# A CONCEPTUAL DESIGN METHODOLOGY FOR EVALUATION OF ALTERNATE PROPULSION SYSTEM MODIFICATIONS ON SMALL AIRCRAFT

A Thesis submitted by


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# Abstract

Conceptual design is often considered to be the most important step in the design of a new product or the modification of an existing product. The important steps in this conceptual design phase is the *synthesis* of potential solutions into concepts, the *evaluation* of these concepts within a repeatable and robust design methodology framework and *analysis* to identify and characterise the preferred solution concept. This research has arisen from problems associated with developing aircraft-based design modification concepts and predicting the impact of these changes as they propagate or flow down through the various aircraft subsystems, impacting engineering design, and leading to certification and operations challenges. This research problem is particularly evident in highly integrated systems such as high-performance military aircraft, helicopters, and complex civil aircraft. To illustrate this methodology the author has selected two case studies which apply two different alternate propulsion system technologies to small aircraft. These case studies were selected to provide a diverse design modification space encompassing differing aircraft roles and mission types, differing technologies and subsystems integration scope, and different data sources collection and analysis methods.

In order to combine the elements of design synthesis, evaluation of concept alternatives and analysis of outputs, this thesis has formulated a matrix-based conceptual design methodology. This methodology extends current knowledge by implementing the concepts of design synthesis, evaluation and analysis as an iterative process, and building and linking together existing techniques. This new methodology combined various techniques and methods such as Quality Function Deployment (QFD), quantified morphological matrices (QMM), Pugh's decision matrices, change options Multiple-Domain Matrices (MDM), and has adapted the Change Propagation Method (CPM).

The second extension to current knowledge in this area was the development of Engineering and Certification Domain Mapping Matrix (DMM) techniques based on Design Structure Matrices (DSM). This extension into engineering and certification domain was undertaken to ensure that important modification-related risks and costs were incorporated into the early stages of design. The extension adopted existing DSM

and DMM-based techniques and tools to evaluate the impact of changes to subsystems and hence impact of risks and costs resulting from aircraft modifications using change propagation method analysis techniques.

The validation of this conceptual design methodology was achieved by verifying and assessing the adequacy of its application through an analysis process which examined (1) coverage of the design space attributes; (2) validation of the methodology against accepted scientific and industry conceptual design frameworks; and (3) confirmation of the existing techniques, structures and tools applied within the methodology.

# Certification of Thesis

This thesis is entirely the work of Warren Williams except where otherwise acknowledged. The work is original and has not previously been submitted for any other award, except where acknowledged.

Student: Warren Raymond Williams

Principal Supervisor: Assoc Prof David Thorpe

Associate Supervisor: Dr Steven Goh

Student and supervisors' signatures of endorsement are held at the University.

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# Publications during Candidature

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# List of Abbreviations

| | |
|---|---|
| AC | Analogous Costing or Alternating Current |
| AC/CBR | Analogous Costing/ Case Based Reasoning |
| ACARE | Advisory Council for Aeronautics in Europe |
| ADO | Approved Design Organisation |
| AFSV | Automatic Flow Shutoff Valve |
| AGL | Above Ground Level |
| ASTM | American Society for Testing & Materials |
| ATA | Air Transport Association (America) |
| AVGAS | Aviation Gasoline |
| AWB | Airworthiness Bulletins |
| BHP | Brake Horsepower |
| BMS | Battery Management System |
| CAS | Calibrated Airspeed |
| CASA | Civil Aviation Safety Authority (Australia) |
| CBG | Compressed Bio Gas |
| CBR | Case Based Reasoning |
| CBS | Cost Breakdown Structure |
| CDR | Critical Design Review |
| CER | Cost Estimation Relationship |
| CFD | Computational Fluid Dynamics (aerodynamic analysis) |
| CFR | Code of Federal Regulations |
| CN | Customer Needs |
| CNG | Compressed Natural Gas |
| CO | Carbon Monoxide |
| $CO_2$ | Carbon Dioxide |
| CPM | Change Prediction Method |
| CPP | Certification Program Plan |
| DAPCA | Development and Procurement Costs Aircraft |
| DAU | Defense Acquisition University |
| DC | Direct Current |
| DDP | Design Dependent Parameter |
| DE | Detailed Estimating |
| DMM | Domain Mapping Matrix |
| DOC | Direct Operating Cost |
| DSD | Design Specification Document |
| DSM | Design Structure Matrix |
| EC | Engineering Characteristics |
| ECU | Engine Control Unit |
| EFV | Excess Flow Valve |
| EGT | Exhaust Gas Temperature |
| EM | Electric Motor |
| EPA | Environmental Protection Authority |
| EP | Electric Propulsion |
| EV | Electric Vehicle |

| | |
|---|---|
| FAA | Federal Aviation Administration (USA) |
| FAR | Federal Aviation Regulations |
| FEM | Finite Element Method (structural analysis) |
| FRACAS | Failure Reporting Analysis and Corrective Action System |
| GA | General Aviation |
| GCC | Genetic Causal Cost (cost modelling technique) |
| GGE | Gasoline Gallon Equivalent |
| GHG | Green House Gas |
| GMA | General Morphological Analysis |
| GSE | Ground Support Equipment |
| HC | Hydrocarbons |
| HOQ | House of Quality |
| HSCT | High Speed Civil Transport |
| IATA | International Air Transport Association |
| IC | Internal Combustion |
| ICAO | International Civil Aviation Organisation |
| IFR | Instrument Flight Rules |
| INCOSE | International Council on Systems Engineering |
| IPCC | Intergovernmental Panel on Climate Change |
| KCAS | Knots Calibrated Airspeed |
| KTAS | Knots True Airspeed |
| LBG | Liquefied Bio Gas |
| LCC | Life Cycle Cost |
| $LH_2$ | Liquid Hydrogen |
| LLC | Limited Liability Company (USA) |
| LNG | Liquefied Natural Gas |
| MDM | Multiple-Domain Matrix |
| MDO | Multi-Disciplinary Optimisation |
| MIT | Massachusetts Institute of Technology |
| Mod | Modification |
| MoM | Measure of Merit |
| NAA | National Airworthiness Authority |
| NASA | National Aeronautics and Space Administration |
| NFPA | National Fire Protection Association |
| NG | Natural Gas |
| NGV | Natural Gas Vehicle |
| NM | Nautical Mile |
| $NO_x$ | Nitrogen Oxides (emissions) |
| ODM | Overall Design Metric |
| OEM | Original Equipment Manufacturer |
| OT&E | Operational Test & Evaluation |
| PAFI | Piston Engine Fuels Initiative |
| PDR | Preliminary Design Review |
| PE | Parametric Estimating |
| PM | Pugh’s Matrix |
| PMA | Parts Manufacturer Authorisation |
| PPM | Parts Per Million |
| P-R | Payload-Range |

| | |
|---|---|
| PRV | Primary Relief Valve |
| QFD | Quality Function Deployment |
| QMM | Quantified Morphological Matrix |
| R&D | Research and Development |
| RDTE | Research Development Test and Evaluation |
| ROC | Rate of Climb |
| RPM | Revolutions Per Minute |
| RPT | Regular Public Transport |
| SE | Systems Engineering |
| SER | System Evaluation Report |
| SFC | Specific Fuel Consumption |
| SI | Spark Ignition (reciprocating engine) |
| SI | Système Internationale |
| SME | Subject Matter Expert |
| SOC | State of Charge |
| SOH | State of Health |
| SRR | System Requirements Review |
| SRV | Secondary Relief Valve |
| STC | Supplemental Type Certificate |
| SUGAR | Subsonic Ultra Green Aircraft Research |
| TAROC | Total Aircraft Related Operating Cost |
| TCB | Type Certification Basis |
| TCDS | Type Certification Data Sheet |
| T&E | Test and Evaluation |
| TEL | Tetra-ethyl Lead |
| TIES | Technology Identification Evaluation and Selection |
| TPM | Technical Performance Measure |
| TLR | Top Level Requirements |
| TRIZ | Theory of Inventive Machines |
| TRL | Technology Readiness Level |
| TSO | Technical Standard Order |
| UAS | Unmanned Aerial System |
| VDD | Value Driven Design |
| VFR | Visual Flight Rules |
| WBS | Work Breakdown Structure |

# List of Symbols

| | |
|---|---|
| $AF$ | Activity factor |
| $C_{ACF}$ | Cost of certification compliance finding activity |
| $C_{CD}$ | Cost of equipment development & instrumentation support for certification |
| $C_{CDL}$ | Cost of production prototype fabrication and assembly labour for certification |
| $C_{CDM}$ | Cost of production prototype material bought in instrumentation for certification |
| $C_{CDT}$ | Cost of flight/ground test operations and support for certification – specific test |
| $C_{CM}$ | Cost of certification management activity |
| $C_{CT}$ | Cost of certification test operations |
| $C_D$ | Drag coefficient |
| $C_{Dnew}$ | Drag coefficient - new |
| $C_{Dold}$ | Drag coefficient - old |
| $C_{Dtank\_A}$ | Drag coefficient wing tip tank added |
| $C_{Dtank\_R}$ | Drag coefficient wing tip tank removed |
| $\Delta C_D$ | Drag coefficient increment |
| $C_{D0}$ | Zero lift drag coefficient |
| $C_{Db}$ | Base drag coefficient |
| $C_{Df}$ | Coefficient of drag due to friction |
| $C_{DLNG}$ | Coefficient of drag - LNG fuselage belly tank |
| $C_{Dtank}$ | Drag coefficient of fuel tank |
| $C_{Dpod}$ | Drag coefficient of Slipper pod |
| $C_{Dwet}$ | Wetted area drag coefficient |
| $C_{Fb}$ | Base drag friction coefficient |
| $C_L$ | Lift coefficient |
| $C_l/C_D$ | Lift to drag ratio - coefficient |
| $C_p$ | Pressure coefficient |
| $C_{RE}$ | Cost of engineering design |
| $C_{RDL}$ | Cost of prototype fabrication and assembly labour |
| $C_{RDM}$ | Cost of prototype material and bought in components |
| $C_{RDT}$ | Cost of prototype test/operations support – specific test |
| $C_{RM}$ | Cost of engineering management |
| $C_{RT}$ | Cost of engineering development and support |
| $C_{score}$ | Normalised cost score |
| $d$ | Diameter |
| $D_p$ | Diameter of propeller |
| $D_{score}$ | Normalised installed drag score |
| $(D/q)_{cooling}$ | Cooling drag |
| $delT$ | Delta time |

| | |
|---|---|
| $E_m$ | Maximum lift to drag ratio |
| $E_{reserve}$ | Reserve energy requirement |
| $FoM$ | Figure of Merit |
| $h/x$ | Height ratio |
| $HP/W$ | Aircraft power loading |
| $J$ | Advance ratio |
| $L$ | Lift |
| $L/D$ | Lift to Drag ratio |
| $l$ | Length |
| $N$ | Propeller revolutions per second |
| $N$ | Number of engineering design activities (costs) |
| $P$ | Engine power |
| $P_0$ | Engine power at sea level |
| $R$ | Aircraft range |
| $R/C_{max}$ | Maximum rate of climb |
| $Re_l$ | Reynolds number based on length $l$ |
| $S_b$ | Base area |
| $Sigma$ | Atmospheric density ratio |
| $S_{ref}$ | Reference area of aircraft |
| $S_{ref\_wet}$ | Reference area wetted |
| $S_t$ | Wing tip tank frontal area |
| $S_{wet}$ | Wetted area |
| $T$ | Air temperature in K |
| $T_{score}$ | Normalised recharge time score |
| $T_{eng}$ | Engine thrust |
| $v$ | Velocity |
| $V_{pmin}$ | Minimum drag-power airspeed |
| $W_{begin}$ | Weight of aircraft at start of cruise |
| $W_{end}$ | Weight of aircraft at end of cruise |
| $W_{fuel}$ | Normalised score - fuel weight of each concept solution. |
| $W_{score}$ | Normalised score - NG fuel tank weight for each concept solution. |
| $W_{useful\ load}$ | Normalised useful load score |
| $\Delta(C_DS)$ | Wing tip drag coefficient increment |
| $\Delta(P_x/q)$ | Pressure differential |
| $\sigma$ | Atmospheric density ratio |
| $\rho$ | Air density |
| $\rho_0$ | Air density at sea level |
| $\mu$ | Air viscosity |

| | |
|---|---|
| $\eta_i+\Delta\eta_i$ | Actual induced propeller efficiency |
| $\eta_p$ | Propeller efficiency |
| $0.6\frac{P_R}{P}$ | Contrarotating propeller efficiency correction |

# System of Units

Whilst it is acknowledged the that Système Internationale (SI) unit system be used in all academic publications, the Imperial Unit system is used in preference to the SI unit system in some places of this document. This usage of the Imperial Unit system maintains commonality with aeronautical systems of measures as determined by the International Civil Aviation Organisation (ICAO), particularly those measures that relate to aircraft airspeed, altitude, weight, fuel loading and fuel quantity.

# Chapter 1. Introduction

*We are searching for some kind of harmony between two intangibles: a form which we have not yet designed and a context which we cannot properly describe.*

*Christopher Alexander*

## 1.1 BACKGROUND

Conceptual design of a new product or the modification of an existing product is often considered to be the most important step in design. For example, Pahl et al. (2007) states: "*In the subsequent embodiment and detail design phases it is extremely difficult to correct fundamental shortcomings in the solution principle. A lasting and successful solution is more likely to spring from the choice of the most appropriate principle than from the concentration on technical detail"*.

An integral step in the conceptual design phase is the synthesis of potential solutions into concepts to be evaluated within a repeatable and robust design methodology framework. This research is the result of the author's professional involvement in the design and certification of various military and civil aircraft system modifications and upgrades for over 30 years. Specifically, this research has arisen from problems associated with developing design modification concepts and predicting the impact of these changes as they propagate or flow down through the various subsystems and to certification requirements. This research problem is particularly evident in highly integrated systems such as military helicopters and high-performance military aircraft. To illustrate and triangulate this methodology the author has selected two case studies which apply two different propulsion system technologies. The first case study involves a modification to a small civil commuter aircraft to provide alternate aviation propulsion using clean and efficient natural gas methane fuel. This case study forms the main basis for consideration of this conceptual design methodology. The second case study involves an electric propulsion system retrofitted to a single engine normal category aircraft utilised for skydiving missions.

The implementation of upgrades, retrofits and modifications to existing aircraft, can be classified as *variant design*, as described by Otto and Wood (2001) where the size and arrangement of subsystems and components are varied within limits set by the Original Equipment Manufacturer (OEM). However, this does not totally describe the discipline that encompasses aircraft modifications, where this could also include integration of new functionality or design features. In these cases, the design process could be classified as *adaptive design*, being based on the original aircraft configuration, where known and established solution principles are adapted to changed requirements. In these examples, it may be necessary to undertake an original or new design of individual components and assemblies accompanied by new operating concepts, advanced technologies and associated support infrastructure. This embodiment of aircraft modifications is integral to the evolution of the type design as described by Harris (2001). Harris (2001) provides an example evolution by a series of modifications of the Aero Commander aircraft type, through changes to more powerful engines, streamlining of the fuselage, a fuselage stretch, extended wings, redesigned engine nacelles, pressurised fuselage, and the addition of turbo-prop engines.

### 1.1.1 Conceptual design framework applied to modifications

Given that conceptual design methodologies can be applied equally to system modifications, the systems engineering approach does not necessarily provide prescriptive guidance on the application of integrated methods, processes or tools in this early design phase. Although there are various conceptual design methodologies that have been applied to product design and the development of new aircraft types, they are not normally incorporated or applied to the conceptual design phases of aircraft modifications, retrofits, changes or upgrades. Some aircraft modifications or upgrades can be complex integration projects with system changes impacting subsystems, technical functions and performance throughout the entire lifecycle. For example, military aircraft capability upgrade programs can involve design changes through integration of new weapons, crashworthiness upgrades and changes to software within a mission system. These changes propagate down through numerous subsystems, impacting the aircraft type certification basis often requiring extensive redesign and certification effort. If these changes are not well understood then the result is significant adverse impact to technical risk, schedule and project cost. The current approach to aircraft modification or upgrade is generally based on systems engineering based trade studies, and other ad hoc design approaches. These approaches may or may not incorporate a rigorous coverage of all design options, or account for change propagation prediction, or through lifecycle support and operational impacts.

