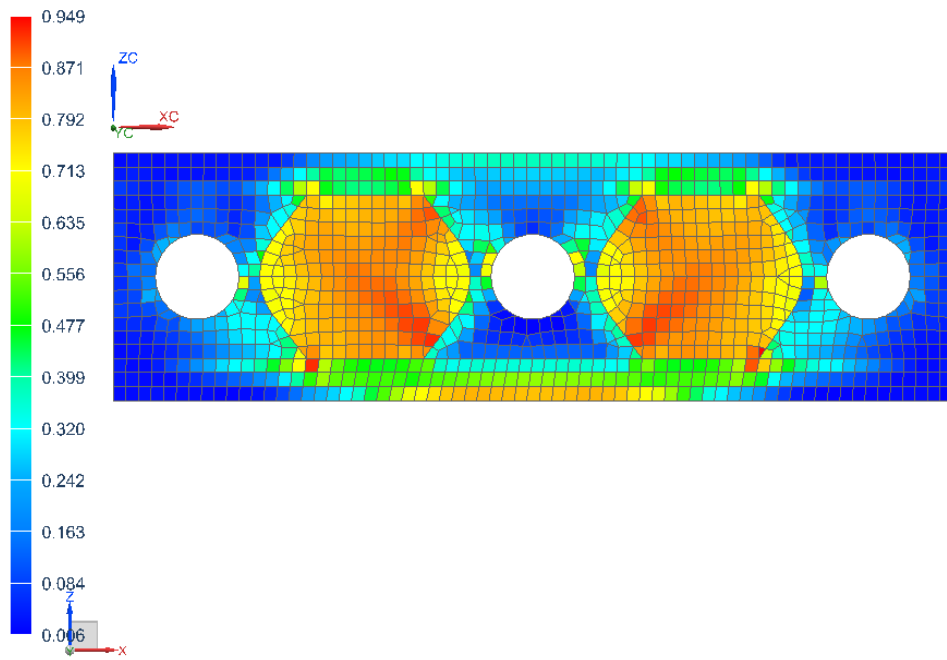


Figure 3-10: FEM results ultimate load

3.4.1.3. Test specimen description and manufacturing

The test specimen was designed to be a 3.3mm thick plate configuration with a length of 100 mm and height of 29mm. Brass bushes were installed 40mm apart, lengthwise and on the centre line of the specimen. Refer to Figure 3-11. The specimen consisted of 6 different components as described in Table 3-6.

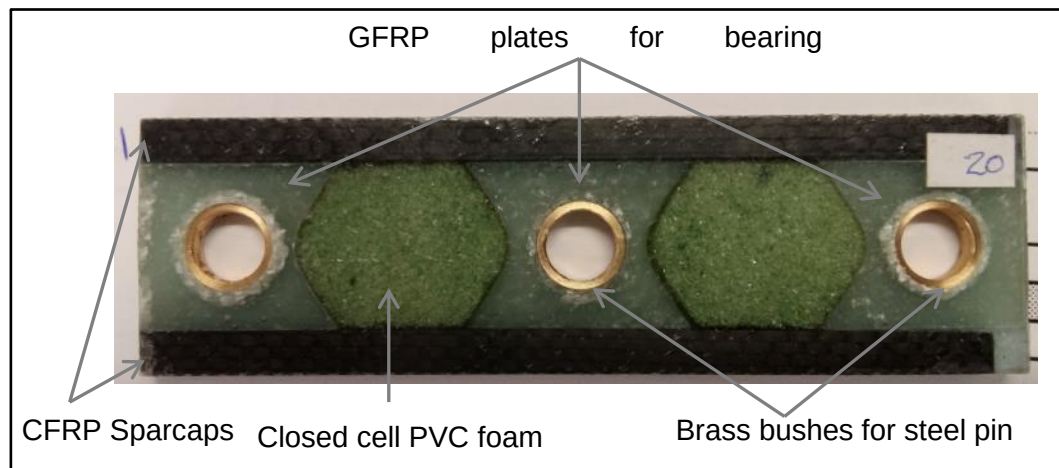


Figure 3-11: Test specimen

Table 3-6: Test specimen description

ID	Part	Material	Function
1	Load introduction point reinforcement side plates	Pre-manufactured GFRP plates Glass fibre: 280gsm twill weave E-glass Epoxy: MGS LR 285	These components ensure that there is enough material area at the load introduction points in order to ensure that these regions are not overly stressed.
2	Load introductions point reinforcement centre plate		
3	Spar caps	Pre-manufactured UD CFRP with sandwiched closed cell PVC foam. Carbon Fibre: 343gsm UD 6K carbon Epoxy: MGS LR 285 Foam: 1.5mm PVC	Component responsible for carrying normal stresses induced by bending.
4	Gauge area foam inserts	Foam: 1.5mm PVC foam	Generates part thickness and ensures the shear web layer is placed on an even surface
5	Shear web	195gsm, E-glass, plain weave, at 45° on either side	Shear load carrying element of beam and component under consideration
6	Bushes	Brass	Bearing area for pin connection to ensure close sliding fit a free rotation

Pre-manufactured 5mm broad carbon spar caps spaced by 3x19mm foam, all sandwiched between two layers of glass fibre, with pre-manufactured GFRP plates at the load introduction points, and brass bushes at the steel contact points. Figure 3-12 shows an exploded view of all the components of the specimen.

Spar caps were a sandwich structure with 343 gsm Carbon UD material enclosing 1.5mm Airex closed cell PVC foam. The glass plate reinforcements around the holes, were pre-manufactured using a balanced layup of 280 gsm twill weave E-glass. Brass bushes were bonded into the structure. The glass fibre which sandwiched the entire structure was a single plain weave 195 gsm lamina on either side.

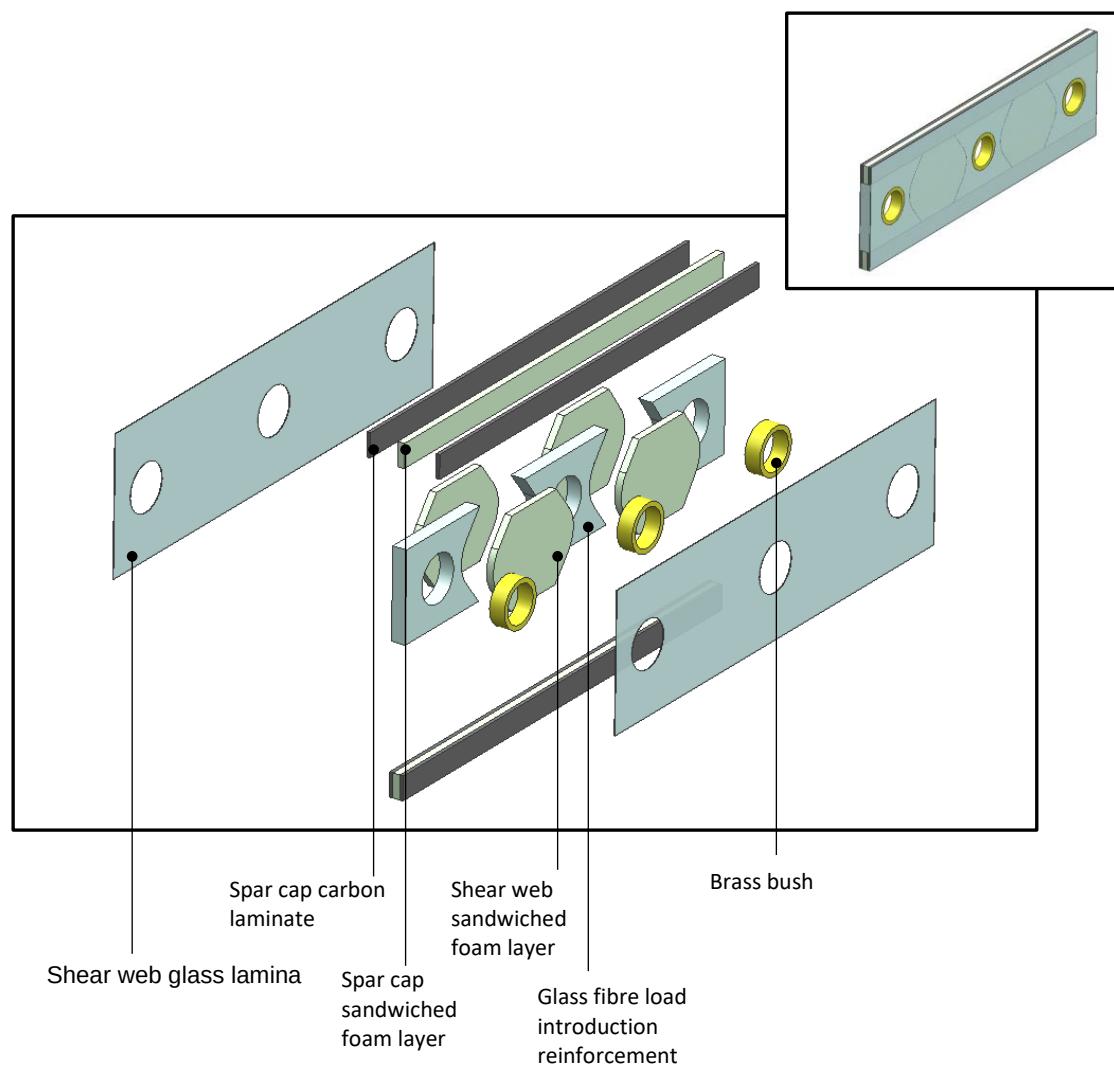


Figure 3-12: Exploded view of specimen

3.4.2. Apparatus

3.4.2.1. Servo hydraulic test bench

The complex loading pattern which is required to be applied to the specimen was achieved by means of a 100kN MTS Landmark Servo Hydraulic Test Bench (SHTB). The centre load and the simple supports which the specimen required could not be achieved by this test bench alone because it only has two hydraulic clamps to clamp conventional tensile test specimens.



Figure 3-13: MTS 100kN servo hydraulic test bench

3.4.2.2. Intermediary jig

The design of an intermediary jig between the specimen and the SHTB was required. The jig was designed according to the following specification:

1. The jig must have the proper interface to connect to the SHTB clamps as well as the test specimen.
2. The jig should be designed for infinite fatigue life, with reference to the maximum load of the specimen.
3. The jig must allow adequate room for the test specimen to deflect naturally.
4. The jig must keep the test specimen from buckling.
5. The jig must generate the stress state, and bending form in the test specimen which is equivalent to a theoretical simply supported centre loaded beam.
6. The jig must be stiff enough to cause a negligible error on the displacement measurement of the LVDT.

The jig was designed as tongue and fork configuration. The tongue was connected to the lower SHTB clamp and the centre specimen hole, while the fork connected to the upper clamp and the two side holes of the specimen. Two I-shaped plates was also part of the jig and placed on either side of the specimen to resist spar cap buckling. Figure 3-14 shows the physical setup and Figure 3-15 the exploded view of all the components.

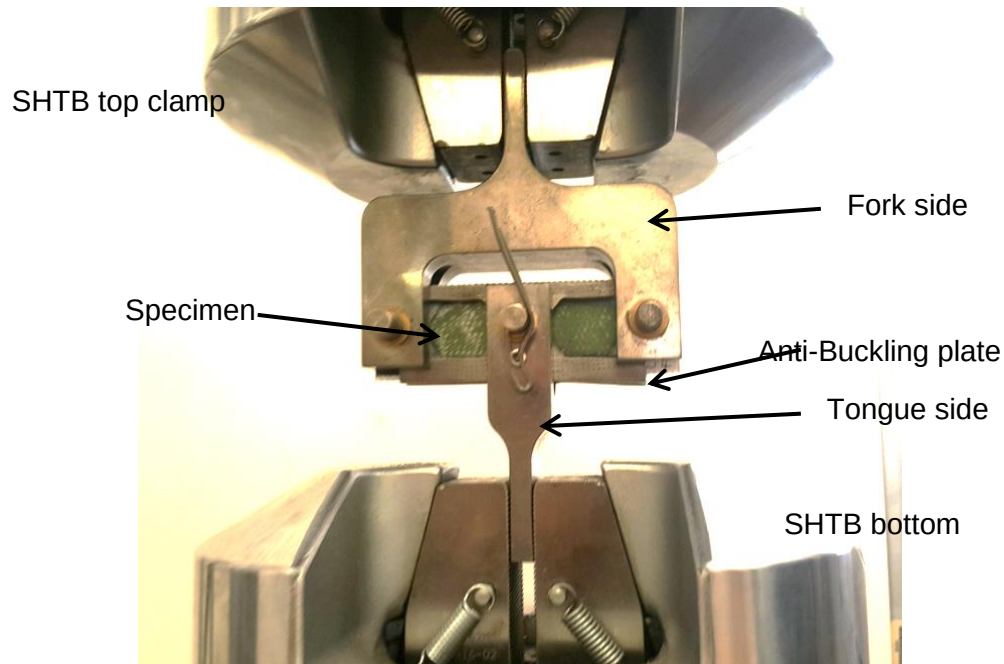


Figure 3-14: Intermediary jig connected to sample and test bench clamps

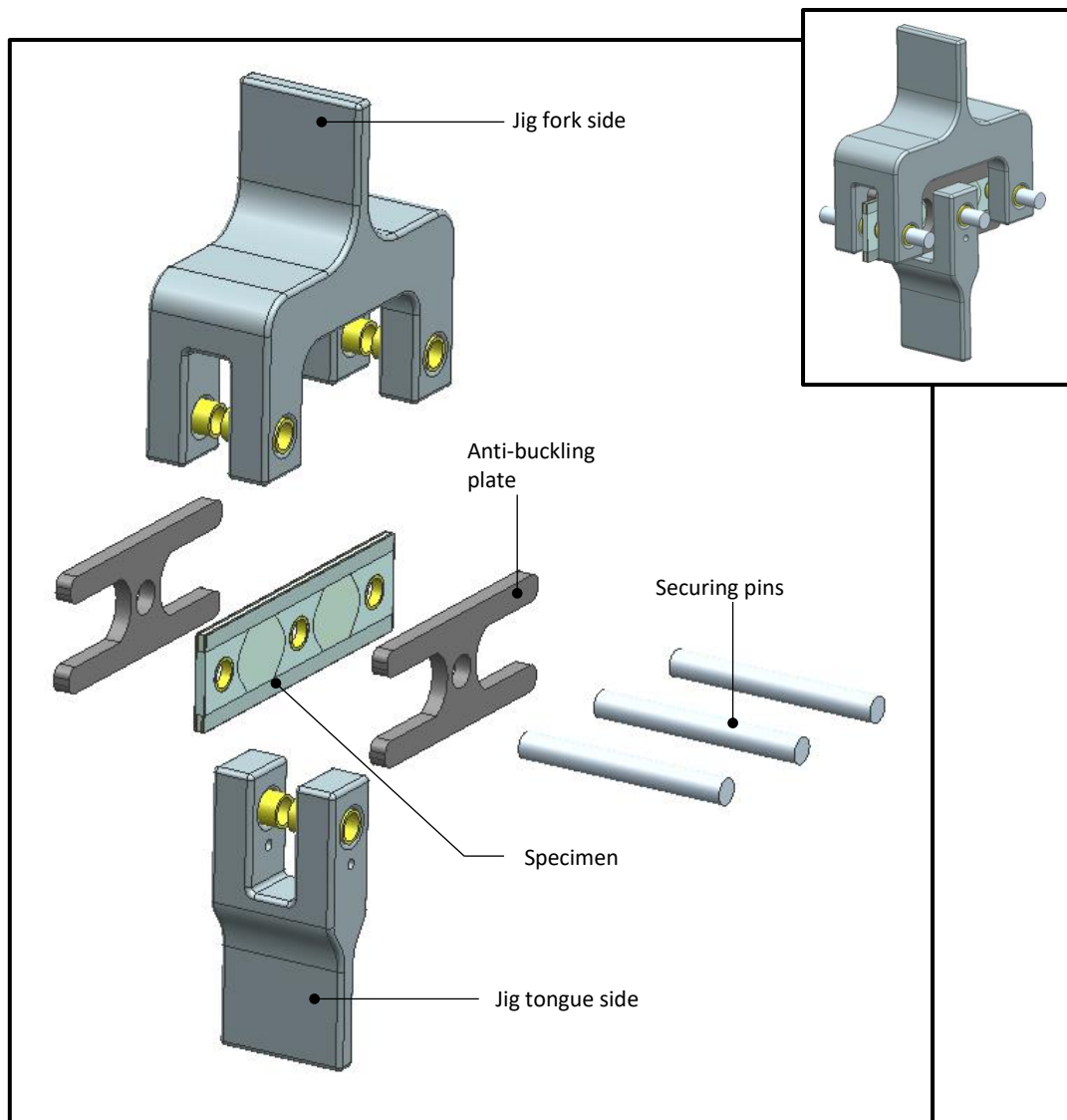


Figure 3-15: Experiment jig

CHAPTER 4 EXPERIMENTAL RESULTS

In the previous chapter a method for finding equivalent constant amplitude cycles as well as an experiment for evaluating the method was proposed. The experiment was done and the results thereof are discussed in this chapter.

4.1. Failure mode

The test specimen was designed to be critical in the shear web. This meant that failure of the shear web would be considered an acceptable failure. The problem of this criterion was that it was very broad and not specific enough. Uncertainty existed regarding the mode of failure and whether it would be possible to discern between a highly damaged and totally failed specimen. Preliminary testing was required to narrow the failure criterion to a more specific understanding.

Preliminary testing was very valuable in gaining a deeper understanding of how the shear web fails and consequently helped to define the failure criteria. Failure was observed in five distinct stages. The five damage stages were then decided to be the prerequisite for a valid failure. Serious deviation from these five stages deemed a specimen invalid. The five stages are shown in Figure 4-1.

The five stages are:

1. Initial whitening of the gauge area. In this stage white dots appear in an unordered fashion. This whitening is attributed to buckling of individual fibres and strands. It was characteristic that the fibre orientation loaded more severely in compression showed this damage mode.
2. Tear initiation. Usually in the vicinity of spar cap and glass fibre reinforcement corner (stress concentration) the first of these whitened strands breaks and consequently causes the first discontinuity in the load path.
3. Tear growth. Adjacent strands also starts to break. Crack grows in a more or less 45° direction, often running from the one spar cap to the other.
4. Tear makes a T-junction. Crack direction rotates 90°. Fibres perpendicular to the snapped fibres start carrying higher stress due to less area to transmit force and consequently start to snap.
5. Spar cap fails. T-junction crack changes to an X- shape. As soon as the crack has grown to such an extent that there are not enough fibres resisting the shear between the spar caps, the entire load is effectively carried by only one spar cap and it fails. Up until stage five the specimen was able to withstand cyclic loads up to limit load.

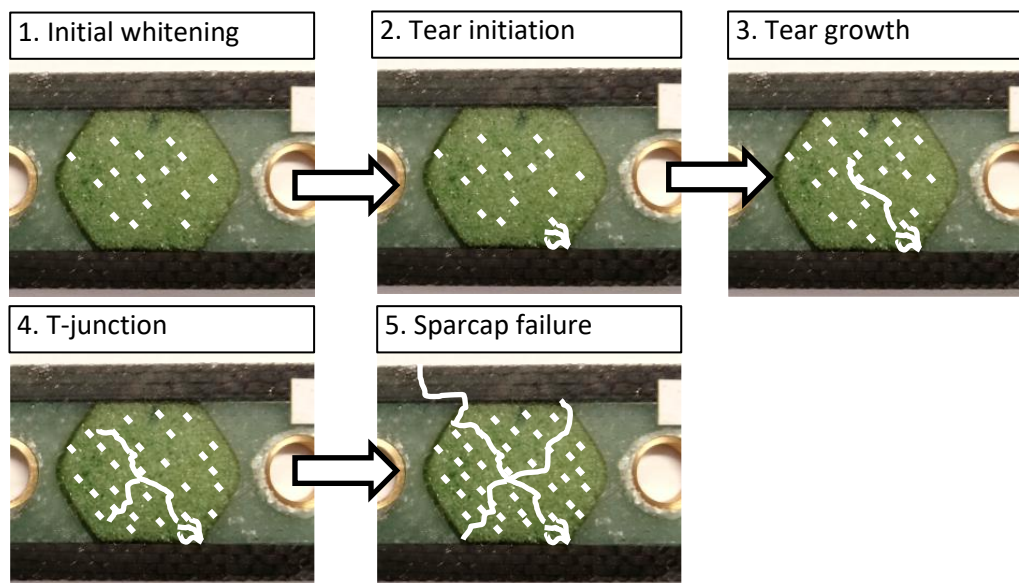


Figure 4-1: Failure

4.2. Observed irregularities

Composite fatigue testing is in its nature prone to large scatter (Tomblin & Seneviratne, 2011, p. 7). Ensuring that extra variability is not introduced during the manufacturing or testing procedure is paramount. Since any external variability might have an artificial contribution to the scatter of the data set, it is honest science to discuss any observations regarding this.

The fatigue data comprised of two experimental runs. Each experimental run consisted of various batches. In order to verify the consistency of both the manufacturing- and experimental procedure both these processes were conducted by different people for each experimental run.

4.2.1. 1st experimental run

The fatigue data comprised of sixteen test specimens extracted out of 2 separate manufactured batches. Six of the specimens showed no suspicious behaviour and their experimental procedure were completed without any deviation from the planned program. Five specimens had some suspicious behaviour and with five of the specimens a mistake was made in the experimental procedure. One specimen's observational report was lost, but the raw data was recovered.

Regarding the behavioural observations two phenomena was recorded. The first is the brass bush which loosened over time and the second is delamination between the spar cap and the shear web. It was concluded that the loosening of the bush may be due to one of two reasons. In four out of the five cases, it was due to poor bonding between the metal-composite interface and in one case slight bearing failure of the surrounding composite structure.

The problem of delamination occurred once and was attributed to poor preparation of the pre-manufactured spar cap. It is important to note that the delamination did not occur prematurely, but that the shear web was degraded in the normal fashion before the delamination occurred. What makes this observation notable is that all other specimen showed spar cap failure after shear web failure while this specimen showed spar cap delamination.

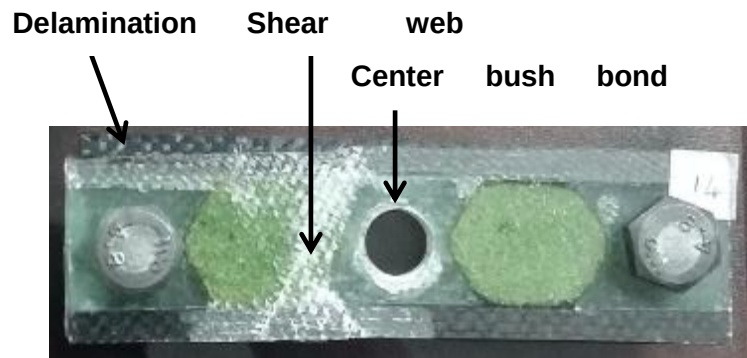


Figure 4-2: Specimen 14. Spar cap delamination after shear web failure

Two types of experimental procedure deviations occurred. The first is servo hydraulic response errors in which the response did not match the signal. This type of error is intrinsic to the servo hydraulic system and is present with every test run, but the compensation software usually solves the problem within the first few cycles. The problem was only logged for two specimens in which the software had trouble with the compensation and a considerable amount of cycles were completed before the response was within acceptable tolerance.

The second type of experimental procedure error was erroneous application of multiple test runs on a single specimen. Most test programs were frequently paused for inspection of the specimen or some other reason after which the test program was simply resumed. In some instances, the test program was mistakenly not paused, but stopped and restarted from the beginning. Consequently, the random spectrum would restart and the loading sequence was in principal different than that of another specimen. This mistake was made on three of the specimens.

ID	Bush came loose	Spar cap delamination	Compensation trouble	Accidentally restarting test program
14	X	X		
17	X		X	
22				
23	X			
24	X			
29				
30				X
31			X	X
32				
33				X
34	No observations logged (Lost report)			
35				
36				
37				
38	X			
39				

Figure 4-3: Observational summary of fatigue specimens

The observations noted here might bring doubt to some of the results. It is important to note that all the specimens used to compile these results exhibited the five stages of failure, which shows consistency with one another and also with the design, since failure occurs as expected.

4.2.2. 2nd experimental run

The fatigue data is comprised of eleven test specimens extracted out of 3 separate manufactured batches. The reason for more batches and fewer specimens than the 1st experimental run is twofold: Some specimens were used for other experiments and the manufacturing quality screening was stricter. Consequently, many specimens were not added to the data population due to poor manufacturing quality.

No irregularities regarding the damage or failure modes were reported. Deviations from the experimental procedure as documented for the first experimental run were not made for this experiment.

Though the scatter in data was still high for the second experimental run, it was an improvement from the first. This is attributed to the stricter quality control regarding both the manufacturing and experimental procedures.

4.3. Fatigue test results

Here the experimental results are displayed. Two types of graphical presentations are used namely: The Combined Cycles to Failure (CCF) and the equivalent cycles plot (Figure 4-4). The CCF plot is used to show the results obtained from the MLS testing program. N_{VA} refers to the fraction of KoSMOS II completed before a CA fatigue program was applied to the same specimen. N_{CA} refers to the number of constant amplitude cycles to failure of the specimen. N_{CA} is zero when failure occurred during the KoSMOS II loading program. N_{VA} is plotted on the x-axis and N_{CA} on the y-axis. Data points on the y-axis indicate the cases where no KoSMOS II cycles were applied.