In some cases, sustainability, environmental and economic impacts are ignored in conceptual design phase. This thesis argues that a structured approach is required to enable robust conceptual design of modifications and upgrades within a framework that encompasses the entire system lifecycle. This methodology proposes a robust approach that sets out to provide (1) processes defining needs and requirements in a rigorous and repeatable construct, (2) complete structural coverage of the concept options generation process, (3) use of simple yet effective evaluation and decision making methods, processes or tools, (4) analysis methods to evaluate candidate solutions within a rigorous structured propagation framework, and (5) methods to estimate lifecycle costs and performance within sustainability and environmental constraints.

### 1.1.2 Change propagation in complex aeronautical systems

Engineering design changes are fundamental in the development of new products, particularly in the case of complex aeronautical systems where changes are made to meet new mission, role or performance needs. These engineering design changes, otherwise known as modifications or retrofits, are undertaken instead of designing or developing new aeronautical systems from scratch. Engineering changes are therefore seen to be a far more efficient means to achieve mission or performance goals without need to develop a new aircraft, helicopter or unmanned aerial systems.

Although modifications appear to be simple, all subsystems are interconnected with direct and indirect dependencies. A simple design change resulting from a modification may set off a series of other changes, transforming the initial modification into a flow-down of changes that propagates, sometimes unexpectedly through areas of the design. The degree at which this change propagates through a system depends on the complexity of the system.

Complexity in aeronautical systems is a characteristic common to many highly interconnected systems where these connections between subsystems cannot be avoided. Eckert et al. (2004) states that many industries, attempt to integrate modular architectures, with clearly defined interfaces between subsystems, to reduce the complexity of their products and to facilitate the reuse of subsystems. To further characterise complexity in systems, Eckert et al. (2004) indicates that deterministic chaos is apparent when insignificant changes to a specification lead to a considerable variation in design function and performance with accompanying increases in development risk and product costs. Further knowledge or information about the impacts of the proposed changes and solutions may be sparse or may reveal unpredicted subsystems connections. More often than not, a small alteration introduced during the conceptual design stage may result in increased costs. However, “small alterations” occurring later in the design process can have catastrophic impacts to project non-recurring costs, technical risk and schedule. Therefore, it is imperative that these engineering changes be managed adequately in the early conceptual design stage using robust and repeatable processes.

### 1.1.3 Engineering changes in context

Changes and change processes can be considered in a wider context by three areas of research and engineering management practice, comprising:

- Design studies, which in the broadest sense is concerned with the creation of new products, as stated by Cross (1994).
- Design reuse, which is concerned with the reuse of existing concepts, systems, subsystems and components. This research is represented in research literature where it is dealt with in different disciplines as detailed by Eckert et al. (2001). This field of design reuse within a conceptual design framework is the research area under consideration in this thesis.
- Configuration management, which is concerned about managing and controlling changes on systems, subsystems and components, ensuring that functions and interfaces are maintained over the life-cycle over the system.

In any engineering organisation, engineering changes are usually managed at two levels, where changes in configuration are managed by a configuration board, and design change impacts are managed within the project team within the limits of an approval process by the configuration board. In this thesis, the approach is to further research and extend research into intrinsic problems associated with changes made to highly connected complex aeronautical systems bounded by airworthiness regulations, sustainability and performance requirements.

### 1.1.4 Sustainability in aviation

The impact of sustainability and environmental issues in aviation has become an important issue in more recent times. The Advisory Council for Aeronautics in Europe (ACARE, 2004) have set a research agenda which states that aviation is at a sustainability cross-road where considerable research resources will be applied to alternate fuels and propulsion technologies. This is described as the third age of aviation, where the sector is approaching a phase of sustainable growth requiring cost effective, clean, quiet, safe and secure air travel. As stated by IATA (2015), this phase is characterised by the increase in aviation traffic and by the transformation of travel behaviour and core values of passengers. Eres et al. (2014) state that this requires a fundamental change in the way engineering design is initiated to cater for the upward demands for air travel and the broader economic and environmental needs of society. It is generally accepted that significant improvements to the environmental acceptability of aircraft will be needed if the long-term growth of air transport is to be sustained.

## 1.2 PROBLEM DEFINITION

This thesis sets out to investigate and formulate a conceptual design methodology to evaluate aircraft modifications (otherwise known as design changes) within a systems life-cycle framework. This research will formulate a methodology that can be used to generate viable design change concepts, quantify the impact of design change decisions, assess change propagation through the system and conduct performance evaluation within a value-based framework throughout the lifecycle covering research, development, modification, testing and certification, and operations. In order to achieve this, new techniques and approaches will be required to evaluate design modification variable interactions in this complex systems space using structured techniques supported by aircraft design and analysis tools and software. Given that this thesis will involve sub-problems, then the research space will be characterised by the unique combination of these parameters to determine the value and sustainable proposition for the aircraft system/modification life-cycle.

These research sub-problems are posed as questions within the framework as described below.

### 1.2.1 Systems design

- What lifecycle-based systems design processes, approaches or techniques can be adapted or developed to manage design changes in aeronautical systems?

### 1.2.2 Synthesis of design changes

- What methods, tools or techniques can be applied to elicit, document, derive and prioritise design change needs and requirements?
- Can synthesis and concept generation methodologies provide comprehensive and structured coverage of the design space to provide robust alternatives?

### 1.2.3 Evaluation of concept alternatives

- What methods, tools or techniques can be applied to down-select design change alternatives?
- What methods can be incorporated to evaluate the impact of design change propagation throughout the system?
- What methods or tools can be incorporated to characterise engineering design, certification, operations and sustainability impacts of the design change?

### 1.2.4 Analysis and outputs

- How can simplified models and tools be used to evaluate aircraft performance, life-cycle costs, technical risks and sustainability?

## 1.3 SIGNIFICANCE

### 1.3.1 Overview

The main application of this conceptual design methodology is to a range of aircraft modification projects involving low emission alternate propulsion systems such as those given by electric aircraft retrofit modifications. The aircraft modification process associated with retrofitting electric propulsion technologies is also characterised by a multi-dimensional design space covering aircraft performance, design configuration options, operating modes, sustainability, and lifecycle cost factors.

It is possible that this conceptual design methodology could be extended to other projects that involve modifications or upgrades to complex systems. Military aircraft are often subject capability upgrades during their operating lives, and this often includes hardware and software changes which need to be integrated, tested, certificated, and supported. These modifications and upgrades are designed to enhance the function and performance of the platform with development costs comprising many millions of dollars, and life-cycle costs many times the development costs.

Therefore, a conceptual design methodology applied to modification projects can assist in making defensible design decisions within a robust and structured integrated life-cycle framework using appropriate tools and models. Although not a major focus within military systems, sustainability and environmental issues may become an increasing issue, particularly in those areas that might impact platform operating and support costs. It is therefore important that sustainability requirements be included in any design methodology framework.

### 1.3.2 Low emission alternative aviation fuels and propulsion systems

It is accepted that improvements to the environmental sustainability of air transportation will be needed to maintain long-term growth. The aviation industry is at a cross road where aviation research is driving to develop and integrate alternative fuels and propulsion systems to reduce emissions and impact to the environment. Considerable resources have been applied to electric propulsion system research and development, and how the longer-term impacts of these systems may affect the environment, even though harmful emissions might be reduced to zero. Liquefied Natural Gas (LNG) is a cryogenic fluid which could play an important part in electric-hybrid aircraft where a thermal energy source may be required to generate power for electric fan propulsion systems. In this instance, high electric power introduces high thermal loads on electrical conductors, inverters, regulators etc. which impacts efficiency of these systems. A potential solution considered by researchers Roberts & Wolff (2015) is the application of LNG as an aviation fuel, as well as acting as the cryogenic coolant for on board electrical and environmental control systems.

## 1.4 CASE STUDIES

Two case studies are presented in this thesis as a means to triangulate and validate the conceptual design methodology applied to aircraft design changes or modifications. These two case studies involve modifications to small General Aviation (GA) aircraft to provide cleaner and sustainable fuel or propulsion options. The first case study involves a modification to a small commuter category aircraft to incorporate natural gas fuels. The second involves a modification to a small single engine aircraft to replace the existing Internal Combustion (IC) reciprocating engine with an electric propulsion system.

### 1.4.1 Natural gas fuel modification

Small twin-engine aircraft are used worldwide as feeder airliners and commuter aircraft, and range in size from 8 through to 10 passenger seats. These aircraft are typically used to transport passengers and freight to regional centres usually in remote locations, and are powered by reciprocating engines. These reciprocating engines use leaded AVGAS (aviation gasoline) as fuel. An example of these aircraft includes the Cessna 400 types as shown by Figure 1. These aircraft fall within the weight and performance category associated with the FAR Part 23 airworthiness certification regulations. Given that these commuter transport aircraft are ageing (Gauntlett et al., 2010), and there are no immediate replacements, then structurally refurbishment or upgrades may be required in the near future to ensure continued operations. The structural upgrades of these commuter aircraft also present an industry opportunity to integrate “greener” alternate fuel technologies (replacing AVGAS fuels) as part of a continued airworthiness modification program. These alternate fuel technologies can benefit this sector of the air transportation market by reducing fuel-related operating costs and harmful AVGAS emissions as a result of toxic lead fuel additive. This case study applies the conceptual design methodology to a commuter aircraft fuel modification utilising natural gas fuels.

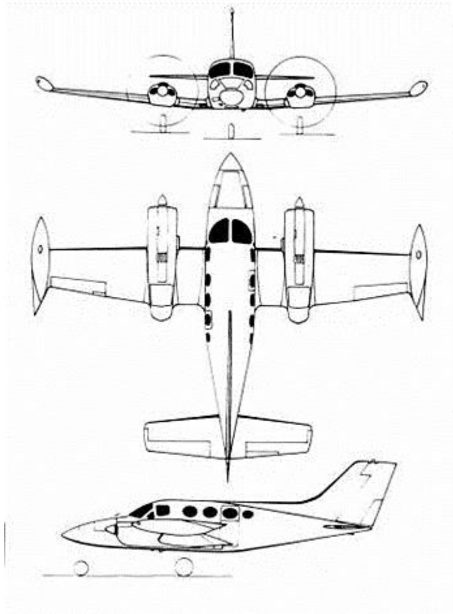

Figure 1. Typical small normal category commuter aircraft
Cessna Aircraft Company (1974)

#### *1.4.1.1 Natural gas fuels*

Natural Gas has been used as an alternative fuel for automotive and heavy vehicle transportation for many years. It is also well established as a primary energy source for domestic and commercial heating and large-scale electricity generation (Astbury, 2008). As the bulk density of natural gas is very low, it is necessary to store it either as Compressed Natural Gas (CNG), or in the liquefied form as Liquefied Natural Gas (LNG) to make it practical to use for transport. Biogas is another form of natural gas produced from the anaerobic digestion of sewage which can be purified to about 95% methane for use in engines for electricity generation (United Utilities, 2007). In addition, biogas can be used in the compressed form as compressed biogas (CBG), or in the Liquefied Biogas (LBG) form. As described by Yang et al. (2014), biogas has the potential to be a true renewable resource, where applications of CBG and LBG are the same as CNG and LNG.

#### *1.4.1.2 AVGAS emissions*

AVGAS is an aviation fuel used in reciprocating internal-combustion engines such as commuter aircraft. Unlike motor gasoline, AVGAS continues to use tetraethyl lead (TEL) additive, which is a highly toxic substance used to prevent engine knocking (detonation). TEL was phased out of use in automotive applications in most countries in the late 1990’s.

Scientific research has shown that TEL found in leaded AVGAS and its combustion products are potent neurotoxins that interfere with brain development in children. The United States Environmental Protection Agency (EPA), 2008 has noted that exposure to very low levels of lead contamination has been conclusively linked to loss of IQ in children's brain function tests. This therefore provides a high degree of motivation to eliminate lead and its compounds from the environment.

In addition to the highly toxic TEL emissions, AVGAS also contributes to $CO_2$, $NO_x$, CO and other particulate emissions, in much the same concentrations as other internal combustion engines. However, unlike other internal engines used in ground transportation, aircraft engines do not incorporate catalytic converters which act to remove harmful $NO_x$ and CO emissions. Although emissions provided by AVGAS fuelled commuter category aircraft are small in proportion to other transportation modes, the TEL emissions are extremely harmful to society as described above.

### 1.4.2 Electric propulsion system modification

The use of electrical power as a means of propulsion for aircraft is not a new concept. Over the last two decades, several key technologies for aviation electric propulsion systems have matured to the extent where the power and energy per weight ratio has become suitable for specific applications and missions. The concept of integrating electric propulsion systems has received recent attention for thin-haul commuter and on-demand transportation as reported by numerous researchers including Moore & Fredericks, (2014), Patterson et al. (2012) and Stoll & Mikic (2016).

One mission that lends itself to electric propulsion systems is skydiving. Skydiving, also known as parachuting, is a popular aviation sport throughout the world (Glassock et al., 2017). While the overall number of participants is relatively small compared to many other usual sports, there are hundreds of thousands of active skydivers operating from approximately 1000 centres worldwide, as reported by Dropzone Inc. (2018). The United States Parachute Association (2014) alone recorded 36,770 members at the end of 2014.

As compared to military parachuting operations, sports skydiving typically requires commercial operators or clubs to utilise converted general aviation and small commuter type aircraft to get participants to the required altitude, typically up to

4000m Above Ground Level (AGL). The types in use for commercial operations range from 4-seat to 10-seat aircraft, which include light aircraft and commuter category types. Examples of aircraft types used include Cessna Model 182, Cessna Model 206, Piper PA-31, Pilatus PC-6, PC-12, and Twin Otter types, or similar piston or turbo-prop powered aircraft (Glassock et al., 2017). As with any aircraft operation, the commercial viability of any choice is dependent on many factors such as the demand and utilisation, operating costs (maintenance, fuel), and insurance as described by Glesk (2018, pers. comm., 13 June).

However, it should be noted that aircraft that incorporate battery-electric propulsion systems have significantly different characteristics, such as no power loss with altitude and no weight loss through fuel burn, contrasting conventional reciprocating and turbine engine design assumptions (Patterson et al., 2012). Additionally, there are no engine shock cooling limitations that are applicable to reciprocating engines operating under high power settings and then transitioning to low power and low airspeed settings for descent. As stated by Glassock et al. (2017) this shock cooling limitation is applicable to air-cooled internal combustion engines where damage can occur to cylinder heads during high duty cycle flight profiles used in skydiving operations. Additionally, these missions that involve high descent rates after a climb to altitude can result in engine over-speed damage due to a wind-milling propeller. This damage must be avoided by limiting the descent rate and airspeed, which impacts the cost-economics of skydiving.

However, electric aircraft are not without disadvantages, as there are airworthiness regulatory challenges involved in the practical adoption of fully electric aircraft into the air transportation system. One major challenge is the development of industry consensus standards specifically for electric aircraft (Patterson et al., 2012). This requires the aviation industry to collaborate and agree on evolving electric technology, configuration architectures and associated airworthiness requirements.

In addition to regulatory issues, there are also technical challenges associated with fully electric aircraft. As described by Patterson et al. (2012) the specific energy density of batteries are currently one to two orders of magnitude lower than that of conventional aviation fuels. Furthermore, Patterson et al. (2012) state that while the weight of electric motors is considerably less than that of internal combustion engines and the efficiency of electric motors is considerably higher, the low specific energy density of the batteries currently leads to much higher total installed weight for the equivalent amount of energy.

The Directorate-General for Mobility and Transport (2011) in Europe states that the utilisation of renewable or green energy, will reduce the carbon footprint and offer reduced participation costs for skydiving as a sport. Rather than the development of a specific aircraft type for skydiving missions, modification of existing aircraft may be a cost-effective alternative at least in the short term as these propulsion technologies mature. This case study therefore considers an electric/ modification of a typical normal category light aircraft, as shown by Figure 2. This modification will rely on a firewall forward system installation comprising the battery-electric propulsion system.

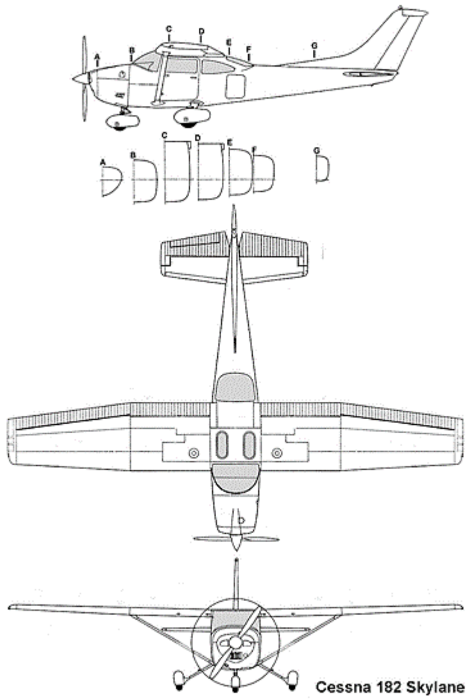


Figure 2. Typical normal category light aircraft
Cessna Aircraft Company (1976)

## 1.5 DELIMITATIONS

This thesis will confine its research to a conceptual design methodology associated with integration of alternate fuel and propulsion technologies on small general aviation category aircraft and associated technical and operational interfaces only. This delimitation is implemented to restrict this research to an appropriate and practical system boundary. This means that the system boundary will be restricted to the aircraft platform and those direct interfaces such as ground support infrastructure and support systems. What is not included is a detailed assessment of ground support infrastructure solution options, or detailed aircraft system life-cycle costs and performance characteristics. Rather this research will focus on the conceptual design methodology within this aircraft system boundary, noting that the methodology could be equally applied to the other side of the boundary, once an established alternate fuel system candidate solution output is defined. Therefore, it is the aim of this research to establish a general conceptual design methodology that can be equally applied or adapted to other aircraft systems design change or modification projects.