The equivalent cycles plot (Figure 4-4) is derived from the Residual CA plot by means of Equation (5) and shows the number of CA cycles equivalent to a certain number VA cycles. N_{VA} refers to a fraction of KoSMOS and describes the x-axis. N_{CA} describes the y-axis and refers to the number of CA cycles.

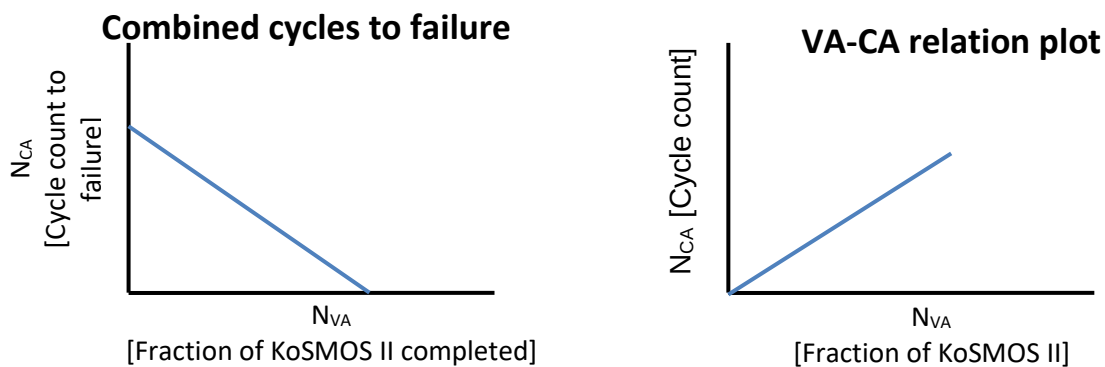


Figure 4-4: Forms of representing results

The data is categorised into three groups (Figure 3-5). Group 1 is the data which was only cycled by means of the CA fatigue program. Group 2 is the specimens which were cycled by both sequence types (CA and VA) and group 3 is the specimens which were cycled to failure by the VA program.

Table 4-1 shows the results as cycles to failure of both the experimental runs. Figure 4-5 represents the raw data with the mean linear regression lines of both experimental runs. The second run only had a single specimen in the 3rd group and consequently this point is not represented very well for this experimental run. The statistics for each experimental run are shown in Table 4-2.

Table 4-1: Raw data of two experimental runs

Experimental run 1			
Specimen ID	Group	N _{VA}	N _{CA}
		[Fraction of KoSMOS II completed]	[Nr of cycles]
22	1	0	910
23	1	0	484
24	1	0	373
29	1	0	326
14	1	0	227
17	1	0	694
36	1	0	379
31	2	0.11	64
32	2	0.10	351
33	2	0.11	106
37	2	0.10	180
38	2	0.10	227
30	3	0.24	0
34	3	0.40	0
35	3	0.24	0
39	3	0.21	0

Experimental run 2			
Specimen ID	Group	N _{VA}	N _{CA}
		[Fraction of KoSMOS II completed]	[Nr of cycles]
104	1	0	419
109	1	0	352
120	1	0	427
125	1	0	468
127	1	0	638
107	2	0.10	362
108	2	0.10	56
110	2	0.10	118
119	2	0.10	120
124	2	0.10	228
113	3	0.22	0

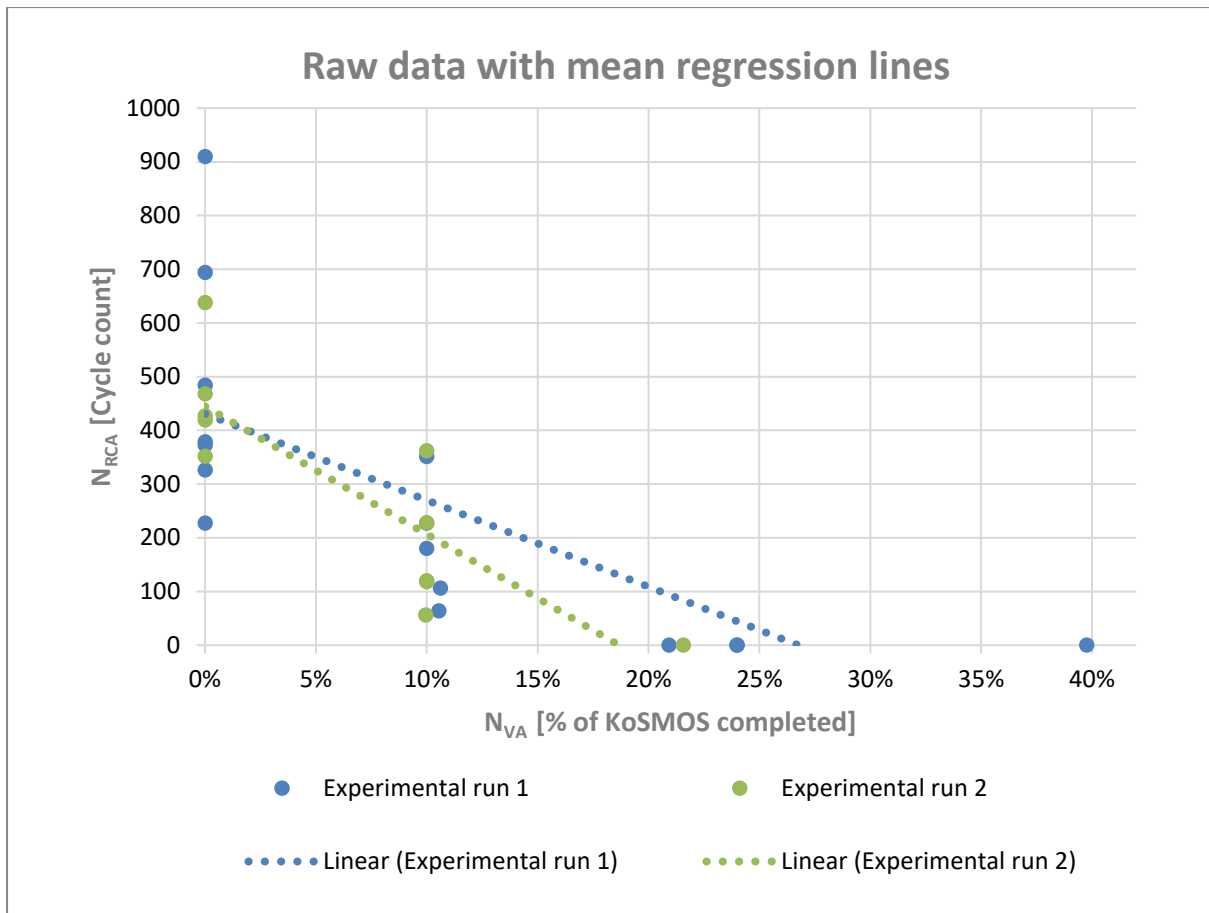


Figure 4-5: CCF with two experimental batches and linear regressions

Table 4-2: Statistics of the respective experimental batches

Experimental run 1					Experimental run 2				
		Mean	Std div	% Std error			Mean	Std div	%Std error
1	CA test	485	238	19%	1	CA test	461	107	10%
2	MLS	185.6	112	27%	2	MLS	176.8	121	34%
3	VA test	0.27	0.09	16%	3	VA test	0.22	-	-
Regression line		$y = -1611.1x + 431$			Regression line		$y = -2381.8x + 444.67$		
R^2		0.52			R^2		0.71		

Figure 4-6 displays the consolidated data of the two experimental runs. The three points displayed are the mean values of the entire population with the error bars representing the standard deviation. The mean regression line is plotted as well as the regression line of the 10th percentile data.

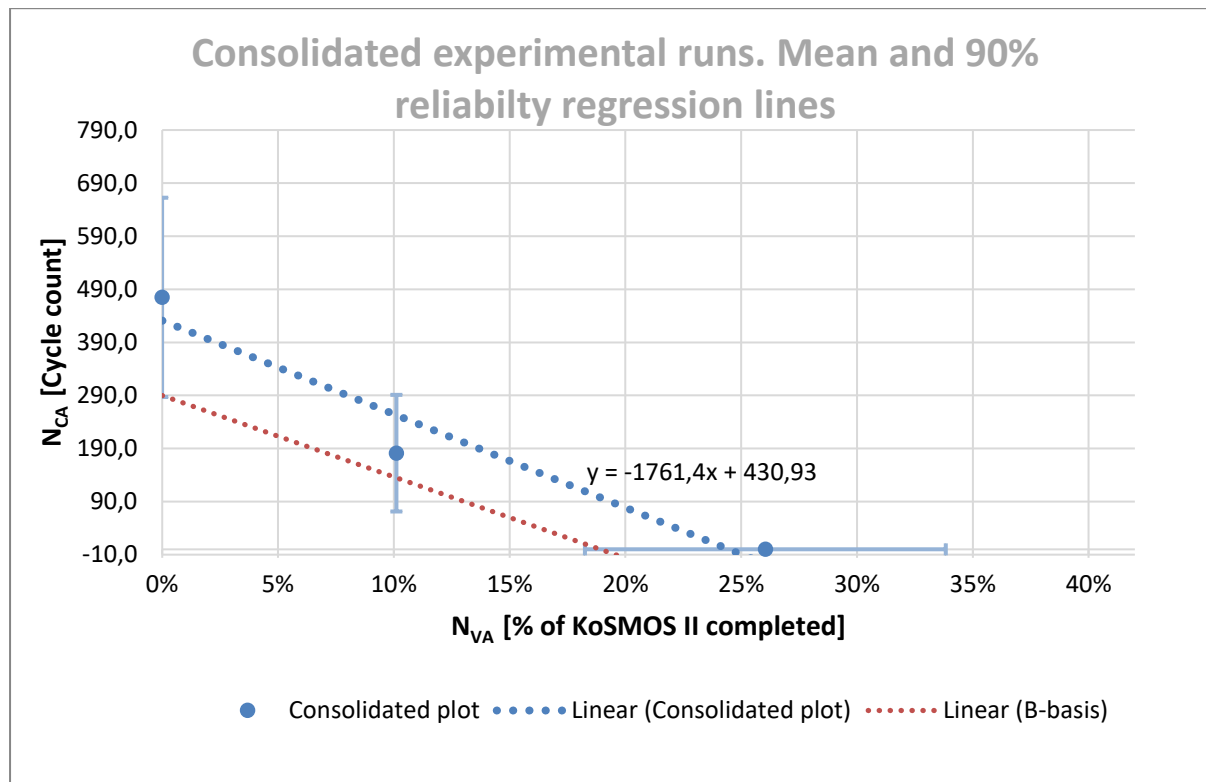


Figure 4-6: Two result batches, consolidated

Table 4-3: Two result batches consolidated

Consolidated data				
		Mean	Std div	%Std. error
1	CA test	475	188	11%
2	MLS	181.2	110	19%
3	VA test	0.26	0.08	13%
Regression line		$y = -1724x + 425.82$		
R ²		0.55		

The linear regression of the data yielded a $R^2 = 0.55$. It can be clearly observed that group C data is particularly poorly represented by the regression line. Also, when a 90% reliability curve was fit, this group was not well represented by this curve.

A quadratic fit was attempted (Figure 4-7) and yielded a much more satisfactory result. R^2 improved notably to 0.66, but due to large scatter a very high R^2 could not be expected. The 90% confidence line also represented all three groups of data much better.