## 1.6 THESIS OUTLINE

This thesis is structured so as to present the data, arguments and supporting evidence as follows:

- Chapter 1. Introduction – This Chapter outlines the background, context, purpose, significance and statement of the research problem.
- Chapter 2. State of the Art– This Chapter provides a review of literature which underpins this research problem.
- Chapter 3. Design Modification Space – This Chapter provides the background design space context associated with this research problem.
- Chapter 4. Formulation – This Chapter describes the formulation of the conceptual design and evaluation methodology including underpinning mathematical concepts.
- Chapter 5. Implementation and validation – This Chapter describes implementation of the conceptual design methodology to two case studies, and provides the substantiation supporting validation of the methodology.
- Chapter 6. Conclusions. – This Chapter provides concluding remarks, comments and recommendations associated with this research problem.
- References – This section provides a list of references associated with this research problem.
- Appendix 1. Natural Gas fuels – Case study

- Appendix 2. Natural Gas fuels case study– Supporting analysis
- Appendix 3. Electric Propulsion system – Case study

In addition, Figure 3 illustrates the thesis outline including relationships between Chapters and Appendices and the Research concepts.

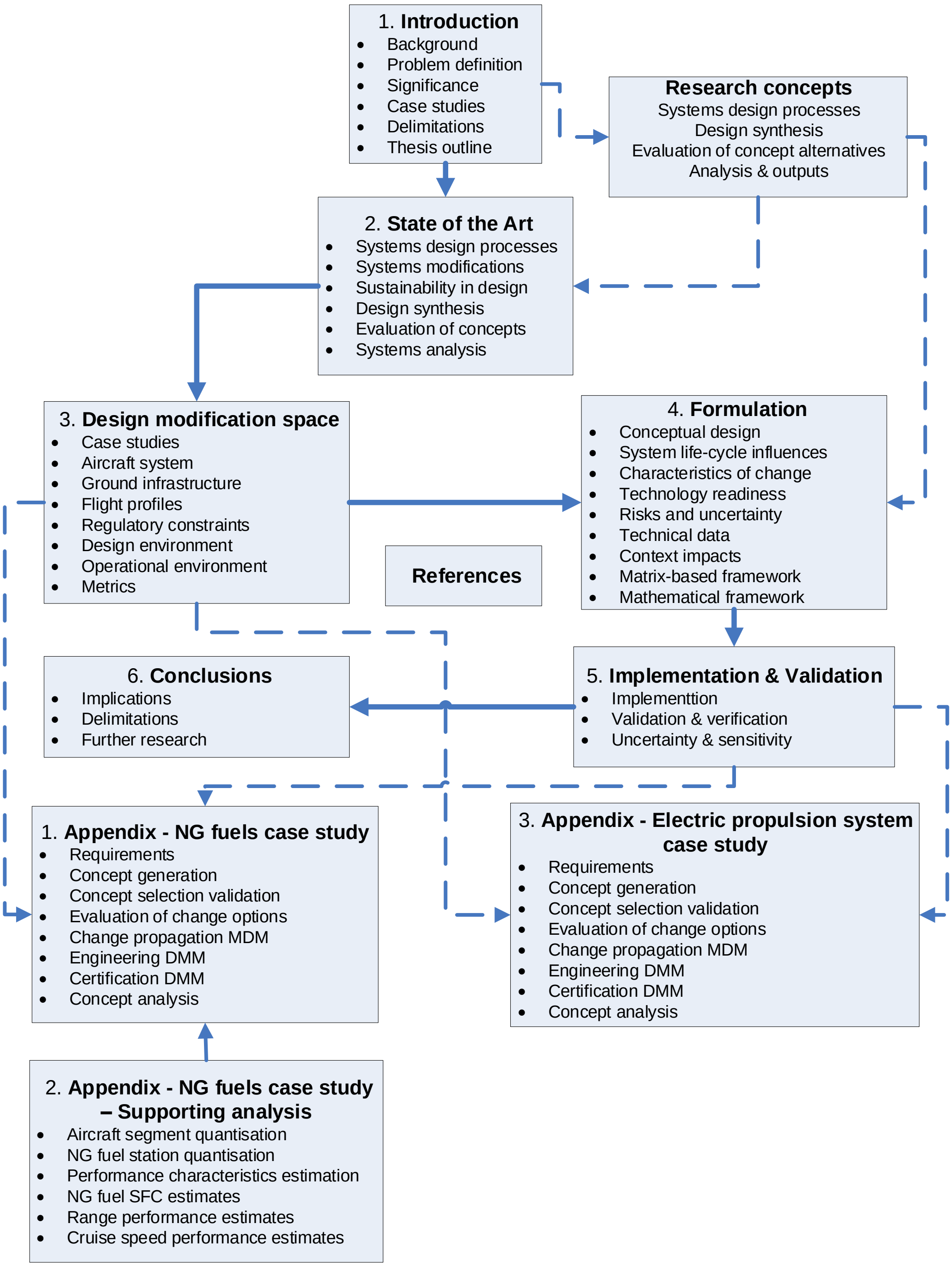


Figure 3. Thesis outline

# Chapter 2. State of the Art

*Design is an iterative process. The necessary number of iterations is one more than the number currently done. This is true at any point in time.*

*John R. Page*

## 2.1 OVERVIEW

### 2.1.1 Design methodology

Pahl et al. (2007) defines a "*Design methodology* as a concrete course of action for the design of technical systems that derives its knowledge from design science, cognitive psychology and from practical experience in different domains". "Design methodologies are developed to guide the abilities of designers, promote creativity, and at the same time emphasise the need for objective evaluation of the results and options".

The development of design methodologies makes it easier to re-establish viable solutions from previous projects, and to re-use past design data to support other *adaptive design* projects. Systematic design processes as described by Pahl et al. (2007) can be an effective way to rationalise the design, which can be ordered and stepwise, and can provide solution that can be re-used, or applied again.

A design methodology is also a pre-requisite for the implementation of a computer-based design process using product design models or numerical simulation. Without this design methodology, it is not possible to develop the knowledge-based systems; access stored data, and link separate programs and data from different company entities. Systematic design methodologies facilitate division of the work between designers and computers in a logical way.

The history of traditional design methodologies is extensively reviewed by Pahl et al. (2007) with the focus being on technical artefacts. Pahl et al. (2007) goes on to state that design methods have been developed on the basis that technical artefacts can be represented as systems. This then led to the application of systems theory, which related socio-economic-technical processes, design procedures and methods. It is this systems theory that underpins good design methods that reflects the requirement to tackle complex problems in fixed steps, involving analysis and synthesis under what could be referred to as a systems approach. These engineering design methods are characterised as several steps that firstly involve acquiring information about the system needs through market analyses and trend studies, and can be referred to as problem or requirements analysis. The second step involves formal statement of the system goals, with these goals being the criteria for subsequent evaluation of the solution concepts, and hence identification of the optimum solution. Before the solution concepts can be evaluated, the performance of each must be predicted. In the evaluation, the performance is compared with the original requirements or goals, and on this basis a decision is made, and the best candidate is selected. In a systems theory process these steps repeat themselves in a life-cycle phase which progresses from abstract through to concrete solutions. This process of analysis, synthesis and evaluation occurs in an iterative fashion within these steps.

Systems theory in design has been implemented in several contexts with variations to account for the technical artefacts concerned. In the following sections systematic design methods can be described by various approaches and frameworks that have been adapted for specific artefacts or products.

### 2.1.2 Conceptual design

Blanchard and Fabrycki (1998) state that conceptual design process progresses from an identified need, to the definition of the system requirements in functional terms, establishment of the design criteria, and then the development of a system specification. The accomplishment of a feasibility analysis is a major step within conceptual design that involves three main steps. These steps being (1) identification of possible design approaches, (2) evaluation of these approaches based on performance, effectiveness, maintenance, logistic support, and cost economics, and (3) a recommendation of the preferred course of action. In addition, considerations are given to applications of different technologies as part of the design approach. The system requirements analysis steps within this process involve definition of requirements (operational, maintenance and support), provision of Technical Performance Measures (TPMs), functional analysis allocation and design synthesis and evaluation. These TPMs are described by Blanchard and Fabrycki (1998) as the quantitative factors or metrics associated with the system under development.

### 2.1.3 Systems engineering

The Defense Acquisition University (DAU) (2001) states that Systems Engineering (SE) “is an all-encompassing, iterative and recursive problem-solving process, applied sequentially in a top-down approach”. It transforms customer needs and requirements into a set of system functional and performance descriptions, generates information for decision making, and provides output for the next design and development phase. The process is applied sequentially by adding additional detail and definition with completion each phase of development. Several definitions of SE exist, and various handbooks provided by International Council on Systems Engineering (INCOSE), (2010) DAU (2001), and National Aeronautics and Space Administration (NASA) (2007) provide aligning and consistent definitions and source information for practitioners in this discipline. SE provides a framework for system development life-cycle processes which can be adopted also for modification of existing systems or upgrades. The focus in this thesis is how SE processes accommodate the early life-cycle design stages, and how these processes are defined in the early conceptual design stage. SE analysis methods, principles and techniques have been extensively documented in various references including the Handbooks referenced earlier. In order to establish the basis for application or adoption of SE, various references have been selected as representative of this systems discipline knowledge and thus provide a range of design techniques and methodologies, built upon by this thesis. These references include works by Blanchard and Fabrycki (1998), Faulconbridge and Ryan (2014), Kossiakoff and Sweet (2003) and Price et al. (2006) the latter providing a comprehensive paper outlining the challenges specifically associated with innovative integrated approaches to SE life-cycle design of aircraft.

### 2.1.4 Product design

Product design or the mechanical design processes are those that also can be applied to complex systems. Numerous authors have contributed to this field, with research in this field dating back to Redtenbacher (1852), as cited in Pahl et al. (2007), where the principles of machine design became increasingly focussed on mechanical design characteristics and a more structured approach to design development. More recent accounts of contemporary methods and approaches can be found in texts by Pahl et al. (2007), Ullman (2010) and Ulrich & Eppinger (2012).

#### 2.1.4.1 *Product lifecycle*

Pahl et al. (2007) provides an account of engineering design theory in relation to a product life-cycle. An extensive account is provided in relation to systems theory, design methods including systematic solution finding processes, and detail of steps in the product planning and design life-cycle which is summarised in steps shown in Figure 4. It is noted by Pahl et al. (2007) that the steps do not lead to the final solution, but often require an iterative approach, with these steps repeating as part of system life-cycle phases.

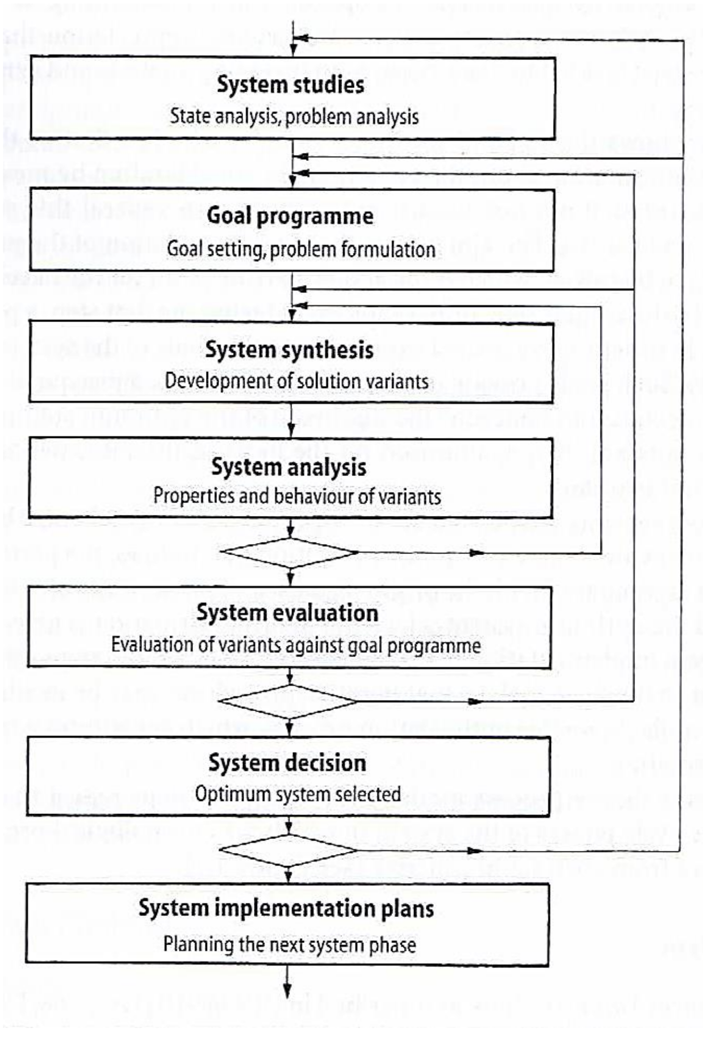


Figure 4. Steps of a product planning life-cycle
Pahl et al. (2007)

#### 2.1.4.2 *Concept development*

Ulrich & Eppinger (2012) describe concept development as front-end process which generally involves several inter-related steps which are usually repeated in an iterative approach. These phases occur sequentially albeit approached iteratively as illustrated in Figure 5.

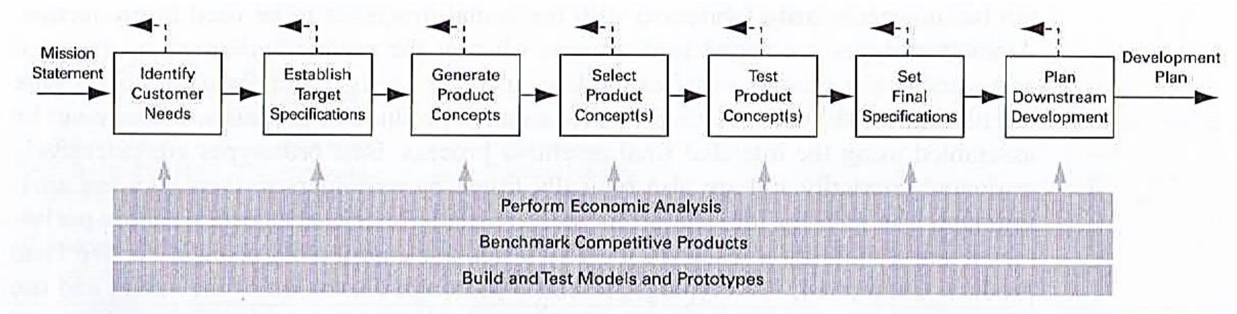


Figure 5. Front end concept development activities
Ulrich & Eppinger (2012)

### 2.1.5 Mechanical design processes

Ullman (2010) provides an account of the mechanical design process as applied to conceptual design and product design. Unlike earlier descriptions of the design process, Ullman (2010) describes the mechanical design lifecycle within a framework comprising sequential steps involving (1) Product discovery, (2) Project planning, (3) Product definition, (4) Conceptual design, (5) Product development, and (6) Product support. This process is said to apply to design from systems level down to component level. It is also said to apply to the design of new systems as well as modifications to existing products. Although the terminology differs from that presented by Pahl et al. (2007) and Ulrich & Eppinger (2012) the underpinning principles remain the same, starting with a definition of product needs, leading through product planning, definition, conceptual design and development, and finishing with product support.

### 2.1.6 General aircraft design

An aircraft is an example of a highly complex system comprising many interrelated subsystems and components. There has been much written in relation to aircraft configuration design processes and the conceptual design phase. This work is encapsulated in various aircraft design texts by Raymer (2012), Torenbeek (2013, 1982), Roskam (2002), Takahashi (2016), and Kundu (2014) which are summarised here.