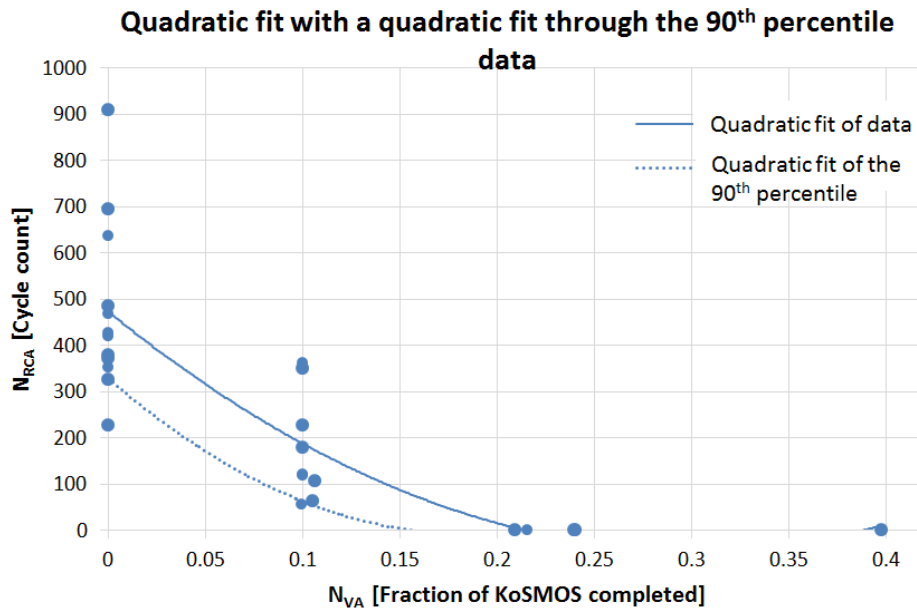


Figure 4-7: Second order fit of the entire population

Using Equation (5) the mean data is transformed to show the equivalence between VA and CA. The equivalence was then described as in Equation (8).

$$f(x) = -6910.2 \cdot x^2 + 3421 \cdot x \quad \{for \ 0 \leq x \leq 0.21\} \quad (8)$$

Where the function $f(x)$ being undefined outside these limits due to specimen failure and

$$f(x) = CA \text{ cycle count}$$

and

$$x = \text{fraction of KoSMOS II completed}$$

According to this equation the equivalence between the two sequence types can be shown in table form. This is also displayed in graphical form in Figure 4-8.

Table 4-4: CA-VA equivalence

% of total KoSMOS II	KoSMOS cycles	Mean curve equivalent CA cycles	10 th percentile curve equivalent CA cycles	90 th percentile curve equivalent CA cycles
0.01%	56	0	0	0
1%	5566	34	37	35
10%	55661	273	321	266
21%	116888	414	561	330

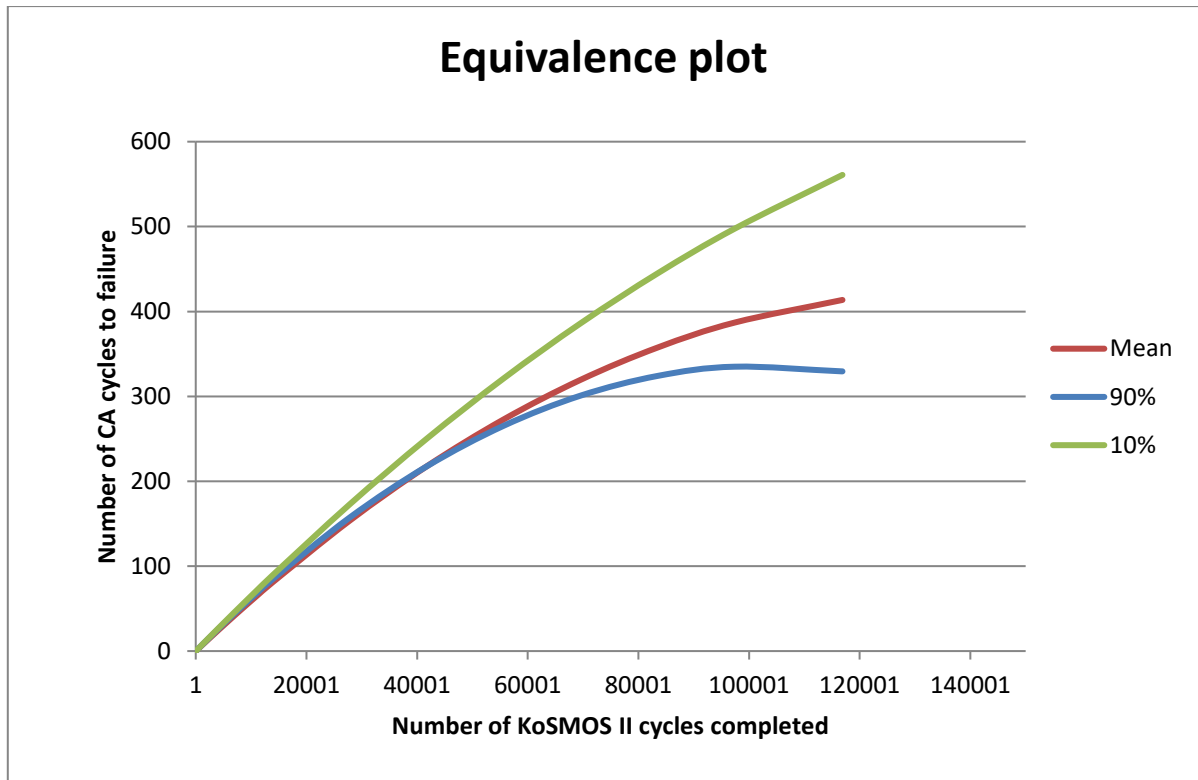


Figure 4-8: Equivalence plot according to entire population of results

Figure 4-8 is the essence of the study. The study evaluated the RCA method to establish whether the equivalence curve would indeed be linear or not. What is observed from this result is that the equivalence can resemble linear behaviour in some cases, namely:

- Low cycles: The first third of the data shall not be poorly represented by a linear relationship. This shows that in the early stages when damage is initiating linear behavior can be expected and the Palmgren-Miner assumption is valid.
- Strong specimen: The 10th percentile line basically illustrates the behavior of all the strong specimen. This line is considerably flatter than the mean and 90th percentile, showing also that the Palmgren-Miner assumption is a good assumption when considering the structurally best specimen in the population.

Unfortunately, when considering the entire population linear behaviour cannot be assumed. The tail end of the curve shows a dramatic increase in uncertainty, which is due to variation in damage growth when nearing the end of the specimen life. As discussed in 4.1, only the specimen exhibiting the 5 stages of damage development was considered valid data. This means from a macroscopic observational point of view damage accumulation variation was not because of different failure modes, but due to variation in the rate of failure following similar damage mechanisms.

Further discussion about the results and possible future applications and topics for study are given in the final chapter.

CHAPTER 5 CONCLUSION AND RECOMMENDATION

In this chapter the study is concluded by evaluating the extent to which the experimental results have addressed the problem statement of chapter one. Some critique and suggestions for future studies are also discussed.

5.1. Conclusion

5.1.1. Summary

The experiment conducted and results attained, do address the problem statement, but more indirectly than initially projected. Also, the empirical investigation which was conducted is very preliminary in nature. The proposed solution to the research question is only partially discovered and requires additional empirical work.

The goal of the study was to find some method which would make a contribution in accelerating the process of fatigue certification in order to be able to certify longer service lives at less financial cost. Literature showed that replacing the time-consuming flight load spectrum with a CA test sequence has a considerable effect in reducing the testing time of structures. In this study an alternative method for finding the equivalence between these two test sequences was proposed. The proposal was investigated empirically.

The results did not show that the proposal is invalid, but that the application thereof is more complex and time consuming than initially hoped for. Still there is some advantage to using it rather than other current methods.

5.1.2. Detail discussion

The proposal or hypothesis was: The amount of CA cycles equivalent to a certain amount of VA cycles can be found by means of establishing two points on the *Combined Cycles to Failure* graph. This method was founded on the assumption that damage accumulates linearly as predicted by PM. In order to verify the applicability of the proposal it was required to evaluate this foundational assumption, namely, linear damage accumulation.

By empirically finding various points on the CCF graph it was possible to investigate the linearity of damage accumulation when using these two specific load sequence types (namely CA and KoSMOS II VA) at a specific reference load.

This method showed that a quadratic rather than a linear fit of the data is more appropriate. Consequently, the PM assumption of linear damage accumulation is not valid and the equivalent cycles cannot be established by the trivial method of finding only two points on the CCF graph.

Nonetheless, the equivalence was still found using the method, but required more points on the graph. This in turn means longer testing time than initially hoped for. This is unfortunate, because the method may now tend to amount to a similar or larger empirical test time as is traditionally required for CL and SN diagrams, but this is a dilemma which needs investigation in a subsequent study.

An advantage of this new proposal is the confidence in the equivalent cycles. Considering Figure 4-8, the mean, ninetieth and tenth percentile line is shown here. It is important to note, they result in very similar equivalence curves. It is only when reaching the point close to failure when deviation from one another starts to occur.

Thus, it can be said that there exists considerable confidence in this curve up to the point close to failure. This confidence teaches us that though the specimen under consideration is weak or strong, it does not have an influence on what the equivalence plot looks like in the initial stages of damage accumulation.

Furthermore, the problem of the equivalence scatter at the tail end of the curve is not serious when viewed in practical terms. The goal of a testing program, as investigated here, is to extend the service life of a sailplane. This is not to actually find the point of failure, but merely to extend the life. Consequently, the life extension program might most probably never reach this point of equivalence deviation, and if it does the most conservative curve could simply be chosen as the governing curve.

Contrary to this, traditionally the equivalent cycles are purely based on a PM calculation which has been shown to be erroneous in so many cases. To compensate for this reliability issue large life factors are used when testing the full-scale specimen. The new proposal is based on an actual damage metric, namely residual CA fatigue cycles (RCA cycles) and the empirical investigation showed high confidence in the equivalence. Consequently, if investigated in more detail, the required life factor on a full-scale specimen test could be reduced considerably.

5.2. Critique and recommendations

Experimental results exhibited large scatter. Though this is not uncommon for composite fatigue testing, it is believed that it could have been reduced if a specimen of less complex nature was used. Since this research is still in an exploratory phase it is suggested to use a specimen design which is as simple as possible in subsequent investigations.

The method proposed used residual CA cycles to quantify the amount of damage in a specimen. Through this value the equivalence between VA and CA was derived. It is proposed that the validity of this equivalence is investigated through residual strength or stiffness experiments.

The experiments were done at excessive strain values which might bring the validity of its application to sailplanes into doubt, as these strain levels is not representative to strains experienced during flight. This was done to achieve quick damage accumulation. It is

suggested to experiment at strain levels more representative to operational strain levels of a sailplane. The VA-CA relationship might even be closer to the linear approximation of the PM hypothesis when testing at lower strain values.

The section of the VA sequence tested must be representative of the entire VA sequence in future experimentation. This was not the case, in the experiments done. Sections of the VA sequence were used, but these sections were not necessarily a very good representation of the whole. An analysis of the deviation is shown in annexure⁸.

Only three points were determined in the experimental space. This limited the choices of curve fitting. In subsequent experiments it is proposed to have more variance in the x-axis so as to gain greater confidence in the curve fitting.

5.3. Application of the proposal

At this point the RCA method has been proposed, evaluated and scrutinised, but the question remain: Is there a possibility for a real-world application thereof? The author means to encourage an argument that there may truly be a real-world application, based on the following.

1. The method is not novel in the sense that it is a total new approach to fatigue certification. The proposal is simply replacing one phase in the three-phase process (namely empirical data collection) with a novel approach. It is thus still grounded in the trialed and tested process. Furthermore, CA testing has become the standard work practice in certain industries. The only adjustment is how to established a CA testing program.
2. This preliminary study has shown that a CA test program is attainable using the RCA data collection method.
3. The CA equivalent cycles are directly related to empirical data whereas the empirical methods used up to now are rather indirectly related to the empirical since the CA equivalent cycles are derived from the empirical using Palmgren-Miner.

The application of the proposal to an actual sailplane life certification program can be trivially explained by two steps

1. Establishing of CA-VA equivalency for a specific structural layout (Phase 1 of FCP)
2. Conducting a CA certification test on a full-scale structure based on the CA-VA equivalency (Phase 3 of FCP)

⁸ KoSMOS spectrum analysis – Annexure 9

The establishment of the equivalency (Step 1) is by far the bulk of the work, but once this is done it can be used in various certification programs as long as the aircraft being certified is of similar structural layout and built of similar materials.

The equivalency is determined experimentally as described earlier. This is started on material coupon level and gradually scaled up to structural details and components as dictated by the building block approach. Analysis of the data scatter in a lower level indicates the minimum amount of specimen required in higher levels of the building block pyramid until finally it is possible to test a single full-scale specimen with a high confidence and a minimum scatter factor.

Once this is completed an equivalence graph is established with high confidence and can be used to substantiate various aircraft in the future (given their structural layout is similar to the structure used to establish equivalency). In some preliminary tests completed on specimen the equivalence ratio was shown to be 200. This means that full scale life substantiations could be completed in a 200th of the normal testing time. Not only does this reduce the certification budget but it also allows us to conduct life extensions further than 12000 flight hours which, up to now, has been very impractical.

This order of a magnitude time being saved is not novel- it is the time being saved when using CA rather than VA, and has been proposed for many years now. Transforming industry to CA is a tremendous leap in methodology improvement which industry has been yearning for, for decades. For this reason, it remains a topic which requires much attention.

The area where this study focussed was not on transformation to CA -this according to the author is a topic above debate simply due to enormous potential in time saving. This study rather focussed on the preliminary empirical data used to establish the CA testing program and it was shown that the RCA method is a possible alternative. The question which remains is whether the RCA method is superior to the traditional Palmgren-Miner method in finding the equivalent CA cycles.