Raymer (2012) describes *Design* as “Creating the geometric definition of a thing to be built”. In the aeronautical context, aircraft design is about establishing the configuration through an iterative process moving to more sophisticated solutions using more sophisticated methods of analysis as the design progresses. Indeed Raymer (2012) states that the initial design process starts with a concept of the configuration under consideration, provision of initial weight estimates, and the requirements; often occurring simultaneously. Apart from the requirements, which should be independent of the design process, bi-directional interactions occur at all levels of the design process. It is noted that this aircraft design methodology is applicable also for design changes or modifications, where the same processes could be utilised.

#### *2.1.6.1 Requirements*

The start of the general aircraft design process is encapsulated in the requirements which Raymer (2012) has classified into three broad areas as follows:

> Top level requirements – Top-level requirements define the purpose of the new aircraft design concept, the assumptions regarding the customer and ultimate operator, including also the level of acceptance of technological risk, and cost targets.
>
> Customer centric requirements – These requirements define the mission and payload capabilities including such parameters as payload, range, cruise speed/altitude, low speed performance, airport compatibility, reliability and environmental issues.
>
> Legalistic requirements - These legalistic requirements are those that are defined by civil or military design specifications or airworthiness regulations.

#### *2.1.6.2 Design phases*

Raymer (2012) suggests that aircraft design can be divided into three major phases as (1) Conceptual design, (2) Preliminary design and (3) Detail design. The design process begins with Conceptual design, where a broad range of aircraft configuration concepts are explored, trade studies are performed of both the designs and the requirements, and ultimately settle on the best design with inputs from the customer to develop a set of well-balanced requirements. Preliminary design can be said to occur when the major changes made in the conceptual design phase are completed. Detailed design is carried out assuming a favourable decision to enter full-scale development, as described by Raymer (2012).

#### *2.1.6.3 Conceptual design*

As stated above conceptual design starts with the requirements. Raymer (2012) notes that before any new design, a decision must be made as to what technologies will be incorporated. An overly optimistic estimate of technology utilisation will provide a lighter, cheaper aircraft to perform a given mission, but will result in higher developmental risk. Conversely, usage of mature technology may result in a heavy under-performing aircraft. To clarify this technology readiness, terminology has been developed by NASA and the US Department of Defense Guidebook (2010) which is referred to as Technology Readiness Levels (TRLs), as described also by Ullman (2010).

Raymer (2012) states that early aircraft conceptual design usually "starts with a conceptual sketch of the overall aircraft configuration, which provides a rough indication of the design layout including the approximate wing and tail geometries, fuselage shape and internal locations of major components". This process is called initial "sizing". Optimisation techniques are then used to determine the lightest and lowest cost solution that will perform the mission and meet all performance requirements. The process then develops a revised layout and following a more detailed analysis and refined sizing and optimisation.

Torenbeek (1982) provides a generalised iterative design process shown in Figure 6. The principal steps are initiated by the requirements, formulation of a trial configuration, conduct of analyses, requirements comparison, configuration changes, and design optimisation. Torenbeek (1982) noted that this generalised design process could be applied equally well to the design of products, as well as aircraft. This general design process includes a convergence test to indicate those situations or cases where no configuration solution satisfies all requirements simultaneously, due to certain requirements in the specification, and constraints being contradictory or too extreme with respect to current technologies.

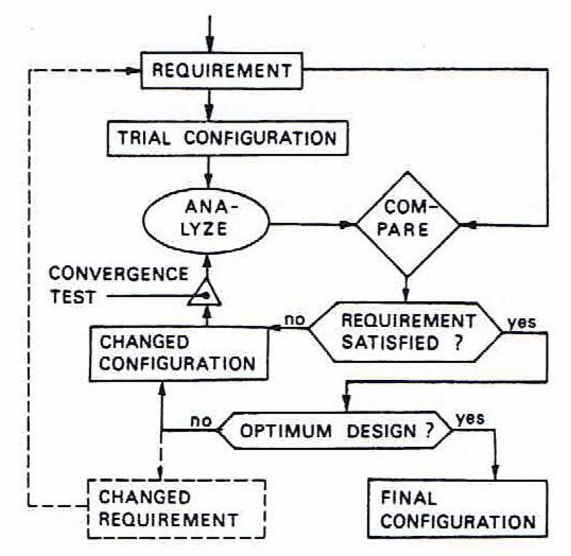


Figure 6. Generalised aircraft design process
Torenbeek (1982)

### 2.1.7 Integrated approaches

#### *2.1.7.1 Aircraft design*

The paper by Price et al. (2006) describes the challenges associated with introducing innovative integrated effective and efficient approaches to Systems Engineering (SE) life-cycle design of aircraft. This paper was driven by Vision 2020, and was related to the ACARE (2004) research agenda with environmental targets to reduce $NO_x$ emissions, CO and $CO_2$ emissions, noise, and to minimise costs and improve aviation safety. SE in this context was described by Price et al., (2006) as a holistic life-cycle-based process applied to the development of a system that comprises subsystems and components. This process starts with customer specifications, and progresses through the system life-cycle starting with conceptual design, functional analysis and architecture, physical architecture, design synthesis, risk analysis, trade studies and optimisation, production, verification and validation, life-cycle costing and project management. The paper by Price et al., (2006) identifies several issues regarding the SE approaches applied to aircraft design. These included lack of basic scientific practical models and tools for interfacing and integrating components of the SE process within a given component, and details in relation to consideration of costs and manufacturing trade-offs within an integrated design environment. Furthermore, Price et al., (2006) details the challenges associated multidisciplinary optimisation and integration within this environment, data flows between one analysis to another, and the low fidelity models applied in early stages to later stages with increasing fidelity.

Price et al. (2006) state that early design life-cycle models usually comprise low fidelity simple equations, look up tables with no associated geometry; medium fidelity models incorporate some form of linear analyses, while later design life-cycle models incorporate considerable detail and use high level tools such as Computational Fluid Dynamics (CFD) and Finite Element Methods (FEM) analyses capabilities.

#### *2.1.7.2 Cost modelling*

Price et al. (2006) highlights the importance of the integration of costs when applying SE to engineering design processes. In this context, emergent behaviour or performance of a design can also manifest itself to costs. This emergence may result in undesired costs as a function of interactions between subsystems. Price et al. (2006) describes a strategy that leads to a methodology to identify, control and manage system development to produce most cost effective, predictable system designs thus eliminating emergent behaviour.

Various alternative approaches to cost modelling exist, and are described later in this thesis. The bottom-up or Detailed Estimation (DE) technique involves collating all relevant cost information directly attributable to the final subsystem or component. Parametric Estimating (PE) typically involves linking of cost to high-level product parameters through probabilistic analysis to establish estimating relationships that can be combined into a cost estimating model. Price et al. (2006) also detail Analogous Costing (AC) and Case Based Reasoning (CBR) or (AC/CBR) techniques that mainly rely on similarity and differentiation with other products, to provide comparative cost estimates. However, Price et al. (2006) states that costing methodologies should be able to operate at the interfaces and at various levels and all stages of the design life-cycle. As stated by Price et al. (2006), this is main concept underpinning the development of the Genetic Causal Costing (GCC) technique, described later.

Table 1 illustrates the cost estimating techniques for various life-cycle phases as described by Price et al. (2006).

Table 1. Cost estimating techniques

| | Techniques: | | | | | |
|---|---|---|---|---|---|---|
| Life cycle phase: | DE | AC/CBR | NN/FL | PE | ABC | GCC |
| Concept and technical development | ✓ | ✓ | | ✓ | | ✓ |
| System development and demo | ✓ | ✓ | ✓ | ✓ | ✓ | ✓ |
| Production and acquisition | ✓ | ✓ | ✓ | ✓ | ✓ | ✓ |
| Operations and disposal | | | | ✓ | | ✓ |

Price et al. (2006)

### 2.1.8 Technology, Identification Evaluation and Selection

Other work in this area that can be considered as integrated approaches, has been that undertaken by Georgia Institute of Technology. Georgia Institute of Technology has applied value-based statistical methods to commercial aircraft conceptual and preliminary design mostly in relation to High Speed Civil Transport (HSCT) studies. The papers by DeLaurentis, Mavris & Schrage (1996), DeLaurentis et al. (1997), Kirby & Mavris (1999, 2001), Marx, Mavris & Schrage (1995), Mavris & DeLaurentis (1998), Mavris et al. (1998, 1999), Mavris & Bandte (1997), Mavris & Kirby (1999), Mavris, Mantis & Kirby (1997) address stochastic and probabilistic methods to evaluate value within the aircraft development and operational life-cycle. Indeed, two papers by Kirby & Mavris (1999, 2001), and a doctoral thesis by Kirby (2001) outline a method that manages Research and Development (R&D) and technology inputs using various approaches including Technology, Identification Evaluation and Selection (TIES). An example of this TIES approach is provided by Figure 7. This method provides a useful starting point for the study as proposed here noting that this research may be adapted to a specific aircraft modification concept.

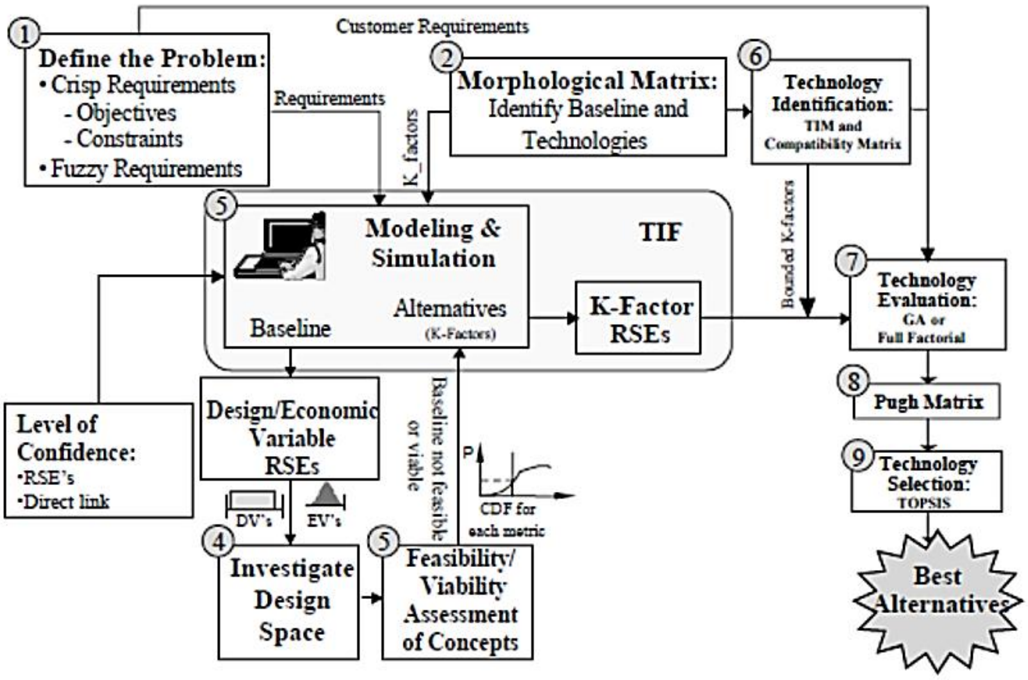


Figure 7. Technology Identification Evaluation and Selection Method
Mavris & Kirby (1999)

## 2.2 SYSTEM MODIFICATIONS

Kossiakoff and Sweet (2003) discuss the effect of technological improvements and obsolescence of complex systems and how this results in an opportunity to improve the system by modifications, design changes or upgrades. These modifications, design changes, or retro-fits are undertaken to restore the overall system effectiveness by replacing subsystems and components at a fraction of the cost of the total system. Such a modification is referred to as a system upgrade. Military aircraft generally undergo upgrades or modifications during their operating life, with avionics, weapons systems and propulsion systems being popular subsystems to be replaced. In the civil aviation field, it is often economical and advantageous to replace obsolete avionics or passenger entertainment systems as an upgrade program.

### 2.2.1 Modification lifecycle

Kossiakoff and Sweet (2003) state that the development, manufacture and installation of a system upgrade or modification can be considered to have a systems development life-cycle of its own, with phases like, or the same as those of the main life-cycle. In this case SE is also applicable to these upgrades and modifications.

In this context, the concept development stage described by Kossiakoff and Sweet (2003) begins with recognition of a need for a major capability improvement to address mission or economic deficiencies in the current system. The concept exploration stage starts with a process which compares several options of upgrading a subsystem with a total replacement by a new or superior subsystem. Kossiakoff and Sweet (2003) indicate that a convincing need for a limited system upgrade or modification, will lead to a decision to proceed, and hence the next stage in concept development. This concept definition phase for a modification resembles a new system, except the scope of the system architecture and functional allocation is limited to certain subsystems or components. Proportionally greater effort is required to achieve compatibility with the unmodified subsystems or components, to ensure that the original functional and physical architecture is maintained. Therefore, these constraints require of high level of SE input to accommodate a variety of interfaces and interactions between the existing subsystems and the new subsystems. Furthermore, this process must accomplish this with a minimum of redesign, whist assuring that performance and reliability attributes have not been compromised. Similarly, the engineering development stage is limited to the new components that are to be introduced to the system under modification or upgrade. The integration of the modified system faces other challenges beyond those normally associated with a new system, which is related to two main factors as described by Kossiakoff and Sweet (2003). Firstly, the system being modified is more than likely been subject of numerous repairs as a result of a number of years of operation. During this time these repairs may have not been adequately documented or poorly configuration-managed. Furthermore, in the case of a fleet of systems some may have been subject to different repairs or no repairs at all, making the fleet increasingly different over a period. This system configuration uncertainty requires extensive audits or diagnostic testing case of software and adaption during the modification process.

Kossiakoff and Sweet (2003) state that the level and scope of subsystem test and evaluation required after a major upgrade or modification can range considerably from an evaluation limited to the new capabilities provided, to a full repeat of the original system evaluation and certification efforts. The level of test and evaluation effort is determined by the degree that the modifications affect the system capabilities that can be verified separately. Alternatively, when the modification alters the central or core functions of the system, it is necessary to perform an extensive re-evaluation of the total system. This may mean that an extensive re-certification program be required.

Lastly Kossiakoff and Sweet (2003) state that major system modifications always require correspondingly major changes in logistic support, particularly in those areas of spare parts inventory, publications updates and training. These latter stages of the design life-cycle require the same SE guidance as associated with the development of the original system. While the scope of the systems engineering effort is less, the criticality of design decisions and management of their impacts is no less important.

Faulconbridge and Ryan (2014) also discuss the impact of modifications to a system that has seen operational service or use – with this in this context being referred to as the *Utilisation* phase. The major activities during this phase include system operational use, system life-cycle support and modifications (or sometimes referred to as system *Upgrade* by Kossiakoff and Sweet (2003). Configuration management plays an important role during the utilisation phase to ensure that the configuration is managed, maintained and updated as required. Differences in physical configuration and the system documentation can make maintenance and operation potentially difficult and dangerous, particularly when a fleet of systems is involved. Faulconbridge and Ryan (2014) state that modifications may be required to rectify deficiencies with the performance of the system that were not identified during the acquisition phase. These deficiencies may be identified during the Operational Test & Evaluation (OT&E) phase or later operational use, where the system is placed in its operational environment and used by operational personnel. Other reasons for modifications may be a result of susceptibility to failure as part of the Failure Reporting Analysis and Corrective Action System (FRACAS), and that engineering changes are required via a modification to correct failures or system unreliability. Modifications may be undertaken to changing system level requirements caused by a range of factors including operational support (technology obsolescence) and sustainability issues, or environmental factors. The latter may be the phasing-in or enforcement of new environmental controls or regulations impacting the operations of these systems. As stated by Faulconbridge and Ryan (2014) there may be opportunities to increase efficiency of the system, reduce weight, or to reduce costs, through the replacement of system elements with improved designs. Given these reasons, depending on the modification size and scope, there is a potential to significantly impact system performance and functionality. As shown in Figure 8, significant modifications can be considered as a systems development activity, and that SE methodologies may be employed to achieve modifications.