It is the author's opinion that when the superiority is weighed in subsequent studies the time factor is not the only factor which should be weighed. Accuracy and precision also become important factors, since great accuracy and precision shall ultimately improve confidence and reduce the required scatter factor in the full-scale specimen test, consequently reducing test time in the 3rd phase of testing.

In other words, even if the RCA is slower than traditional empirical methods, and increases phase 1 of the FCP, but due to higher confidence reduces test time in phase 3, it could also count as a total improvement of the FCP.

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ANNEXURES

1. Sailplane main spar design

The main structure of the wing of a typical FRP sailplane is called the main spar. This spar consists of three critical components namely:

- *Bottom and top spar caps*: Responsible for carrying the normal stress (Tension and compression) induced in the wing due to the bending moment of the wing load (See Figure 7-2). The spar caps consist of uni-directional glass or carbon fibres (also) directed in the longitudinal direction of the wing. The top spar cap is mostly in pure compression while the bottom spar cap is mostly in pure tension.
- *Shear web*: Responsible for carrying the shear stress (See Figure 7-2) induced in the wing due to the wing loading. The shear web usually takes the form of a sandwich composite material, where a core material is wrapped with a composite bi-directional laminate orientated 45° (See Figure 7-1).

It was shown that the shear web of a typical spar is mainly loaded in shear and that the normal stresses induced by bending are carried by the spar caps. When the principal stresses of a body loaded in pure shear are examined, it can be observed to be orientated at 45° (Hibbeler, 2008, p. 479). In figure____, the finite element model a FRP box structure spar is shown. The principal stresses are indicated by vectors orientated at 45° , showing that the shear web is loaded almost in pure shear.

Since the principal stresses are loading the shear web at 45° , the bi-directional glass fibre used in the shear web is also designed to be orientated at 45° . This means that the shear web laminate is loaded similar to a $0^\circ/90^\circ$ laminate loaded normally.

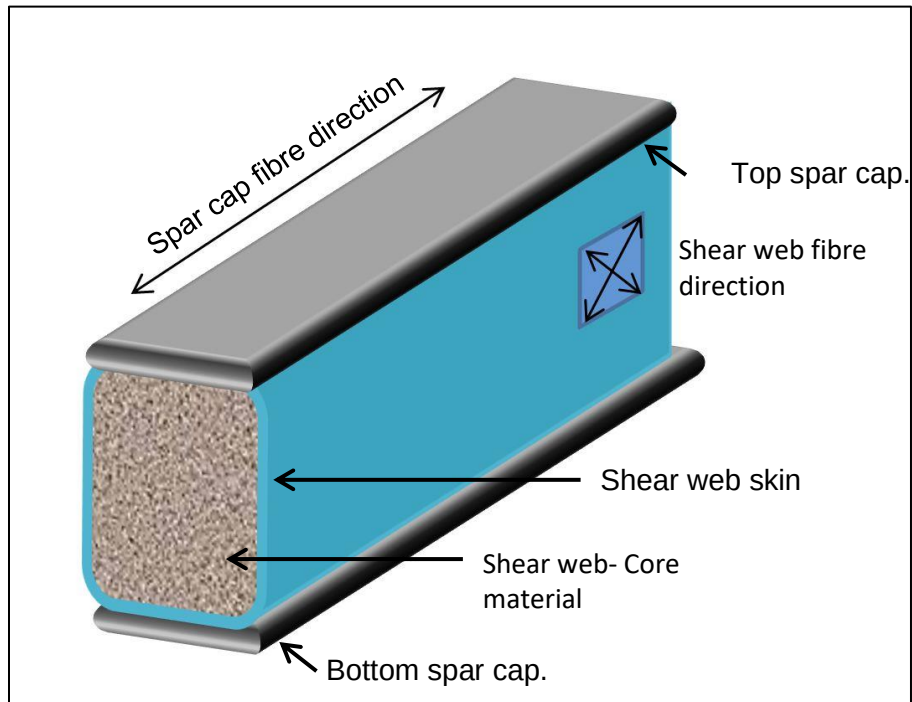


Figure 7-1: Typical glider spar design cross sectional view.

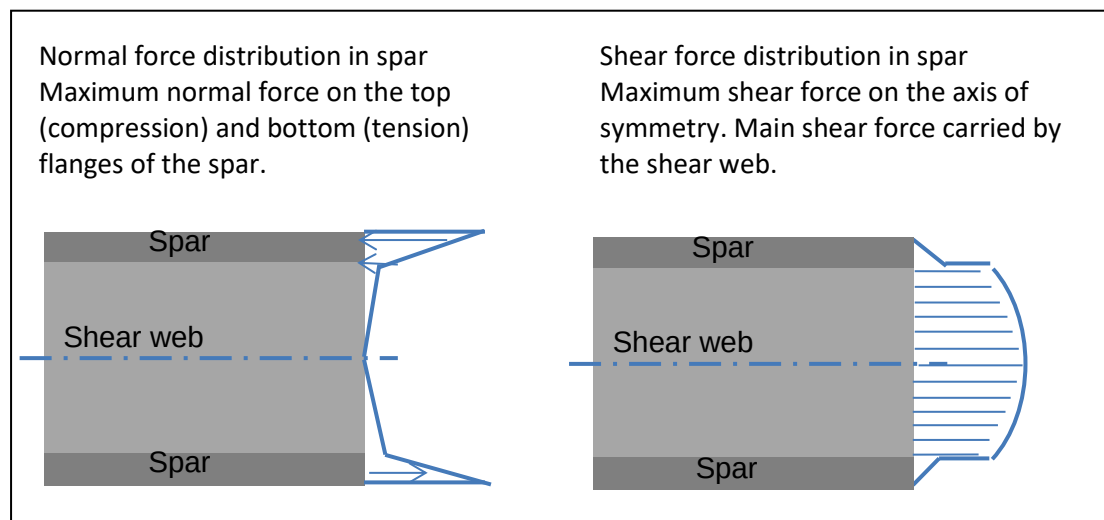


Figure 7-2: Load distribution on a sailplane main spar (Simplified representation).

2. Loading pattern

Considering all the building blocks of testing, two types of fatigue tests on the specimens may be conducted, either constant amplitude (CA) or variable amplitude (VA). CA tests may be used to determine SN and CL diagrams in order to be used for spectrum life predictions. It may also be used to determine load amplitudes which cause negligible damage and could be omitted in the VA test or to investigate the effects of different variables on the fatigue (Baker, et al., 2004, p. 483). VA testing is most often used as validation of life predictions. Figure 7-3 and Figure 7-4

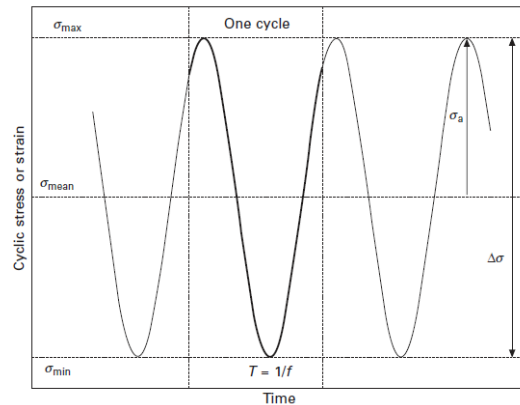


Figure 7-3: Constant amplitude fatigue program (Vassilopoulos, 2010, p. 9)

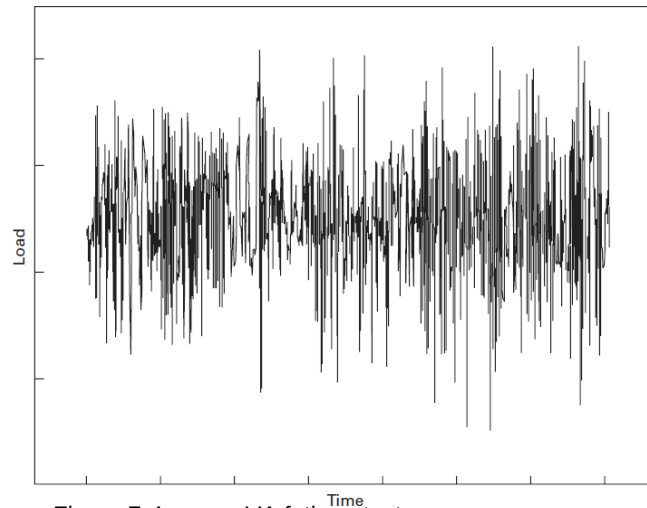


Figure 7-4: VA fatigue test sequence (Vassilopoulos, 2010, p. 10)

3. Variable amplitude loading

According to Megson (Megson, 2007, p. 377) one of the main topics of concern regarding fatigue in airframes is the knowledge of the loads causing fatigue damage. These loads are also referred to as the load history, meaning the spectrum of loads which the airframe will experience throughout its life cycle. Without knowledge of the load history neither life prediction nor life cycle substantiation tests can be done, since the loading amplitude and cycle count need to be connected to actual fh.

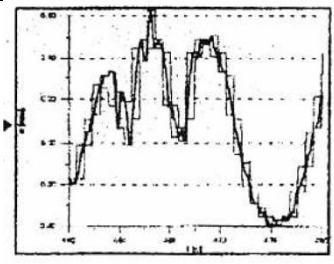
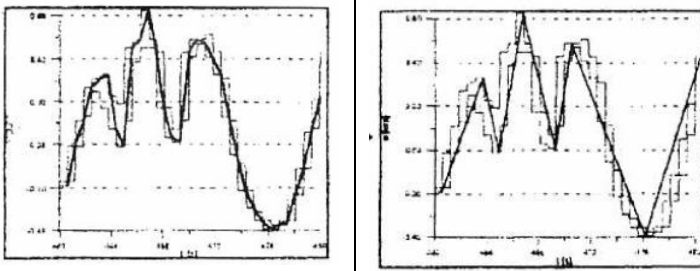
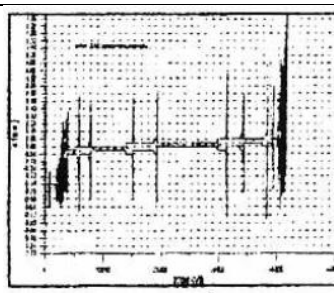
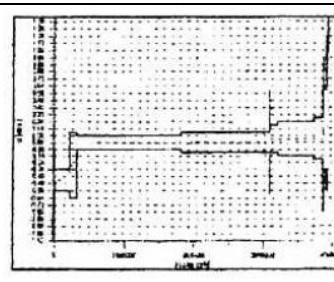
Life cycle substantiation for certification programs cannot make use of the load history for testing, because it will simply take too much time. This problem is solved by deriving a fatigue test spectrum from the load history (Niu, 1990, p. 570). The spectrum is designed to decrease the required test time while still preserving a representative load sequence.

In the following figures a few illustrations of the process of spectrum derivation is shown- Rodzewicz (2000, p. 17) explains the process of generating a block fatigue test spectrum from raw measured flight data. This process can be seen in Table 7-1. Figure 7-5 illustrates an alternative process for deriving a VA test sequence from measured flight loads proposed by (Kossira, n.d.) in which the loads are not grouped into blocks. In this process, measured loads are filtered and classified in 32 load classes. The frequency of load occurrences is stored in a half cycle transition matrix from where half cycles can be pulled randomly to generate a loading sequence.

Nui (1990, p. 571), in Figure 7-6, shows four different types of VA load sequences namely:

1. Flight by flight random load sequence taking GAG (ground-air-ground) cycles into account resulting in two mean values, one for the ground loads and one for the flight loads.
2. Flight by flight with high and low loads ordered together.
3. A block sequence not taking ground load into account. Notice the mean value of all the loads stays constant.
4. A block sequence taking ground cycles into account. Two mean values exist, one for the ground loads and one for the flight loads.

Table 7-1: Load spectra derivation according Rodzewicz (2000, p. 17)

Step	Description	Illustration
1	Raw signal determined by in-flight measuring of aircraft. This can be done with an accelerometer to describe the spectrum in terms of load factor. Alternatively, the spectrum can be described by means of a strain measurement of the spar.	
2	Raw data filtering	
4	Creating initial spectrum: Sorting the signals into blocks of loads of equal amplitudes. Amplitudes are sorted into a predetermined number of load levels.	
5	Load spectrum simplification: Reducing the number of blocks by replacing the blocks with new blocks covering the same range of amplitudes.	

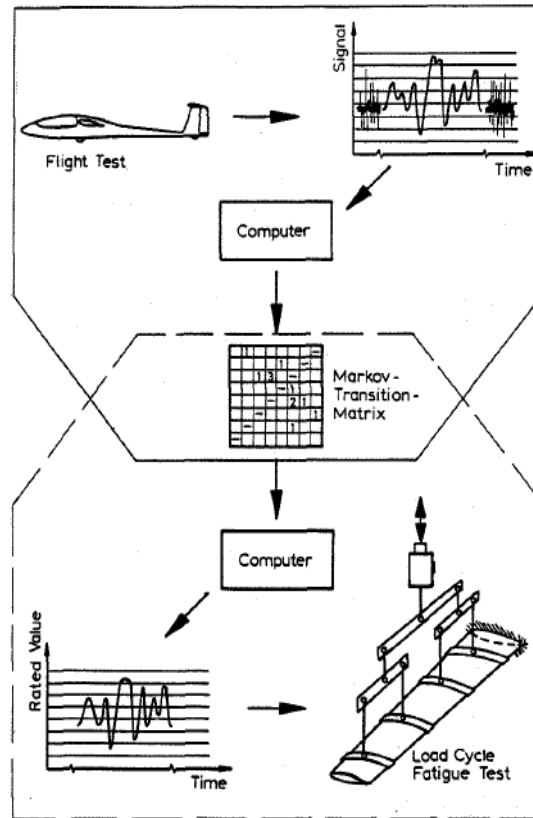


Figure 7-5: Flow chart of determining a VA load spectrum. From flight loads to full-scale fatigue test according to Kossira (n.d., p. 1331)

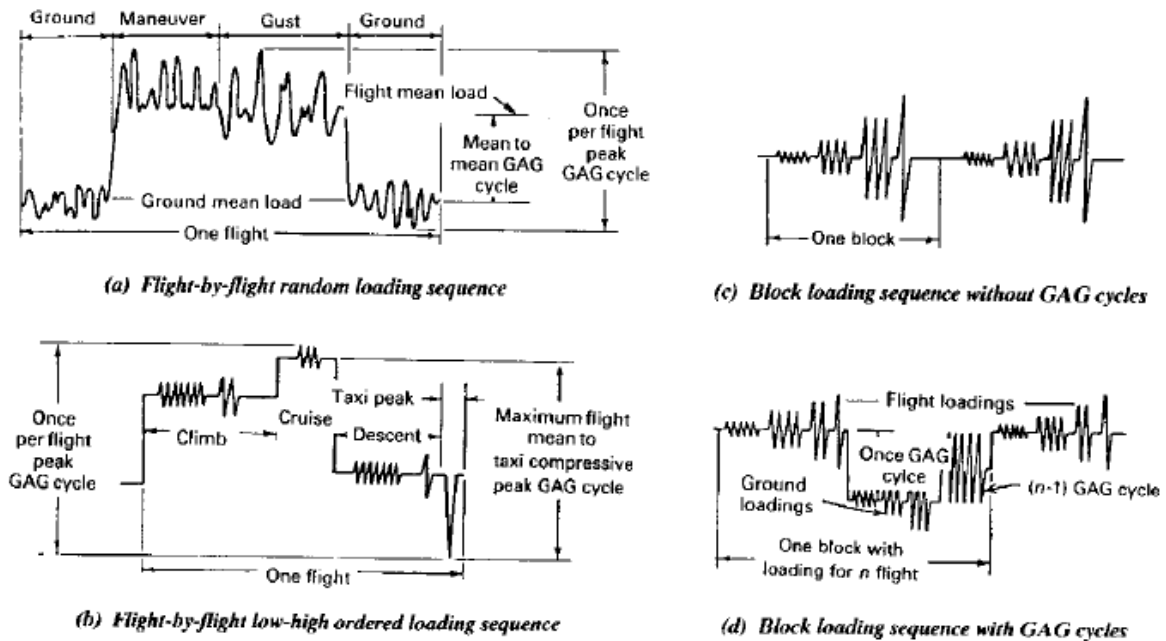


Figure 7-6: Different type of VA load sequences (Niu, 1990, p. 571)

4. Sailplane load spectra

Numerous loading spectra have been developed for sailplane certification purposes. In the introductory chapter the first VA sequences of Eugen Hänle and of Thielemann/Franzmeyer was briefly discussed.

The Thielemann/Franzmeyer spectrum was a block spectrum developed for German glider life certification. In 1938 H.W. Kaul made c.g. acceleration measurements of the frequency and density of gusts (Kossira, n.d., p. 1330). The spectrum, though developed from measurement, was constructed using very theoretical methods. The spectrum is assumed equivalent to 3000 fh and consists of 1.96×10^6 cycles divided into six load factors and four blocks, each block with a different mean value and R-ratio. The four blocks are:

1. gust loading during winch launching
2. take-off and landing
3. gust loading during aero towing and cross country flying
4. landing impact and Taxiing

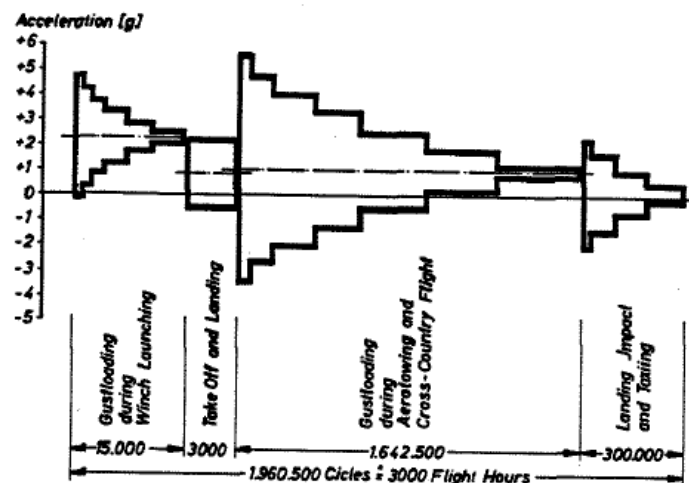


Figure 7-7: Thielemann/Franzmeyer loading spectrum (Kossira, n.d., p. 1331)

The arrival of the fourth sailplane era changed so many things that the applicability of the Franzmeyer spectrum was doubted. New material systems allowed higher aspect ratios, thinner airfoils and higher wing loading. Annual flight time increased due to the possibility to fly larger distances. The changes required the development of a load spectrum which was more representative to the new era flight characteristics.

Kossira and Reinke (Kossira & Reinke, n.d., p. 374) developed the *Kollektiv für Segelflugzeuge, Motorflugzeuge und Motor-Segler* (KoSMOS) which is a random load spectrum equivalent to 6000 fh. The spectrum was derived from in-flight strain measurements of the spar of a Janus two-seater glider and is applicable to sailplanes, light aircraft and motor-gliders. The KoSMOS spectrum consists of $1,74 \times 10^6$ cycles organised in 32 amplitude levels. The mean and R-ratio of the entire spectrum is constant ($R=-0.55$) (Kossira, n.d.).

The spectrum, though 5.36% of low amplitude cycles have been omitted, is considered a very conservative spectrum, since it has taken many different flight conditions into account. 12,5% of the spectrum is aerobatic flight and according to Soenne (2015, p. 12) this is the only sailplane spectrum which has included this. The flight conditions considered in this spectrum are: Aircraft towing up to 700 m; Aero towing; Winch launching; Landings; Flight training around the airfield; Cross country flying; Mountain flying; Wave flying; Aerobatics.

Figure 7-8 illustrates the KoSMoS spectrum. The y-axis being the percentage of limit load for the 32 classes and the x-axis the frequency of occurrence.

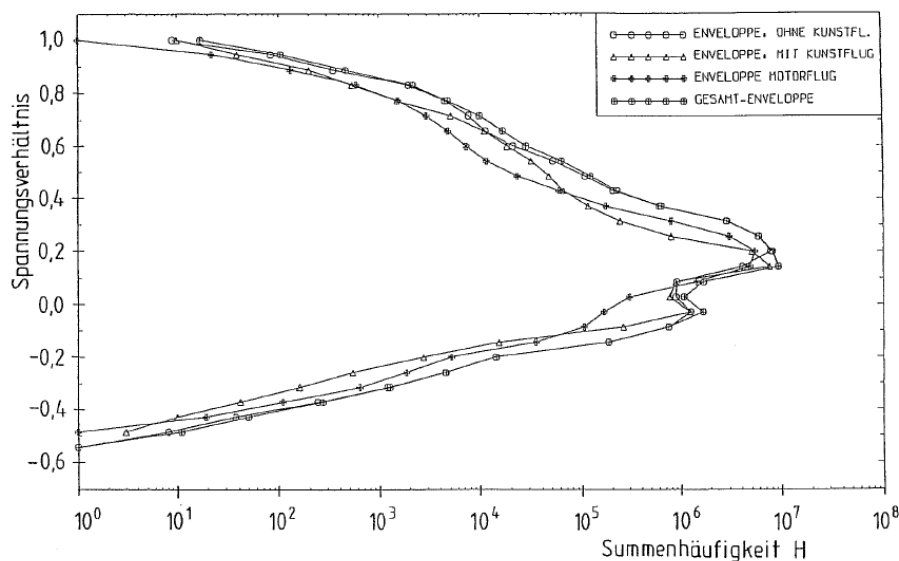


Figure 7-8: KoSMoS spectrum (Kossira & Reinke, n.d., p. 374)

Further development of the KoSMOS spectrum was done to reduce the amount of cycles to consequently reduce testing time. Omission of all $\pm 7.14\%$ amplitudes resulted in the KoSMOS II spectrum with only $0,56 \times 10^6$. This is now considered a sufficient spectrum by EASA and LBA for life certification testing.

Other glider load spectra are:

- The Dorning spectrum developed for Australian flying conditions. It was developed using the measured load signals on various sailplanes. This is the spectrum which was used on the Janus fatigue test conducted in Australia.
- SZD-51 "Junior" spectrum used by polish researcher Miroslaw Rodzewicz (Rodzewicz, 2000, p. 15) to substantiate 9000 fh of the PW-5 sailplane. This spectrum is described by classes of load factors and is divided into 11 blocks and consists of 900×10^3 cycles which accounts for 1000 simulated fh.

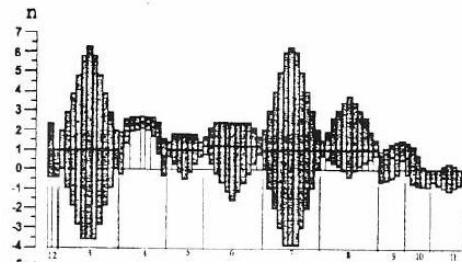


Figure 7-9: SZD-51 "Junior" block load spectrum

- Kenturi spectrum, which was used for the fatigue test on the PIK-20 sailplane. This spectrum was based on power-spectral methods as proposed by the FAA.
- Nystrom spectrum which was derived from Kenturi and used for the fatigue evaluation of the PIK-20D. This spectrum simulated 16000 flight hours and was a block type consisting out of 40 blocks of 400 fh each (Nystrom, 1978, p. 38).

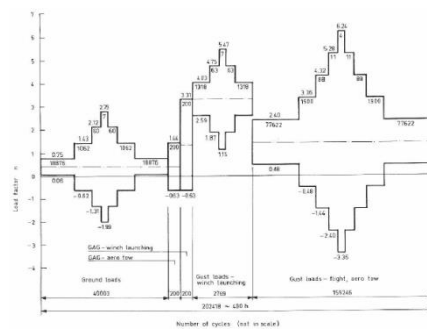


Figure 7-10: Nystrom block load

5. Fatigue data representation

Data obtained from fatigue testing of either test specimen type can be represented graphically. Two methods of representation are discussed here namely stress-cycle and constant-life representation.