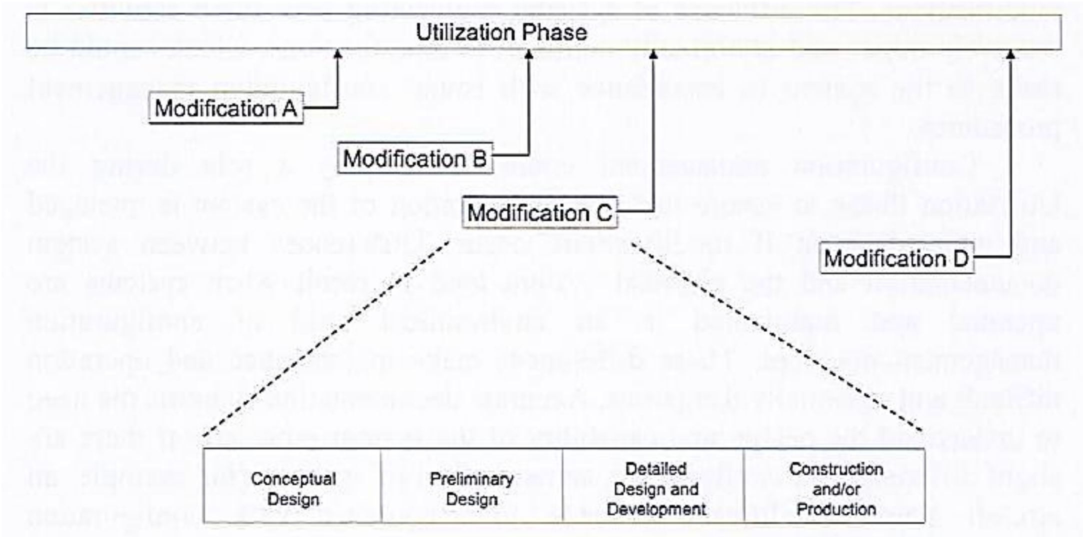


Figure 8. System modification impacts in the utilisation phase
Faulconbridge and Ryan (2014)

### 2.2.2 Cost of changes

Faulconbridge and Ryan (2014) state that the principal causes of cost and schedule overruns on large scale complex systems engineering development projects can be traced to various factors. These factors could include overambitious promotion of the modification, selection of low Technology Readiness Level (TRL) technology, lack of corporate strategic guidance, requirements instability or uncertainty, unrealistic project baselines, inexperienced project staff, and more generally inadequate SE. Faulconbridge and Ryan (2014) account for SE costs through implementation of systems processes and methodologies. These cost and schedule difficulties are often a result of inadequate requirements engineering practices, where poor requirements cannot be rectified by design. Faulconbridge and Ryan (2014) indicate that the SE has its greatest impact through structured application of these processes during the earliest phases of the project where changes can be affected easily and modification cost is the lowest. Consequently, SE provides the ideal opportunity to have the greatest impact on a project at time when these changes are easiest and inexpensive to make. There is therefore a strong incentive to manage and control these early phase conceptual design processes.

## 2.3 SUSTAINABILITY IN CONCEPTUAL DESIGN

### 2.3.1 Whole of Systems approach

Stasinopoulos et al. (2009) provide an account of whole of systems approach to sustainable engineering. This account is based on traditional system engineering approaches which are discussed within a sustainability context. Stasinopoulos et al. (2009) indicates that in the past, there has been a lack of appreciation of broader implications of engineering design decisions on larger natural systems. One example cited by Stasinopoulos et al. (2009) is the incorporation of lead additive to petrol with its hazardous impact to the health and wellbeing of community and the environment. Since the 1950s, most engineers and scientists were ignorant of the negative environmental impact of fossil fuel emissions, as it was assumed that the oceans and forests would act as a sink for these releases. Stasinopoulos et al. (2009) reference the Commoner's book, *The Closing Circle*, which shows that designers have in the past seldom sought to protect the environment. Commoner (1971) advocates new technologies that are designed with full knowledge of ecology and that these should integrate with naturally occurring systems. Furthermore Stasinopoulos et al. (2009) states that engineers should ensure that their solutions in the 21$^{st}$ century do not create new, unforeseen issues which may add to environmental degradation. Stasinopoulos et al. (2009) propose that an overarching area of emphasis involving sustainability is required when developing or modifying complex systems. Incorporation of sustainability considerations primarily affects two main areas involving emphasis on sustainable resource usage, and sustainable end of life options. Stasinopoulos et al. (2009) state that these two areas are largely absent from systems engineering theory. Therefore, there is a requirement to integrate sustainability into the SE process as a Whole of Systems approach. This Whole of System Design approach as described by Stasinopoulos et al. (2009) explicitly emphasises the steps required to develop sustainable system through the following:

- Sustainability solutions are considered to be key Technical Performance Measures (TPMs) along with the system function, performance and economic requirements.
- Technology research is emphasised as an initial step during the *Conceptual Design* of a system and continued through *Preliminary Design* and *Detail Design* phases.

### 2.3.2 Sustainability in aviation

As stated earlier, environmental issues are of increasing concern for the aviation industry. The main environmental impacts are associated with propulsion system combustion products, as stated by Torenbeek (2013), with about one-third of the radiative forcing arising from $CO_2$, for which fuel burn emissions is a metric. Other radiative forcing compounds are $NO_x$, $H_2O$, sulphate and soot. Torenbeek (2013) describes goals set by the Advisory Council for Aeronautics in Europe (ACARE) to improve the air transportation system in order to reduce its environmental impact. These goals are summarised as follows:

- Reduce fuel consumption and $CO_2$ emissions by 50%
- Reduce $NO_x$ emissions by 80%
- Reduce perceived external noise by 50%

These requirements were formulated for aviation as a whole and need to be decomposed into specific goals, for consideration in new aircraft design or modifications.

## 2.4 SYSTEM SYNTHESIS

### 2.4.1 Quality Function Deployment

Ullman (2010) states that understanding the design problem is fundamental for ensuring a quality product. Therefore, the early phase of the design process involving the determination of customer requirements is a key feature of the design process. Ullman (2010) states that many techniques can be applied to develop engineering specifications, with one most commonly applied being Quality Function Deployment (QFD). This QFD approach provides a method which organises the major components of information necessary to understand the design problem, and addresses:

- The customer needs;
- The development of the specification or goals of the product;
- Determination of how these specifications measure the customer needs;
- Determination and evaluation of how competitors meet the customer needs; and
- The development of numerical targets and measures.

Ullman (2010) outlines the QFD diagram, which is sometimes referred to as a House of Quality (HoQ). This QFD involves steps that builds a house-shaped diagram as illustrated by Figure 9. This HoQ application of QFD has been applied in the aerospace design domain, with this approach fully described in the General Aviation (GA) aircraft design text by Gudmundsson (2014). This QFD method comprises a number of steps which helps derive the information required in the product definition phase of the engineering design process. Each step populates a block within the QFD diagram, with the numbers in the diagram referring to a specific step. Ullman (2010), states that this method takes considerable time to complete, with experimental results showing that spending more time building a QFD diagram, providing a better understanding of the problem. This also results in better foundations for the latter concept generation steps.

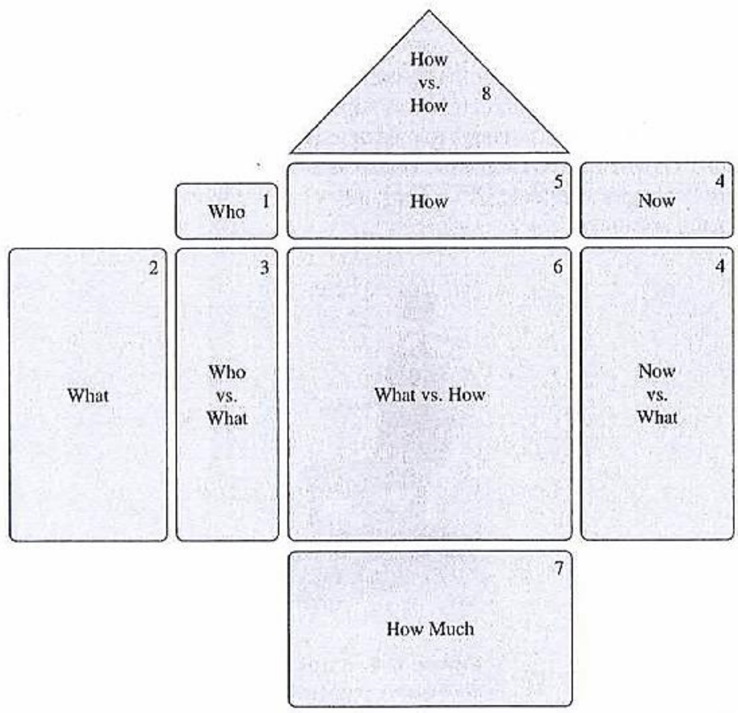


Figure 9. Quality Function Deployment diagram
Ullman (2010)

Raymer (2012) states that House of Quality (HOQ) techniques are often used to define requirements, assess relative importance, selection of design features, and are used in evaluation of the applicability of technologies in advance of making an aircraft layout. Raymer (2012) goes on to say that “in moderation, such tools have merit in the earliest phases of a design project”. However, Raymer (2012) observed that there is a tendency to devote excessive time and attention to such methods, to the detriment of actual aircraft design layout and layout-based trade studies.

### 2.4.2 Morphological techniques

One method used by Mavris & Kirby (1999) and D’Angelo et al. (2010) and Gavel (2007) has been application of General Morphological Analysis techniques to assess the interactive effects of technology on systems as discussed earlier. This is achieved using a morphological matrix approach which develops functional alternatives as defined in matrix columns. This morphological method was introduced by Fritz Zwicky (1898-1974). This is achieved using a morphological matrix as shown in Figure 10, which systematically searches the solution space by assessing all possible combinations within a matrix. This morphological matrix is developed by decomposing the functional elements of the system shown in Figure 10 as the vertical characteristics axis. The possible solutions for each function or sub-function are listed on the horizontal axis. Different solution concepts are therefore developed by combining each functional alternative to ensure that an exhaustive examination of the design space is achieved.

| | Alternatives / Characteristics | 1 | 2 | 3 | 4 |
|---|---|---|---|---|---|
| Config | Vehicle | Wing & Tail | Wing & Canard | Wing, Tail & Canard | Wing |
| | Fuselage | Cylindrical | Oval | None | |
| | Pilot Visibility | Synthetic Vision | Conventional | | |
| Mission | Range (nmi) | 3000 | 3500 | 4000 | |
| | Passengers | 100 | 150 | 200 | |
| | Mach Number | 0.8 | 0.83 | 0.85 | 0.9 |
| Prop | Type | Turbofan | AST Engine | IHPTET | |
| | Combustor | Conventional | RQL | LPP | |
| Control | Static Stability | Stable | Unstable | Relaxed | |
| | Gust control | Conventional | Unloaded | | |
| Aero | Low Speed | Conventional Flaps | Conventional Flaps & Slots | C C | |
| | High Speed | Conventional | LFC | NLFC | HLFC |
| Struct | Wing | Aluminum | Titanium | Composite | |
| | Fuselage | Aluminum | Titanium | Composite | |

Figure 10. Concept morphological matrix
Mavris & Kirby (1999)

A deficiency of the morphological analysis technique is that the design space can identify numerous potential solution candidates that arise from relatively small morphological matrices. That is, if there are $m_1$ possible solutions to function $f_1$, and $m_2$ solutions for $f_2$ and so on, then there are a total of $N = m_1 \cdot m_2 \ldots m_n$ possible solution candidate concepts.

A variation on this method applied to the fuel systems configuration design on fighter aircraft is provided by Gavel et al. (2006), Gavel (2007), Gavel et al. (2008), Ölvander et al. (2009) and Svahn (2006) where matrix-based methods are employed to quantify the morphological matrix. This provides a solution which is characterised with a set of parameters such as system weight, performance and cost. The selection of the individual concept solutions is modelled with decision variables and the optimisation problem is formulated with a mathematical framework as described below. The quantification of the morphological matrix provides access to every potential solution which is described as either a physical or statistical model, or a combination of both. Therefore, using this approach the TPMs are quantified. Further details of this approach are provided later in this study where quantisation of the morphological matrix is used to rate design modification alternatives and then down-select to a smaller manageable solution space.

## 2.5 EVALUATION OF CONCEPTS

### 2.5.1 Pugh matrices

Pugh’s method or decision matrices are a relatively simple and proven approach that can be used for comparing alternative concepts. Pugh’s decision matrices as described by Burge (2009) involves a step by step approach which is analogous to a QFD diagram. Each step fundamentally scores each concept relative to the others in its ability to meet the criteria.

### 2.5.2 Design Structure Matrices

Eppinger & Browning (2012) provides an extensive account of Design Structure Matrix (DSM) methods applied to a range of architectures including product, organisational, process and multi-domain applications. The DSM is a network modelling tool which is particularly useful in characterising the interactions of system elements in such a way that the system architecture is highlighted for further analysis. The DSM is suited to applications involving the development of complex, engineered systems and has seen extensive application to engineering management disciplines. This DSM can also be applied to engineering changes or modifications to a systems architecture and is a useful method highlighting subsystems impacts or interactions. These interactions can be extended into the risk analysis domain to provide a visual indication of modification risks.

However, the main advantage of DSM in the system modification/upgrade context is its value to change propagation within a complex system such as an aircraft or a helicopter. Example 3.6 within Eppinger and Browning (2012) provides an account of how DSM can be applied to aircraft design where upgrades, retrofits or modifications are incorporated to meet a specific need. During this process, a design change to one part of the aircraft will in most cases impact other systems or subsystems. The prediction of such design change relationships provides a significant challenge in the management of retrofits or modification of complex systems where numerous change propagation paths may result. This application of DSM to systems modifications and resultant change propagation is dealt with in Section 2.6.

### 2.5.3 Value-based metrics

Value-based approaches are an extension of those techniques and tools outlined above into the cost, engineering management and systems domain, where Value Driven Design (VDD) is a process activity which takes place iteratively, across all levels of the organisation, as stated by Eres et al. (2014). This VDD methodology provides early multidimensional value information, in order to:

- Enable the selection of early concepts and designs representing the highest value contribution;
- Enable system optimisation at the highest integration level (in terms of the value proposition);
- Promote the development of high quality and high value driven requirements.

A number of research papers have dealt with this domain, focusing on larger aircraft concepts, different technologies, and incorporating the approaches as discussed in the previous section. These papers can be loosely collected into those that deal with value-based life-cycle evaluation methodologies which have been developed by Georgia Institute of Technology; Value-Based Multi-Disciplinary Optimisation (MDO) techniques developed by Massachusetts Institute of Technology (MIT); with VDD undertaken at the Value Driven Design Institute Illinois.

The major contributor to this area is research undertaken by MIT, where a different approach was undertaken which relied more on financial modelling techniques extended to the operational domain; market uncertainty, business risk, development and manufacturing costs and aircraft demand. Papers by March et al. (2009), Markish & Willcox (2002, 2003), Peoples & Willcox (2004), Willcox (2005, 2002), Willcox and Wakayama (2003) optimise these parameters using stochastic and dynamic programming approaches to investigate performance, cost and revenue for single and family of aircraft cases studies. Research by Collopy (2009) and Collopy & Hollingsworth (2011) provides a useful insight into VDD involving value modelling theory, aerospace value models, and guidelines for constructing these models.

#### *2.5.3.1 Cost bottom-up and top-down methods*

Curran et al. (2004, 2005) provides a very good account of cost estimating techniques including analogous, bottom up, neural networks, fuzzy logic and parametric costing methods. Parametric cost estimates utilise Cost Estimate Relationships (CERs) and associated mathematical algorithms (or logic) to derive cost estimates. This approach is commonly used within the aerospace industry which typically involves linear regression analysis for CER development. This is further applied in Roskam (2002), as discussed below. These CERs are derived using a methodology as shown by Figure 11.

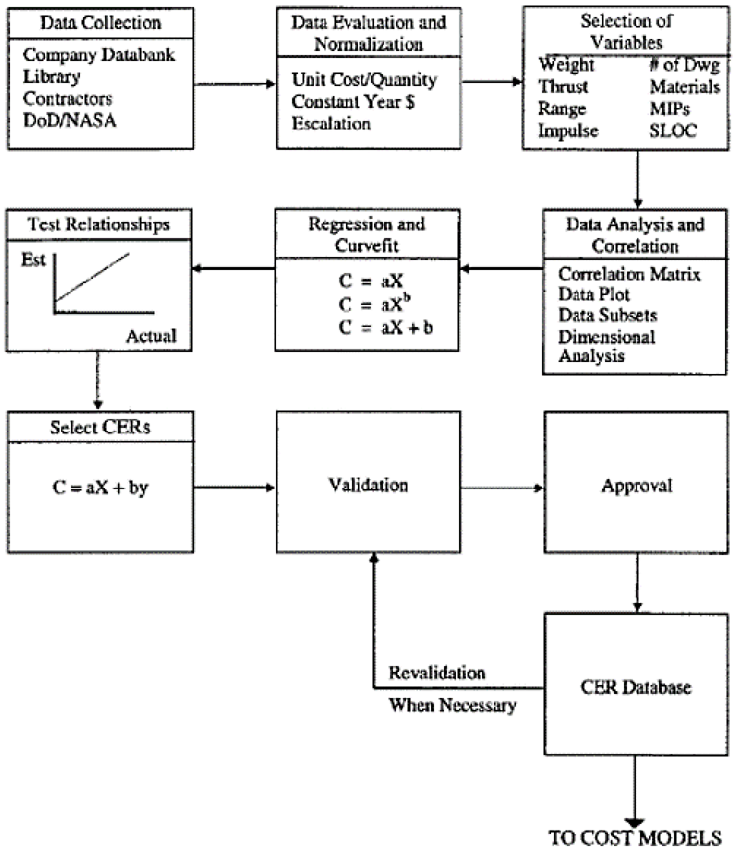


Figure 11. Methodology for developing parametric cost models
Curran et al. (2004)

### *2.5.3.2 Causal cost modelling approaches*

Castagne et al. (2008) states that "ideally, any facilitating costing methodology should be able to operate and interface at various levels and during all stages of the life-cycle". This has been a fundamental consideration in the development of an approach referred to as *Genetic Causal Costing*. This is conceptualised in Figure 12 where the causal definition of cost to design dependencies is seen in the context of product families. The model adopts the scientific principle of categorisation whilst also incorporating the requirement of utilising causal relations. Although this cost modelling approach has been successfully applied to airframe manufacturing, it is noted that it could be also applied throughout the aircraft development life-cycle including aircraft operations.