5.1. Stress-Cycles (S-N) curves

S-N curves are the simplest representation of the fatigue properties of materials or components. Loading amplitude is plotted on the ordinate while the number of cycles to failure is recorded on the abscissa. The curve provides the cycle limit (point of failure) for given loading amplitude. When a component is designed with attributes above the curve, fatigue failure will occur. Each curve on the diagram assumes a constant loading ratio (R). R is given as the minimum load ($\sigma_{\min}$) over the maximum load ($\sigma_{\max}$) as shown in Equation (9).

$$R = \frac{\sigma_{\min}}{\sigma_{\max}} \quad (9)$$

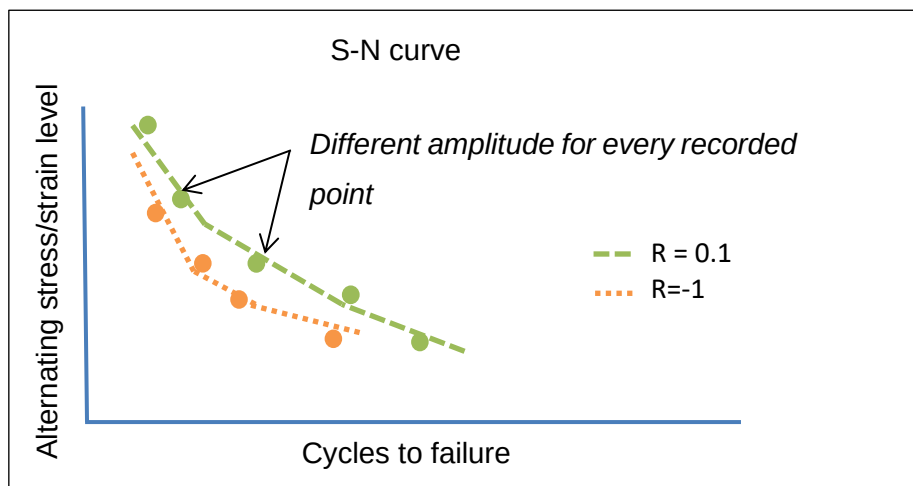


Figure 7-11: S-N curve

5.2. Constant life diagrams

Constant life diagrams (CLDs) are essential tools for estimating fatigue life cycles of materials or components under complex loading patterns (Vassilopoulos, et al, 2009). The CLD is an extension of the SND because it can be seen as a projection of the SN curves on a plane perpendicular to the life axis of the SND (See Figure 7-12). It is called a CLD because it connects points of failure at equal life cycles but different load ratios (Nijssen, 2007, p. 38).

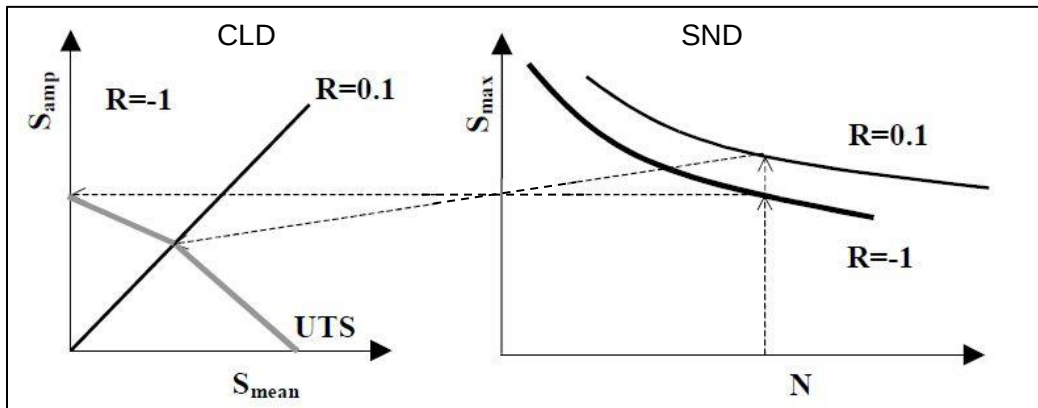


Figure 7-12: Constant life at different R ratios. (Nijssen, 2007, p. 38)

Though this graph yields a curve for varying R-ratios the shape is dependent on the loading amplitude used during testing. Stress amplitude is plotted on the ordinate while the mean stress is recorded on the abscissa. The different R-ratios are linear functions on this graph with a y-intersection at zero.

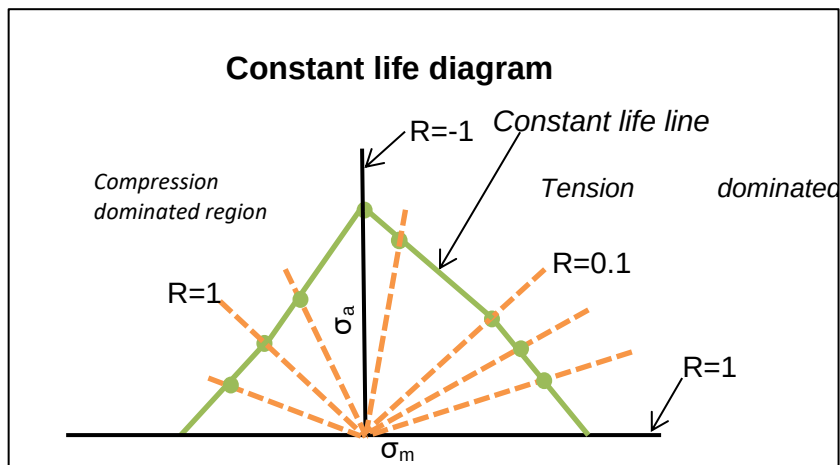


Figure 7-13: Constant life diagram

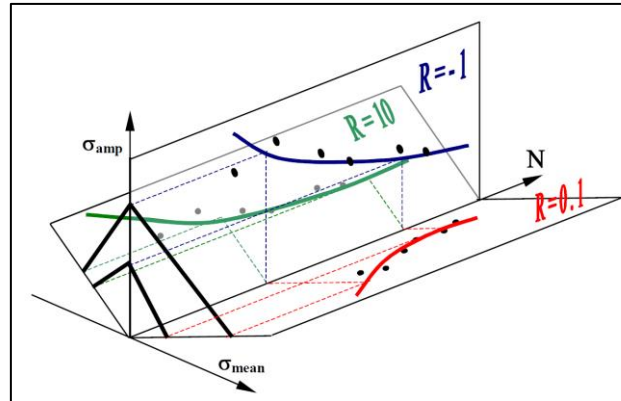


Figure 7-14: Relationship between CLD and SN diagram (Nijssen, 2007, p. 39)

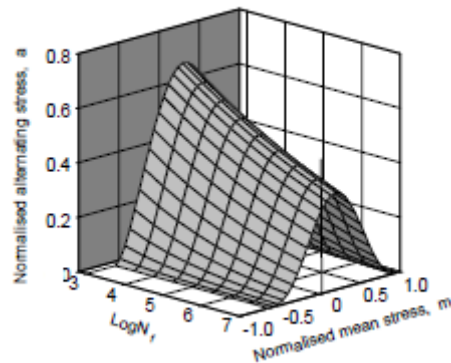


Figure 7-15: CLD curve (Harris, 1999, p. 147)

Limitations associated with both the above mentioned representation methods are the large amount of test specimens required, because of large scatter (Rodzewicz, 2010) and prolonged testing periods because of the cycles for each specimen reaching, kilo, mega or giga ranges before failure. For this reason, most tests are only conducted at load ratio $R = 0.1$; 1.0 and 10 and other R -ratios are interpolated.

6. SN curve fitting

The large scatter in the data of composite experiments requires conservative methods of fitting the SN curve. In a study conducted by Krause et.al. (n.d., p. 4) three commonly used methods are compared namely:

- pure linear regression
- Whitney
- Sendekyj.

Krause et.al. investigated how the life prediction differs, when using different methods of data fitting. As can be seen in Figure 7-16, he concludes that the method of data fitting does not play a large role. The flat curves at the bottom show the normalized data with the Sendekyj method as reference. The Sendekyj method has become popular in the wind turbine and sailplane industry.

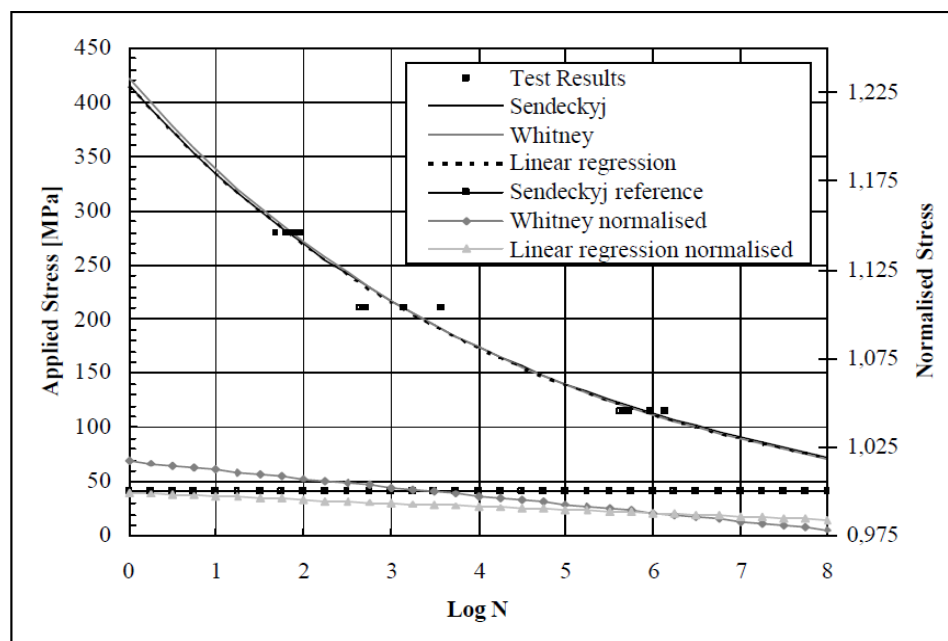


Figure 7-16: SN curve fitting method comparison (Krause, et al., n.d., p. 9)

7. Design allowables

Design allowables determined from the building block method are the limiting values which a designer uses to design the structure. Reliability of these design values are of great importance to ensure a safe structure. The aircraft industries have adopted two types of allowables which are generally accepted as representing good reliability. These are the A-allowable and the B allowable (Baker, et al., 2004):

- A-allowable means 99% of the population achieved this value with 99% confidence (99% reliability with 99% confidence).
- B-allowable means 90% of the population achieved this value with 95% confidence (90% reliability with 95% confidence).

The A-allowable ensures quite high certainty, but can only be used when the data exhibits relatively low scatter. The B-allowable was chosen as an alternative for large data scatter. If the A-allowable for data, which has large scatter is used, the amount of test runs shall become impractical (Baker, et al., 2004, p. 483).

In order to find the A- or B-allowable it is first necessary to find the statistical distribution of the data. The distribution for composites follows the two-parameter Weibull distribution most frequently, but often the normal or log-normal may also be used. The number of test specimens required can be reduced significantly if the parameters controlling the distribution functions are known (Baker, et al., 2004, p. 483). According to Rouchon (2006, p. 9.22) the shape parameter for Weibull distributions of static and fatigue properties of composites are 20 and 1.25 respectively.

8. Phenomenological life prediction models

Passipoularidis and Brondsted (Passipoularidis & Brondsted, 2009, p. 6) define the phenomenological models as contrasting to the empirical models, because it attempts to relate the physically damaged state of the material with a measurable mechanical property, rather than a dimensionless damage parameter D , like the PM method. Most often either residual material strength or residual stiffness is used. Note that similar to the empirical approach, the micro mechanical damage mechanisms are not taken into account, but simple quantifiable macroscopic parameters are.

8.1. Residual strength models

Residual strength models correlate the percentage damage accumulation in the material to the residual static strength of the material after cycling. These models all have a built-in failure criteria. That means that the model will signal failure when the residual strength degrades to the maximum applied stress as shown in Figure 3.10 (Passipoularidis & Brondsted, 2009, p. 19).

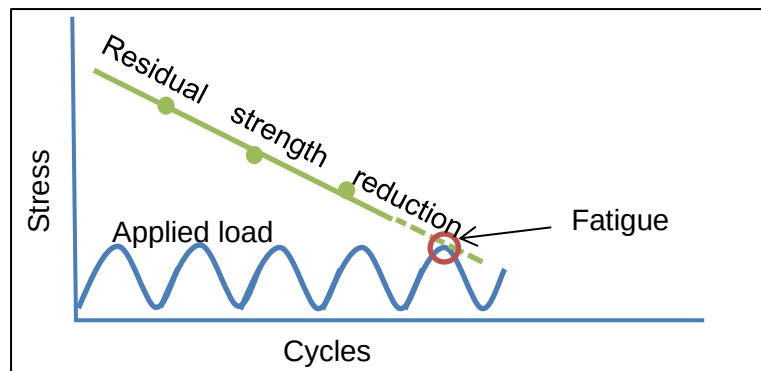


Figure 7-17: Residual strength principal

Contrary to the empirical method, the effect of each load cycle on the material strength is determined. This means that sequence effects can be taken into account (Nijssen, 2007, p. i). He also notes two negative aspects. Compared to the PM method, residual strength models are much more computationally intensive and for the model to work it requires a considerable amount of data on the material. Two other disadvantages are the phenomena of sudden death and the issue destructive testing. Sudden death, meaning that the residual strength reduction is minimal, until it reaches a point close to the end of its life time (Vassilopoulos, 2010, p. 18).

8.2. Residual stiffness models

Residual stiffness models were primarily developed to improve the shortcomings of sudden death and destructive testing which were inherent with the residual strength models

(Vassilopoulos, 2010, p. 18). Stiffness reduction is more gradual in its reduction rate and strain can be readily measured periodically or even during fatigue testing without influencing the specimen.

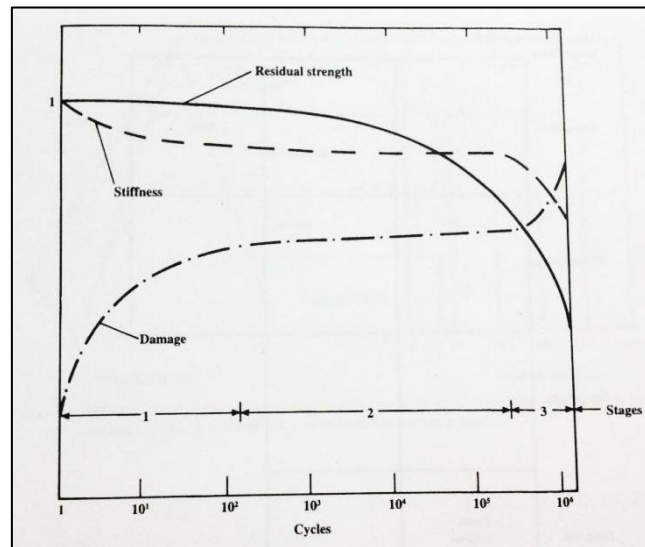


Figure 7-18: Three characteristic stages of composite fatigue (Mayer, 1996, p. 5)

Residual stiffness models correlate the material modulus to the percentage damage accumulation. These models are convenient because the modulus can be measured in time during cycling and can be done non-destructively (Passipoularidis & Brondsted, 2009, p. 23). A problem however is to choose a failure criterion. Various failure criteria have been suggested-all related to the strain parameter. In general, stiffness reduction in FRP occurs in three stages as illustrated in Figure 7-19.

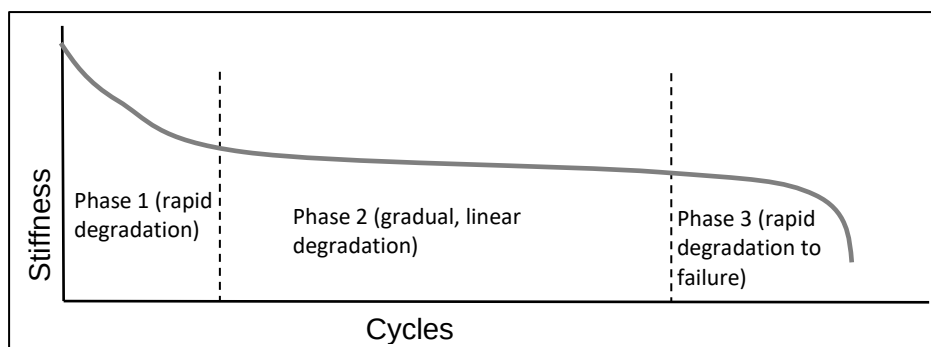


Figure 7-19: Stiffness reduction regions (Nijssen, 2007, p. 12)

9. KoSMOS spectrum analysis

For the study reported upon in this document, the testing time per test run was reduced considerably by increasing the limit load applied to the test specimen. The high limit load reduced the life cycle under VA loading to such an extent that final failure occurred before the VA sequence could be completed. The implications hereof were that only one specimen experienced more than 24% of the VA loading pattern and all others failed before 24%.

An important concern related to not running the entire VA spectrum is that sections of the spectrum are not representative of the entire spectrum. An analysis of different sections was done. The KoSMOS spectrum consists out of 32 classes of loads. In the analysis the number of occurrences of each class is counted for each different section of the spectrum, and compared to one another.

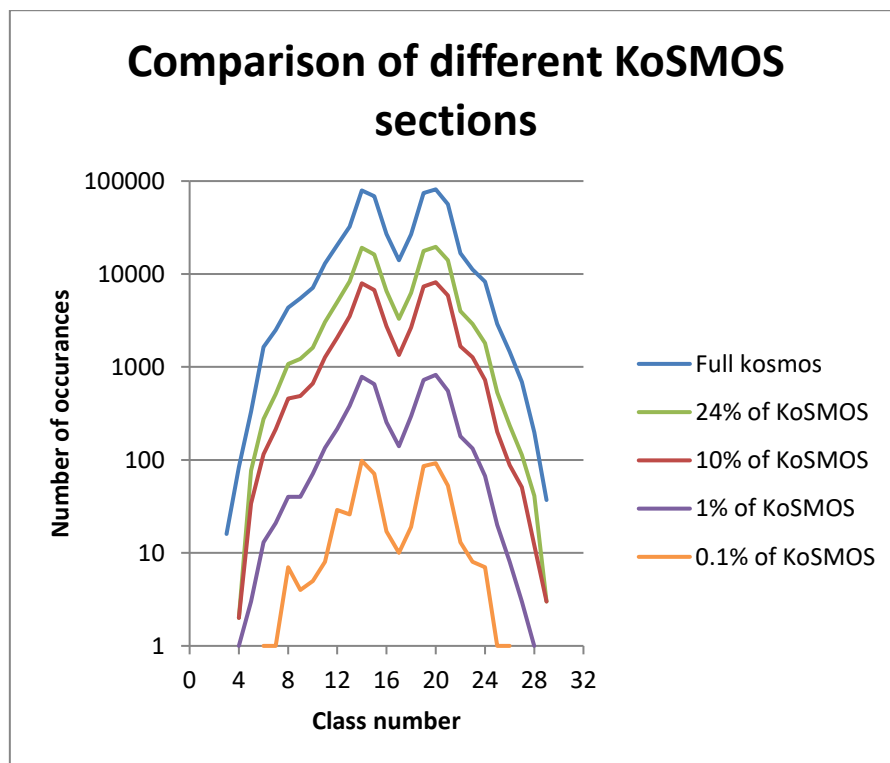


Figure 7-20: KoSMOS class count

Figure 7-20 compares the distribution of occurrences for the different sections. The figure indicates that the distribution function looks similar in shape but not in scale. This shows that a section of KoSMOS may still be representative of the entire spectrum. It is worth noting that the smaller the section becomes the less representative the distribution also becomes. This is evident when you compare the shape of the Full KoSMOS with that of the 0.1% KoSMOS. Normalising the curves of Figure 7-20 shows a very close relation to the full KoSMOS as can be seen in Figure 7-21.

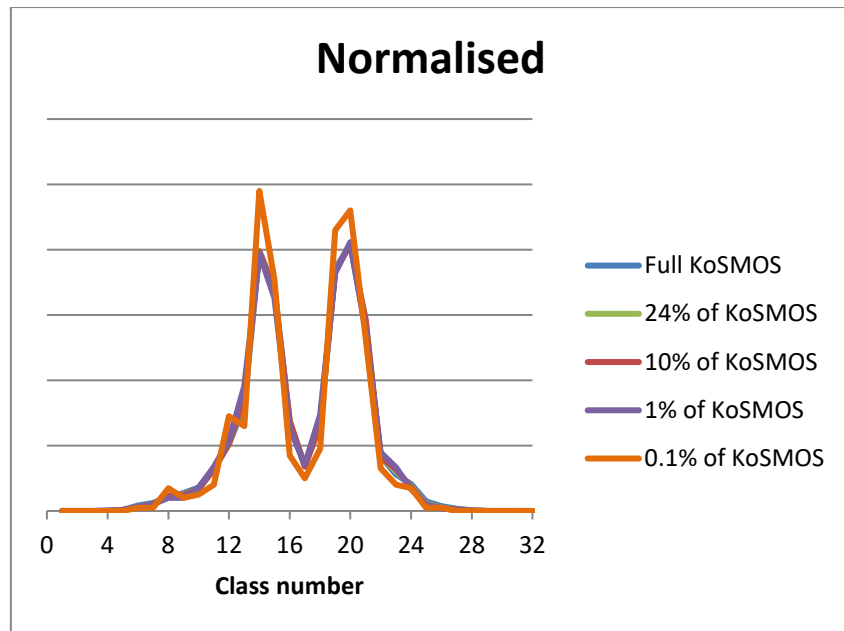


Figure 7-21: Normalised KoSMOS section class occurrences

The normalised curves in Figure 7-21 shows that all the sections except the 0.1% section follows the shape of the full spectrum distribution very well. The percentage deviation from the full spectrum distribution can be seen in Figure 7-22. In this representation one can observe that the 10% and 24% sections has excessive cycles in the high loading classes 1; 2; 30 and 31.

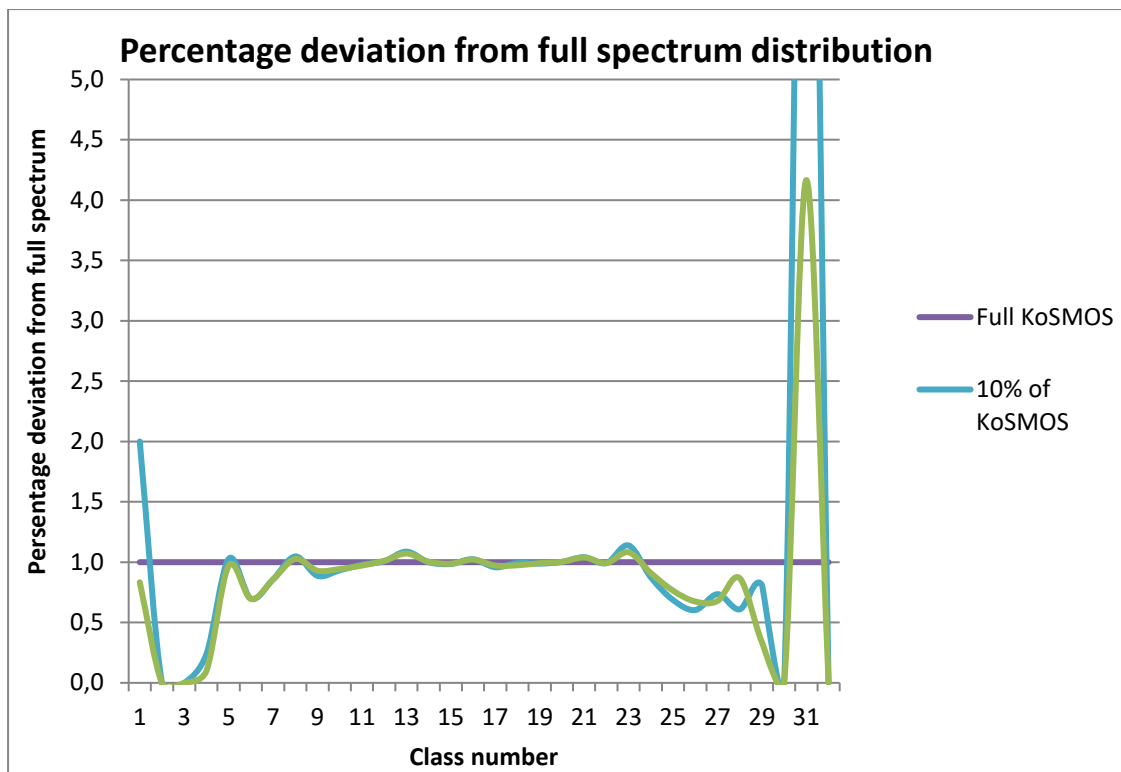


Figure 7-22: Percentage deviation

Table 7-2: Normalised occurrences of classes in different sections

Normalised occurrences of classes in different sections				
Class number	J	Full KoSMOS	24% KoSMOS	10% KoSMOS
		[556661 Cycles]	[133599 cycles]	[55666 cycles]
1	1.11	1.00	0.83	2.00
2	1.06	1.00	0.00	0.00
3	1.00	1.00	0.00	0.00
4	0.94	1.00	0.10	0.24
5	0.89	1.00	0.97	1.03
6	0.83	1.00	0.70	0.70
7	0.78	1.00	0.86	0.86
8	0.72	1.00	1.03	1.05
9	0.67	1.00	0.93	0.89
10	0.61	1.00	0.94	0.93
11	0.56	1.00	0.97	0.98
12	0.50	1.00	1.01	1.01
13	0.44	1.00	1.07	1.09
14	0.39	1.00	1.00	1.00
15	0.33	1.00	0.99	0.99
16	0.28	1.00	1.02	1.02
17	0.22	1.00	0.97	0.96
18	0.17	1.00	0.97	0.99
19	0.11	1.00	0.99	0.99
20	0.06	1.00	1.00	1.00
21	0.00	1.00	1.04	1.04
22	-0.03	1.00	0.99	0.99
23	-0.06	1.00	1.08	1.14
24	-0.09	1.00	0.91	0.87
25	-0.12	1.00	0.76	0.69
26	-0.15	1.00	0.67	0.60
27	-0.18	1.00	0.68	0.73
28	-0.21	1.00	0.87	0.61
29	-0.24	1.00	0.34	0.81
30	-0.28	1.00	0.00	0.00
31	-0.31	1.00	4.17	10.00
32	-0.34	1.00	0.00	0.00