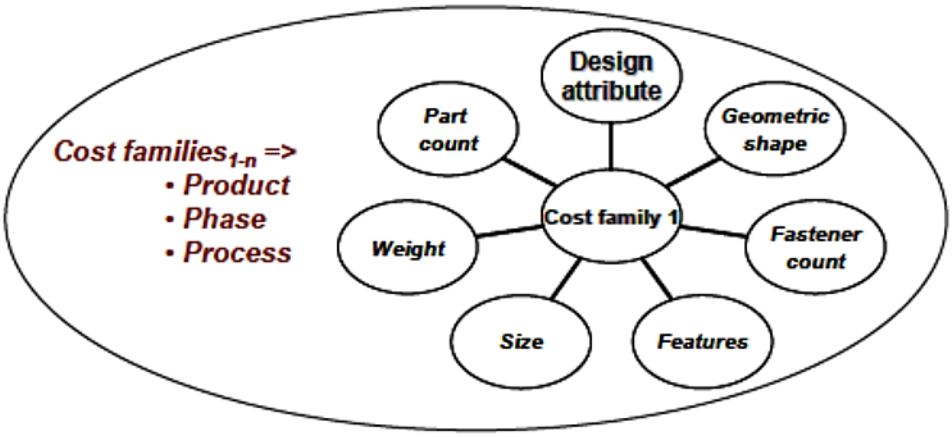


Figure 12. Conceptual illustration of causal cost modelling approach
Curran et al. (2004)

#### 2.5.3.3 Systems life cycle costing

Fabrycki and Blanchard (1991) outlines a Cost Breakdown Structure (CBS) also known as a cost tree framework which can be used to support lifecycle cost analysis. This CBS is provided as a means to facilitate the initial breakdown of costs (top-down) and the subsequent estimation of costs on a functional basis (bottom-up). The CBS includes all costs and is intended to aid in the overall visibility of costs. Fabrycki and Blanchard (1991) state that the CBS is tailored to specific requirements, with the cost categories varying in terms of the depth of coverage and the system being evaluated. In these case studies, the system evaluated comprises the alternate fuel or propulsion system modification of a small aircraft. Therefore, it is a requirement that the CBS shall cover of the modification development lifecycle from research and development through to disposal. An example of a general top-level CBS is presented in Figure 13.

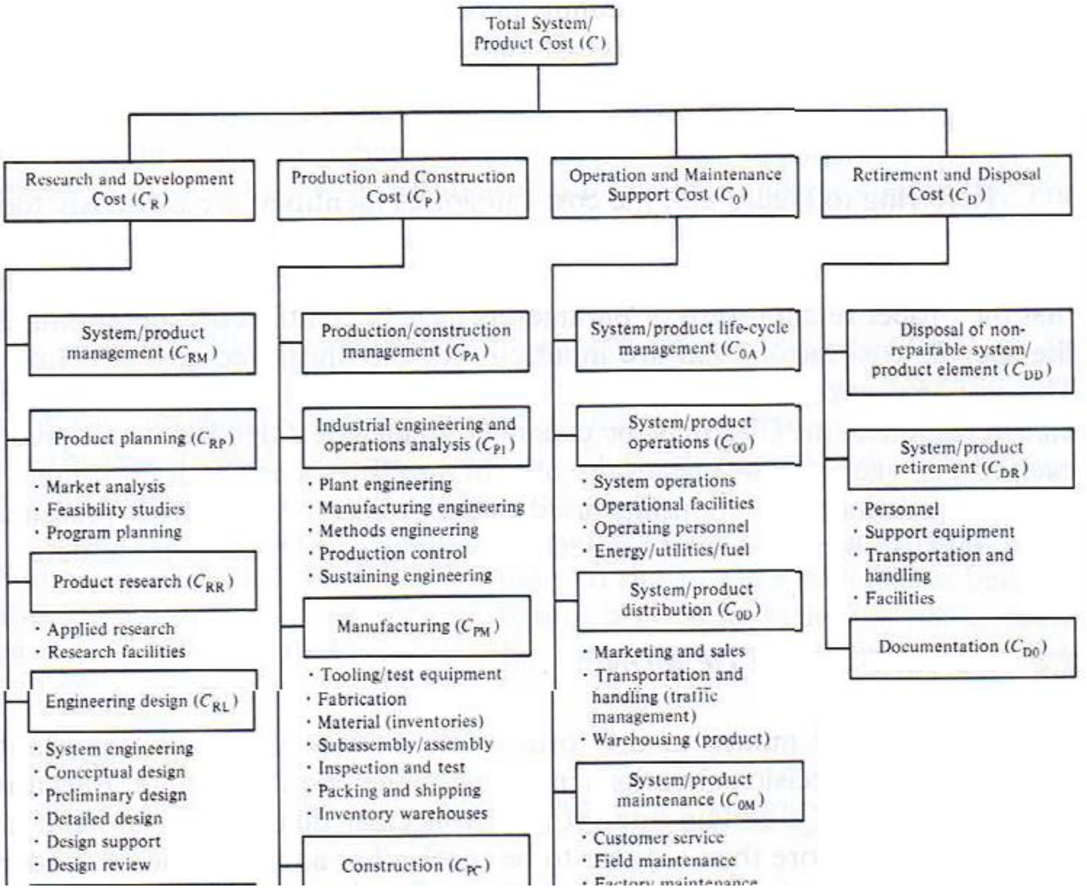


Figure 13. Excerpt - General cost breakdown structure – intentionally cropped
Fabrycki and Blanchard (1991)

### 2.5.3.4 *Aircraft cost estimation methods*

Roskam (2002) provides a general methodology for aircraft cost estimation using linear regression analysis of existing designs. This approach is based on a thorough parametric analysis of general categories of aircraft, such as twin-engine commuter category aircraft, the application of regression analysis to estimate various costs associated with Research, Development, Test and Evaluation (RDTE), manufacturing and operating costs (direct and indirect). It also provides an account of aircraft design optimisation and design to cost and associated constraints. However, this reference does not provide methodology that could be applied to evaluate new technologies apart from general notes and guidance in relation to configuration selection, drag prediction, loads prediction, laminar flow and range prediction.

Nevertheless, this reference is a value comparative data resource from which to derive baseline data to validate the methods and data outputs from this proposed research.

Gudmundsson, (2014) provides another parametric costing methodology based on the Development and Procurement Costs Aircraft (DAPCA-IV) model to estimate development costs associated with General Aviation (GA) aircraft and Business Aircraft. This costing methodology establishes special cost estimating relationships which are a set of parametric equations that predict aircraft acquisition costs using only basic information like empty weight and maximum airspeed. For this reason, the DAPCA-IV model can only be used to estimate cost for RDTE and workforce estimation. It should be noted that the DAPCA-IV model was based on cost structures associated with military aircraft. Therefore, the modifications to this method presented in this reference are based on the “Eastlake” model by Eastlake & Blackwell (2000) which accounts for GA and business aircraft as described above. Like the models presented in Roskam (2002) the basis of these cost models are parametric and statistically based, and therefore do not take account new technologies integration. However, like Roskam (2002) the cost models, particularly those for business aircraft provide validation data for the methods proposed in this research.

### 2.5.4 Change propagation impacts

A comprehensive review of engineering changes within complex products and systems has been previously summarised by Jarrett et al. (2011). The focus of this section describes research relating to engineering changes and propagation of changes in aircraft and helicopter-based changes. As described earlier, modifications are made throughout the lifecycle of an aircraft or helicopter to enhance performance, provide new design features or functionality, or reduce Life Cycle Costs. To this extent, there is a need to understand the causes and sources of change, as well as efficient ways of managing change to ensure high quality and cost-effective design processes. Clarkson et al. (2001) discusses the redesign of systems and the impact of these redesign activities on various subsystems by presenting a change behaviour analysis based on a case study at Westland helicopters. Furthermore, this study discusses the development of a related model to predict the risk of change propagation. The papers by Eckert et al. (2006, 2009) characterises product change based on similar studies in the aerospace field. Furthermore, this study introduced a tool to assist designers in understanding the potential effects of engineering changes. Two approaches are presented; (1)

Probabilistic prediction of change impacts, and (2) Visualisation of change propagation through product interfaces.

### 2.5.5 Multiple-Domain Matrix extensions

The paper by Koh et al. (2012) presents a modelling method supporting change propagation prediction and management within complex engineering design and development projects. Like the work undertaken by Clarkson et al. (2001) and Eckert et al. (2001, 2004 & 2006) this paper builds on the QFD techniques and the Change Prediction Method (CPM) to model the effects of potential change propagation brought about by design change options. A framework is proposed by Koh et al. (2012) extends these methods into different domains given by (1) change options, (2) product requirements and (3) product components. Hence a Multiple-Domain Matrix (MDM) is proposed by Koh et al. (2012) that better illustrates how dependences are modelled between the different domains. It is noted that this MDM approach is essentially a combination of DSM and Design Mapping Matrices (DMM).

This thesis will explore this DSM framework further and extend the approach into Engineering and Certification design mapping domains.

## 2.6 SYSTEMS ANALYSIS AND OUTPUTS

### 2.6.1 Systems analysis process

As a system design modification develops there are numerous trade-offs involving the evaluation of different technologies, alternative system schemes, manufacturing processes and logistic support strategies. In general, the approach followed in undertaking systems analysis is illustrated by Figure 14. Blanchard and Fabrycki (1998) state that the trade-off studies lead into synthesis which develops a feasible system concept based on combination and structuring of components. Systems analysis can be undertaken early to develop concepts, then later to define detail at lower levels.

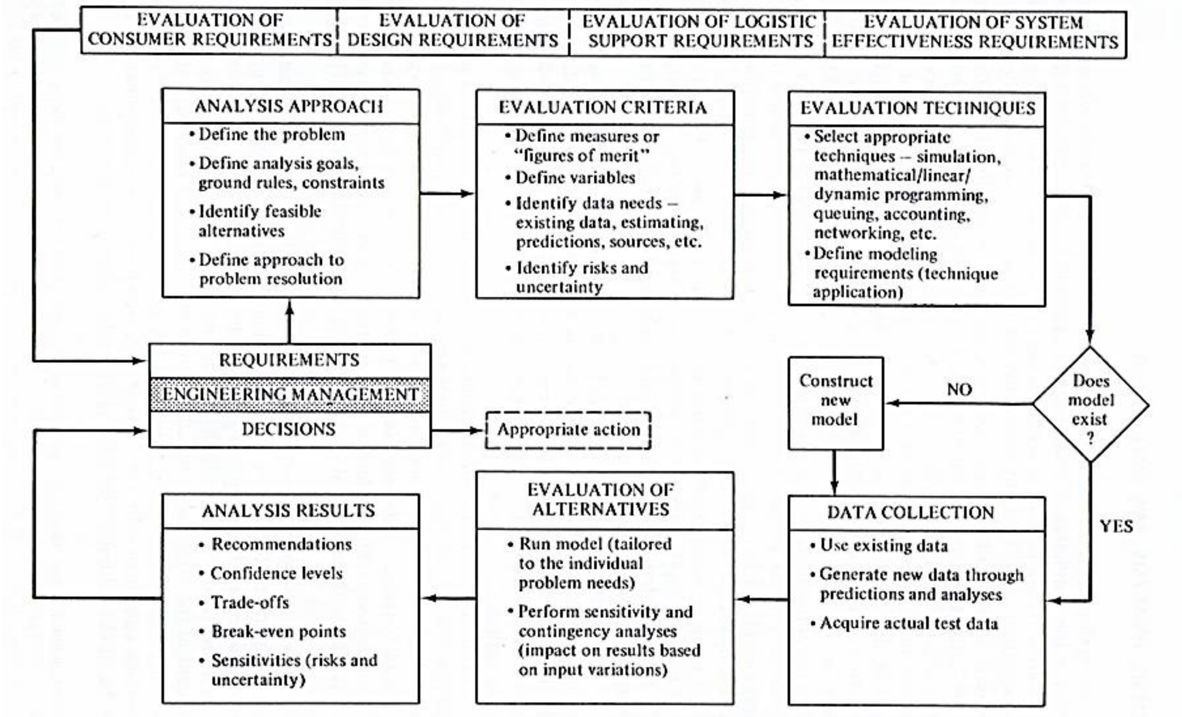


Figure 14. Generic systems analysis process
Blanchard and Fabrycki (1998)

### 2.6.2 Outputs

The output of this systems analysis activity are artefacts to support decisions as shown in Figure 14. This process uses simulation, analysis models and tools to evaluate results against evaluation criteria, such as TPMs, figures of merit or metrics in order to rank the attributes of alternative solutions. The data and information supporting this process is then used to provide artefact outputs which is used in subsequent design phases.

# Chapter 3. Design Modification Space

*In today's world of computer programs, sophisticated analysis, and computer-aided design, the need still exists for quick, cursory methods of estimating weight especially for early conceptual studies. One might say that there is still a need to take a quick look at the forest before examining a few trees.*

*D.P. Marsh (1982)*

## 3.1 DESIGN SPACE CONTEXT

### 3.1.1 Background

The design methodologies employed in general aviation aircraft modification projects are sometimes based on ad hoc aviation industry practice constrained by regulatory requirements and airworthiness design standards. The focus of these modification projects is to achieve compliance with airworthiness design standards whilst also meeting the modification specification. Innovative design methods are not generally employed insofar that the main modification requirements to be satisfied are to achieve certification whilst satisfying minimum weight, minimum drag with the appropriate structural integrity. Other metrics such as those associated the modification project life-cycle may consider Research Development Test and Evaluation (RDTE) activities only, with other considerations associated aircraft operational costs, maintenance support, and infrastructure development included as lower priority. These latter items are not usually considered within the modification specification as they are either not a mandatory regulatory requirement, not directly impacted by the proposed modification, or they are left to the client to address separately. This therefore results in a disjointed development effort that may overlook important impacts of the modification on later life-cycle stages.

Although aviation regulatory requirements provide design standards which have been proven to provide safe and airworthy designs and/or modifications, these design standards sometimes may constrain innovative solutions. In some cases, the standards do not consider all life-cycle considerations, \novel design solutions, or new technologies. For example, the standards consider minimum performance requirements for safe flight, but the cost impact, payload capability, sustainability impacts or infrastructure requirements are not generally the prime focus. However, it is noted that contemporary amendments of FAR Part 23 Amendment 23-64 (2016), now accounts for new technological developments. In order to continue this approach, an evaluation methodology could be employed that incorporates these regulatory requirements, but is inclusive of other life-cycle considerations associated with innovative design features or modifications shall provide sustainable and better value aviation solutions.

### 3.1.2 Design space research boundary

This conceptual study of aircraft modifications (design changes) is life-cycle based. It therefore includes both the aircraft and the supporting infrastructure segments. Detailed costs associated with engineering design or certification components of the modification development lifecycle will not be developed as part of this study, as these can be dealt with in further follow-on research which can fully develop the methodologies to model these costs. Costs associated with manufacture or installation of the modification will not be dealt with as these costs do not significantly differentiate competing options. For example, two similar options comprising down-selected configurations will not be significantly differentiated by the installation costs. By definition these installation costs are those costs associated with the installation of the modification components including also the fabrication of tools and jigs. Rather the modification options may be differentiated by major component costs which are driven by the different technologies and configurations.

## 3.2 CASE STUDY AIRCRAFT DESCRIPTIONS ROLES AND MISSIONS

Two case study aircraft are considered in this thesis in order to triangulate the design methodology. The aircraft are specifically selected for their differences in type, role, and mission profile. The first aircraft type is a twin-engine commuter category aircraft, and the second is a single-engine four-seat small aircraft used for skydiving, both of which are described in the following sections.

### 3.2.1 Cessna Model 421 commuter aircraft

The Cessna Model 421 is an all-metal low-wing aircraft with a retractable landing gear powered by two Continental reciprocating internal combustion engines (Taylor et al., 1983-84). The cabin has seating for six on the basic Cessna 421 version, or up to ten passengers on later versions. A three-view drawing of the Cessna Model 421 aircraft is shown by Figure 15.

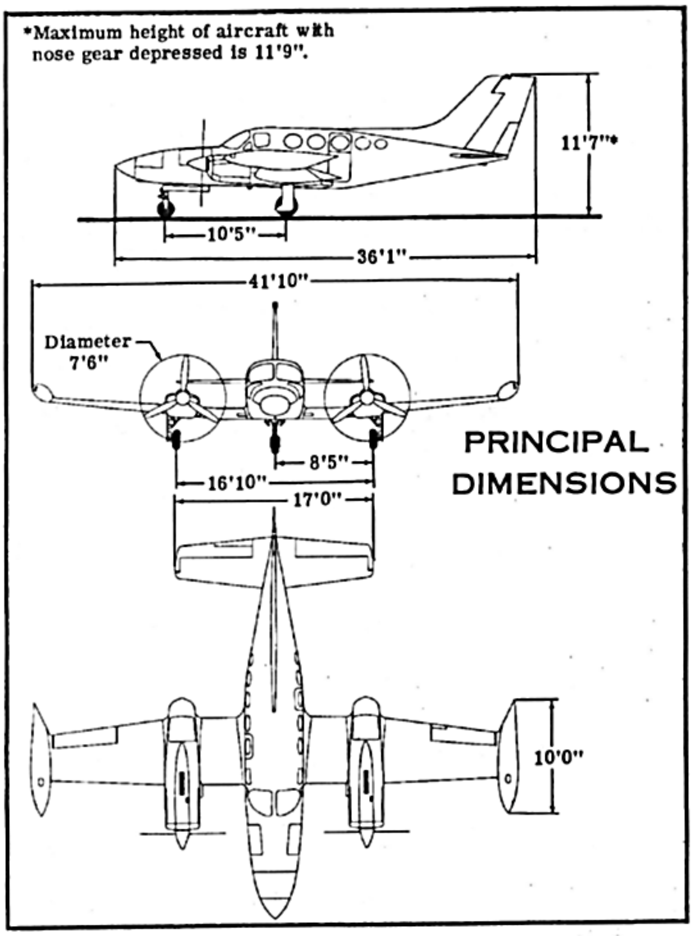


Figure 15. Cessna Model 421B aircraft
Cessna Aircraft Company (1974)

### *3.2.1.1 Commuter aircraft role*

Cessna Model 421B aircraft are typically utilised in a commuter role, and are used for short-range charter or short-range transportation by regional airlines (Torenbeek, 1982). The need for commuter aircraft emerged in the 1970s, when the airline industry adopted a "hub-and-spoke" air transportation strategy (Torenbeek, 1982).

### *3.2.1.2 Cessna Model 421B TCDS Excerpt*

The FAA Type Certificate Data Sheet (TCDS) A7CE (2007) provides the following information in relation to the Cessna Model 421, Golden Eagle aircraft, as shown in Table 2.

Table 2. TCDS excerpt - Cessna Model 421B Golden Eagle

| Parameter | Description |
|---|---|
| Engines | Two Continental GTSIO-520-H reduction gear ratio 0.667:1.<br>These engines described are turbocharged, fuel injected, six cylinders horizontally opposed, air cooled, and incorporate an overhead valve design. |
| Fuel | Grade 100 or 100LL aviation gasoline |
| Engine limits | For all operations, 2275 propeller RPM rated at 375 HP (280 kW) |
| Propeller | Two McCauley fully-featherable 3-bladed propeller installations |
| Airspeed limits | Manoeuvring: 152 knots<br>Max structural cruising: 200 knots<br>Never exceed speed: 238 knots |
| Maximum weight | Landing 7200 lbs (3265 kg), takeoff 7250 lbs (3288 kg) |
| Fuel capacity | 175 US gal (662 l) total<br>2 wingtip tanks 51 US gal (193 l) each, 50 US gal (189 l) usable<br>2 wing tanks 36.5 US gal (138 l) each, 35 US gal (132 l) usable |

Excerpt - FAA Type Certificate Data Sheet (TCDS) A7CE (2007)

#### *3.2.1.3 Mission profile*

The mission profile of a commuter aircraft is important as it is this characteristic that provides the basis for range, payload and speed performance. A typical mission profile is illustrated below based on the Cessna owner's manual (Cessna Aircraft Company, 1974). Aircraft altitude is shown as the vertical axis and the distance flown is shown on the horizontal axis. Note that the altitude scale is exaggerated to show details of the mission profile.

The mission profile consists of two segments: the nominal mission segment and the reserve segment as shown by Figure 16. Each of these is divided into several sub-segments as discussed below.

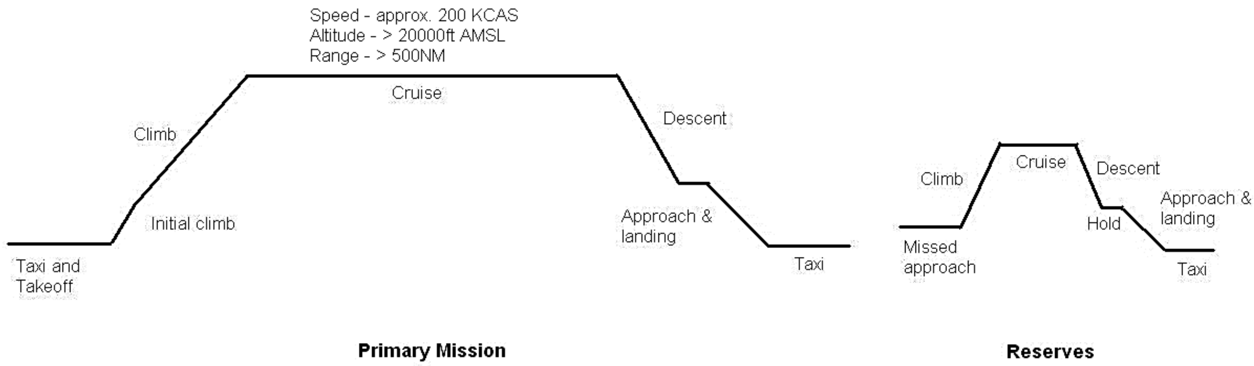


Figure 16. Typical commuter aircraft mission profile

- Taxi and takeoff - Taxi and takeoff is typically a short sub-segment, and is dependent on airport traffic and layout.
- Initial climb - The initial climb is typically constrained by other regulations such as an airspeed limit schedule below 10,000 ft (3000 m).
- Climb – This is typically an enroute climb segment where time and fuel burned during climb may include several climb sub-segments flown at different speeds.
- Cruise – This is the cruise sub-segment. For longer range flights, the initial and final cruise altitudes may vary since the airplane weight changes substantially. However, for shorter commuter category missions the cruise altitude may remain fixed for operational reasons.
- Descent approach, and landing - Like the climb sub-segment, the descent is performed according to a specified airspeed schedule.
- Reserves - Reserve fuel is carried to allow for contingencies, including a requirement for diversion to an alternate airport when the planned destination is unavailable. National Airworthiness Authority (NAA) regulations specify a minimum amount of reserve fuel. However, many commuter airlines have additional operational requirements that result in reserves usually being higher than the NAA minimums.

#### *3.2.1.4 Passenger/baggage weights*

In small aircraft, space for baggage is usually very limited. Normally baggage allowances restrictions apply for each passenger unless otherwise specified (Altitude

Aviation, 2015). However, in smaller commuter aircraft (less than 12 seats), the passenger and baggage weight and volume limits are much more critical and normally individual passenger weights are required. This ensures that the aircraft is loaded within weight and balance limitations and also provides information for fuel planning purposes.

Typically, these smaller commuter aircraft are generally designed to have a trade-off between payload able to be carried and fuel load. Often an aircraft will not be able to carry its full passenger capacity due to fuel loading and weight limitations (Altitude Aviation, 2015). This is a significant operational difference when compared to traditional larger airliner baggage limitations.

#### *3.2.1.5 Fuel stops*

Sometimes the aircraft may require enroute fuel stops, adding about 30 minutes to the trip. If the aircraft is heavily loaded, more fuel stops may be required. Conversely, a lighter payload may not require as many, on no fuel stops at all.

### 3.2.2 Cessna 182 Skylane four-seat light aircraft

The Cessna 182 Skylane aircraft is a 4-seat high wing aircraft with fixed landing gear powered by a single 230 HP (172 kW) Continental O-470-U flat six engine (Taylor et al., 1983-84). A total of 19,364 Cessna 182 Skylanes were built by 1 April 1982. A three-view drawing of the Cessna Model 182 aircraft is shown by Figure 17.

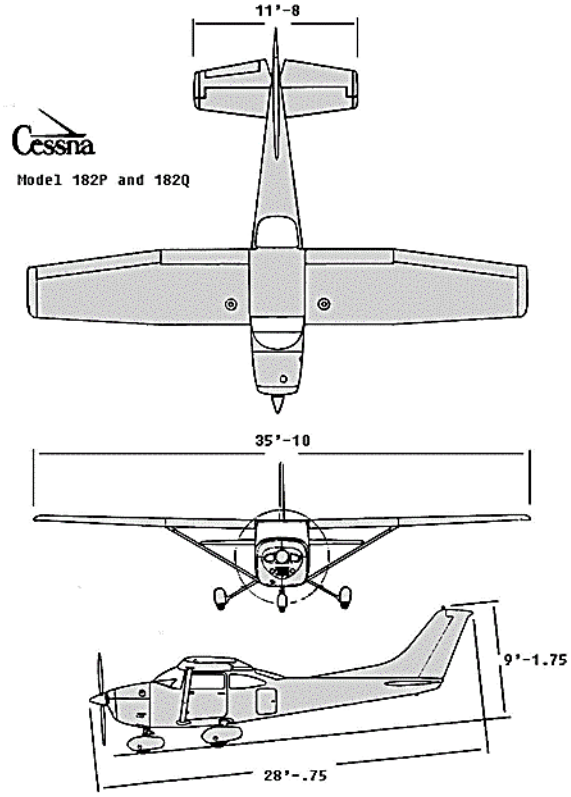


Figure 17. Cessna Model 182 Skylane
Cessna Aircraft Company (1976)

#### *3.2.2.1 Skydiving aircraft role*

The Cessna 182 Skylane is a general-purpose light aircraft used in a variety of roles including private transportation for business, flight training and also air charter as described earlier. In this case this aircraft is adapted for skydiving by removing the passenger seats to allow access around and out of the aircraft for these types of operations. As described in correspondence with Glesk (2018, pers. comm., 13 June) skydiving is typically conducted by clubs as a commercial operation and can involve operation of multiple aircraft used to ferry skydivers to the required jump altitude adopting a mission profile as described below.

#### *3.2.2.2 Cessna Model 182P TCDS Excerpt*

The FAA Type Certificate Data Sheet (TCDS) 3A13 (2006) provides the following information in relation to the Cessna Model 182P, Skylane aircraft, as shown in Table 3.

Table 3. TCDS excerpt - Cessna Model 182P

| Parameter | Description |
|---|---|
| Engine | Continental O-470-S<br><br>The Continental O-470-S engine is a six-cylinder, horizontally opposed, air-cooled aircraft engine developed especially for use in light aircraft by Continental Motors. |
| Fuel | Grade 100 or 100LL aviation gasoline |
| Engine limits | For all operations 2400 RPM, 230 HP (172 kW) |
| Propeller | Two-blade McCauley constant speed propeller installation |
| Airspeed limits | Manoeuvring 111 knots<br>Max structural cruising 143 knots<br>Never exceed speed 179 knots |
| Maximum weight | 2950 lbs (1338 kg) takeoff/flight<br>2950 lbs (1338 kg) landing |
| Fuel capacity | 92 US gal (348 l) - 88 US gal (333 l) usable<br>Two 46 US gal (174 l) integral tanks in wings |

Excerpt - FAA Type Certificate Data Sheet (TCDS) 3A13 (2006)

#### *3.2.2.3 Mission profile*

The mission profile of skydiving aircraft in commercial operations comprises carriage of jumpers (skydivers), to an altitude of 4300 m (14000 ft). A typical skydiving mission profile is shown in Figure 18. This implies a mission time (or endurance) of between 30 and 40 minutes, which when given the weight of the payload and altitude required yields an energy requirement to be satisfied by the propulsion system (Glesk, 2018, pers. comm., 8 October).

As with the commuter aircraft mission, the skydiving mission can be broken into various sub-segments as summarised below:

- Taxi and takeoff - Taxi and takeoff is typically a short sub-segment, and is dependent on airport traffic and layout.
- Climb – Climb is conducted at the speed for best climb rate to minimise the time taken to get to “jump” altitude and also to reduce costs.
- Cruise – Cruise at altitude is typically very short in duration, and comprises positioning the aircraft and stabilisation of airspeed required for the jump phase.

- Descent approach, and landing – The descent phase is typically undertaken at the greatest rate of descent within engine cooling and RPM limitations, and airframe limitations.
- Reserves - Reserve fuel is carried to allow for deviations from the skydiving mission. However, in the case of skydiving operations minimum fuel is maintained to achieve the required duty cycle of 1 to 2 loads per hour (Glesk, 2018, pers. comm., 8 October).

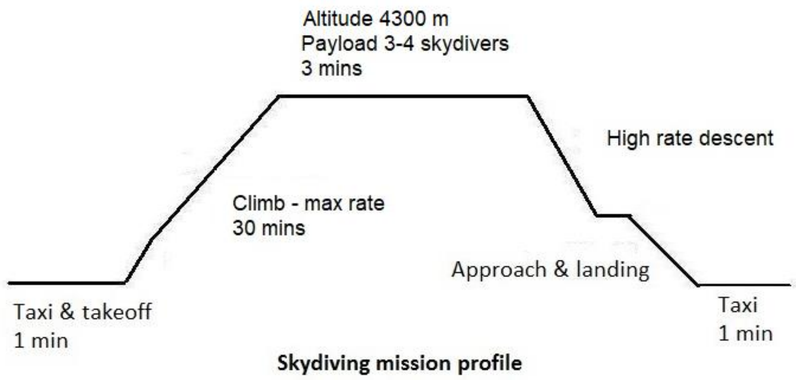


Figure 18. Typical skydiving mission profile

#### *3.2.2.4 Skydiver weights*

In light aircraft, actual passenger weights are used for flight planning purposes. Skydiving operations obviously do not involve the carriage of baggage and any additional weight is minimised to ensure maximum climb performance of the aircraft. Therefore, maximum skydiver weight is determined by the maximum certificated takeoff weight of the aircraft less the pilot weight and minimum fuel allowances. This may provide for a payload of 3 to 4 skydivers depending on individual weights as described by Glesk (2018, pers. comm., 8 October)

## 3.3 EXISTING GROUND FUELLING INFRASTRUCTURE

### 3.3.1 AVGAS fuel infrastructure

Piston engine aircraft use AVGAS fuels which are provided through a significant logistics supply infrastructure. This infrastructure exists to provide re-fuelling services to these aircraft and other similarly powered GA aircraft on a Nation-wide basis (BITRE, 2017). All major General Aviation airports provide aviation gasoline (AVGAS) services supplied by major fuel companies. These services are provided by mobile fuel tankers or by self-serve fuel bowsers. The smaller regional centres usually provide self-serve bowsers, whilst some of the larger regional centres providing fuel tankers in addition to bowsers (Airservices Australia, 2017).

### 3.3.2 Fuel availability

As a result of the harmful AVGAS emissions, the US Environmental Protection Agency EPA (2008, 2010) started a consultative process to phase out AVGAS in the US. Although this process was started in 2008, opponents have successfully argued that there are no viable "drop-in" alternative fuels that achieve the same performance as TEL-based aviation gasolines at that time. This phasing out process has stalled as a result of strong lobbying by aviation groups. Nevertheless, it is likely that these eventual plans to phase out AVGAS will limit its availability at some time in the near future.

## 3.4 CASE STUDY AIRCRAFT MODIFICATIONS

### 3.4.1 Commuter aircraft natural gas fuel modification

This case study proposes a modification that integrates a natural gas fuel system with the existing Cessna 421B fuel system. The major part of this modification is the installation of additional or modified fuel tanks, specifically replacement of the current wing tip fuel tanks with natural gas fuel tanks and/or the addition of a fuselage belly tank underneath the fuselage. Figure 19 shows the Cessna Model 421B aircraft with the proposed natural gas fuel tank modification options.

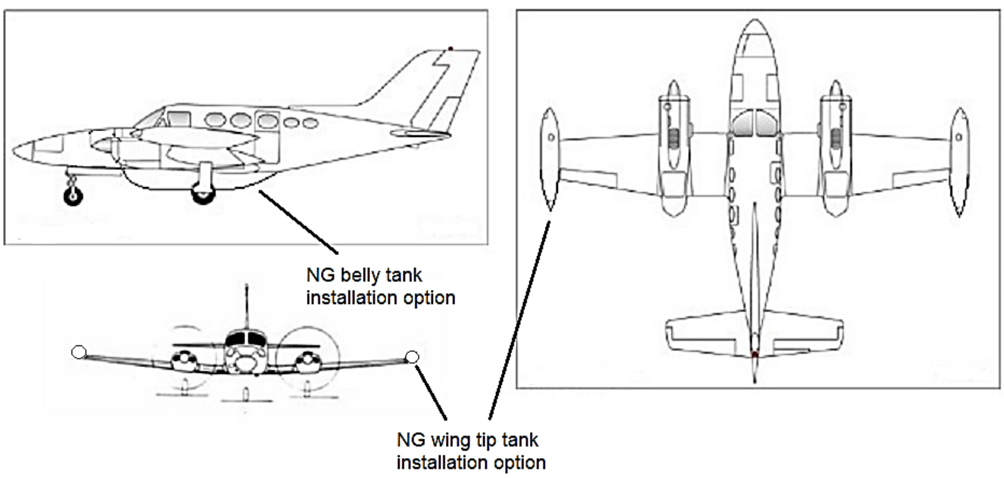


Figure 19. Cessna Model 421B aircraft with proposed natural gas fuel tank options

Although other fuel tank configurations were possible, the case study was restricted, via requirements, to those installations that did not impact the payload carrying capability of the aircraft, or had the least structural impact. It should be noted that various fuelling combination arrangements are available under this arrangement. For example, CNG or LNG-only options are possible, but also available in combination with bi-fuel options comprising CNG/AVGAS or LNG/AVGAS, each with various tank location options as described above.

#### *3.4.1.1 Compressed Natural Gas fuels storage*

To be practical as a transportation fuel, natural gas must be compressed or liquefied to decrease its storage volume. There are three common storage pressures for CNG fuel tanks being: 2400 psi (16.5 MPa), 3000 psi (20.7 MPa), and 3600 psi (24.8 MPa) as described by Sinor (1991). CNG tanks are cylindrical and have much thicker walls than gasoline tanks which lowers the overall amount of fuel that can be stored within a given volume.

#### *3.4.1.2 Liquefied Natural Gas fuels storage*

The advantage of LNG in terms of energy storage density is readily evident from Table 4 and is the reason heavy vehicles prefer using LNG to CNG fuels. It should be noted that LNG fuel systems also are lighter per unit volume of fuel storage compared with CNG fuel systems. As stated by Sinor (1991), LNG is a very clean fuel since no water vapor or sulphur compounds can survive the liquefaction process. Higher hydrocarbons such as ethane and propane can be present, though this is usually undesirable because of a phenomenon called "weathering." LNG is stored at very low temperatures (-120 to -162°C). While the insulation of LNG storage tanks is very good, LNG still experiences a net gain of heat because of temperature differences between the fuel and the ambient surroundings. As heat is gained by the LNG, vapor is generated which must eventually be released from the storage tank to avoid over-pressurization as LNG tanks are not designed to maintain high pressures (if they were, they would lose the advantages in tank weight which come from storing natural gas as a liquid rather than a gas).

Table 4. Selected physical storage and energy density properties of CNG and LNG

| | CNG | LNG |
|---|---|---|
| Physical state | Gas | Liquid |
| Temperature in vehicle tank | Ambient | -162°C |
| Temperature in vehicle tank | Ambient | -259°F |
| Typical pressure in tank | 2,500-3,600 psig | 10-50 psig |
| Typical pressure in tank | 17.3-24.9 MPa | 170-446 kPa |
| Typical density in tank | 1.1-1.6 lb/gal | 3.5 lb/gal |
| Typical specific gravity | 0.13-0.19 | 0.42 |
| Typical energy density (LHV) | 23,000-34,000 Btu/gal | 75,000 Btu/gal |
| Typical energy density (LHV) | 6,500-9,500 MJ/L | 21,000 MJ/L |
| Heat of vaporization | | 220 Btu/lb |

LHV = lower heating value

Sinor, (1991)

### *3.4.1.3 Ground transport applications of CNG and LNG*

Ground transport applications of CNG and LNG to vehicles such as cars, trucks, and buses are well established. Accordingly, the research undertaken in this is area is extensive with a number of papers having investigated the performance of converted reciprocating spark ignition engines, which are similar to the aviation equivalent. Papers by Aslam et al. (2005, 2006) and Jahirul et al. (2010) have researched and tested performance of reciprocating spark ignition automobile engines which have been retrofitted to operate on natural gas and gasoline as a bi-fuel option. These papers discuss comparisons of engine-fuel performance metrics in detail, characterising brake horsepower (BHP) output, Specific Fuel Consumption (SFC), and Exhaust Gas Temperature (EGT) as a function of engine throttle setting for each fuel (gasoline and natural gas).

However, Aslam et al. (2005, 2006) and Jahirul et al., (2010) have not investigated the effects of density/pressure altitude, and other operational factors (temperature, icing etc.) which are significant factors affecting aircraft engine performance. Furthermore, these papers did not consider changes to the engine such as increasing compression ratio by increasing turbocharger boost to compensate for power reduction resulting from the use of natural gas fuel. Nevertheless, it appears that there were no obvious or significant barriers that would prevent adaption of these ground transport CNG/LNG technologies to aviation applications.

#### *3.4.1.4 Aviation applications of CNG and LNG*

**Small aircraft applications** - The use of natural gas in various forms by aviation has been considered by the Beech Aircraft Company, which successfully modified and flew a piston-engine light aircraft in the early 1980's. The article by Flight International (1981) outlined the results of the modifications and results of flight tests conducted on a Beechcraft Sundowner piston-engine aircraft modified for operation on LNG. Flight International (1981) stated that the tests showed positive results indicating that LNG was "cleaner burning" than conventional AVGAS fuel, exhibited lower fuel consumption and demonstrated lower operating costs. However, the article did not characterise payload range performance, or the life-cycle costs associated with the modification.

More recently research and development efforts in this area have seen the conversion of a single engine piston experimental aircraft for operation on CNG fuel. This aircraft conversion is reported by Hirschman (2013) and Wynbrandt (2013), where an Aviat Husky aircraft was converted to operate on a bi-fuel combination comprising conventional AVGAS and CNG. Hirschman (2013) and Wynbrandt (2013) state the changes to the engine consisted of fitting higher compression ratio pistons (increased compression ratio from 8.50:1 to 10:1) and an engine control system which compensated for density altitude, engine timing and "other factors" relating to bi-fuel operation on AVGAS and CNG fuels. Wynbrandt (2013) states that CNG advantages over AVGAS, is cost (CNG is approximately 80% of AVGAS), higher octane (138 vs 100), contains no lead, reduces smog by 90 percent and carbon dioxide by 30 percent. The CNG installation weight was reported to be about 135 lbs (61 kg) and a current generation CNG tank installation may weigh 30 lbs (14 kg) less. Performance of this installation in relation to payload/range, cruise speed, and takeoff metrics was not reported.

**Jet and turboprop aircraft applications** - A number of papers have investigated the use of natural gas fuels as alternatives to conventional gas turbine Jet fuels such as that described by Dorrington (2013) and Withers et al. (2014). Withers et al. (2014) provided an account of the benefits of LNG as a jet fuel noting that it is less costly by 70-80% on an energy basis resulting in a reduction in aircraft operating costs. This paper investigated LNG as a secondary fuel in a military turboprop aircraft and provided an estimate retrofit costs to use LNG in a bi-fuel configuration. It is stated by Withers et al. (2014), that aircraft operators could save up to 14% on fuel expenses (with retrofit expenses included).

Burston et al. (2013) described the conceptual design of a liquid methane (LNG) powered passenger aircraft (Airbus A320 – A350 size aircraft). The focus of this paper was the conceptual design and layout aspects of this medium haul transport aircraft, as well as the technical considerations for converting an **existing** airliner to bio-methane fuels. Burston et al. (2013) approached this conceptual design problem by examining various aspects of system architecture, range-payload comparisons, developing configuration options and methods of evaluation. Burston et al. (2013) concluded that the weight penalties associated with the use of such fuel would be modest, and thus the LNG range and payload capability could be matched to conventional powered aircraft. Kiros & Bil (2014) extended the work by Burston et al. (2013) by evaluating the life-cycle cost elements of LNG fuels on transport category aircraft. This paper carried out an economic analysis of LNG fuels based on a modified Airbus A320 aircraft, as a well as a related environmental study which examined the associated emissions. Kiros & Bil (2014) concluded that a dual fuel (Jet A – LNG) aircraft minimises the required modifications and results in savings of Direct Operating Costs (DOC) with a break-even point in the first year of operation. Furthermore, as a fuel source LNG induces a 20% reduction in $CO_2$ emissions compared to current Jet A fuel.

### 3.4.2 Four-seat skydiving aircraft electric propulsion modification

This case study proposes a retrofit modification by replacing the existing Cessna 182 aircraft engine and fuel system with an electric propulsion system. This retrofit modification is similar to other electric propulsion system modifications carried out on a Cessna 172K light aircraft as described by Fehrenbacher et al. (2011). To that extent the modification will replace the existing Internal Combustion (IC) engine with an electric propulsion option which is sized to fit forward of the engine firewall as shown in Figure 20. This electric propulsion system generally comprises an electric motor, electric controller, wiring, battery systems, and associated flight displays for energy status and condition monitoring. Externally this modified Cessna aircraft will resemble the original aircraft with minor differences potentially involving a redesigned engine cowl to cater for different cooling requirements, and to allow access for exchange batteries if required.

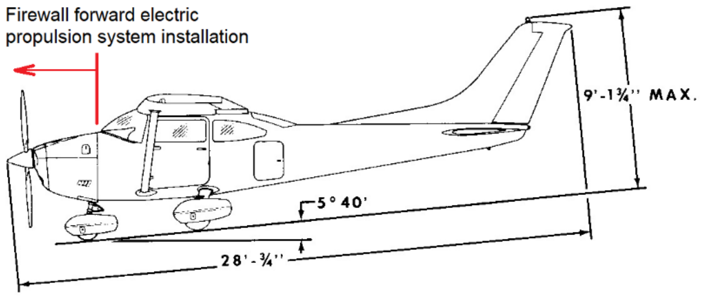


Figure 20. Cessna Model 182 aircraft with proposed electric propulsion system modification

### 3.4.2.1 *Electric propulsion technology*

The use of electrical power as a means of propulsion for aircraft is not a new concept. The integration of electric propulsion systems has received recent attention for thin-haul commuter and on-demand transportation as reported by numerous researchers including Moore & Fredericks (2014), Patterson et al. (2012) and Stoll & Mikic (2016). Patterson et al. (2012) states that these electric aircraft concepts are receiving increased attention for their potential in eliminating emissions. In addition, these electric propulsion system concepts have the potential to substantially reduce noise with significant increases in reliability. Electric motors require smaller volumes and weigh less than the equivalent internal combustion engines. This provides the ability to use redundant motors with minimal penalties. Furthermore, electric motors can provide an emergency power surge capability for 30 seconds, although subject to motor heat saturation limitations. This emergency power surge capability can further improve aircraft safety.

There are differences in the benefits attainable by retrofitting existing aircraft with fully-electric propulsion systems and designing a completely new aircraft for electric propulsion. Aircraft that incorporate electric propulsion systems have different characteristics, such as no power decrease with altitude, and no fuel burn related weight loss. These characteristics violate conventional design assumptions for reciprocating and turbine engines. Additionally, there are no engine shock cooling limitations that are applicable to reciprocating engines operating under high power settings and then transitioning to low power and low airspeed settings for descent. As stated by Glassock et al. (2017) this shock cooling limitation is applicable to air-cooled internal combustion engines where damage can occur to cylinder heads during high duty cycle flight profiles used in skydiving and gliding towing operations.

Electric propulsion systems integrated on aircraft are not without disadvantages. There are numerous airworthiness certification issues involved in the practical adoption of fully electric aircraft into the air transportation system including the development of industry consensus standards specifically for electric aircraft. These are currently being developed at this time and are detailed in the Final Rule provided in FAA (2016). However, challenges lie ahead in that industry needs to collaborate and agree on rapidly evolving electric and hybrid-electric technology and configuration architectures.

In addition to certification issues, there are also technology related challenges associated with fully electric aircraft. The issue of battery storage capacity represents the major obstacle in the widespread adoption of fully electric aircraft. As described by Patterson et al. (2012) the specific energy density of battery technologies is currently lower than that of conventional fossil fuels. Furthermore, Patterson et al. (2012) states that while the weight of electric motors is considerably less than that of comparable IC engines and the electric motor efficiency is considerably higher, the low specific energy density of the batteries currently leads to a much higher aircraft weight for the same amount of practical energy storage.

## 3.5 REGULATORY CONSTRAINTS

### 3.5.1 Aircraft

Airworthiness design standards specify regulatory requirements which must be achieved to provide an airworthy and safe type design, with advisory circulars and other guidance documentation providing information in relation to the acceptable means to demonstrate compliance. The process followed to achieve Type Certificate (TC) or Supplemental Type Certificate (STC) is specified by the NAA type certification manual, which describes the documentation, planning and review steps. An example of this type certification manual is provided by CASA (2017). This manual provides the minimum requirements for Type Certification, and more often than not it is the “Applicant” that needs to develop a development methodology appropriate to the project (TC or STC). Generally, the Applicant processes used are based on standard engineering processes as specified in the Approved Design Organisation (ADO) manual (for a minor change) and implemented the Certification Program Plan (CPP), sometimes referred to as a Certification Plan (CP), if a major change to Type design. It is this latter CPP document that is the key plan that defines the airworthiness requirements impacted by the design change and how compliance with these requirements will be demonstrated.

Torenbeek (1982) indicates that civil and design airworthiness requirements have a significant influence on the design of aircraft structures, systems, equipment installations, performance and flying qualities. This requires that the correct airworthiness design standard be selected applicable to the aircraft type, and operational category.

As stated earlier, the policies and constraints applicable to a complex modification project which might involve alternate fuels or propulsion systems fitted to a small aircraft may be defined by airworthiness standards such as Code of Federal Regulations (CFR) 14 Federal Aviation Regulation (FAR) Part 23 (1965) and FAR Part 33 (1964) as described below.

These airworthiness standards have equivalents in other nations which closely align to the FAR airworthiness standards, so that compliance with local Regulations can also be shown to comply with US Code of Federal Regulations.

One such standard for small aircraft operations of maximum takeoff weight ≤ 12500 lbs (5700 kg) is FAR Part 23. FAR Part 23 describes Airworthiness Standards for Normal, Utility, Acrobatic and Commuter Category Airplanes and is structured into the following Subparts:

- A – General (and definitions)
- B – Flight
- C – Structures

- D – Design and construction
- E – Powerplant (installation)
- F – Equipment (installation)
- G – Operating limitations and information

Each Subpart described above, contains numerous related sub-paragraph requirements that make up the aircraft airworthiness design standard. This airworthiness standard therefore comprises a major set of top-level requirements which are underpinned by law. Non-compliance with this standard will result in non-viable solution.

### 3.5.2 Engines

In a similar structure as described above, FAR Part 33 is an example of one airworthiness standard for aircraft engines. The FAR Part 33 standard prescribes the airworthiness standards for the issue of type certificates and changes to those certificates, for aircraft engines. FAR Part 33 is applicable to both reciprocating engines and turbine aircraft engines and is also structured into Subparts as follows:

- A - General
- B - Design and Construction; General
- C - Design and Construction; Reciprocating Aircraft Engines
- D - Block Tests; Reciprocating Aircraft Engines
- E - Design and Construction; Turbine Aircraft Engines
- F - Block Tests; Turbine Aircraft Engines
- G - Special Requirements: Turbine Aircraft Engines

Given that both case study aircraft are powered by reciprocating engines, then the Subparts A through D are applicable. Like FAR Part 23 described earlier, FAR Part 33 contains numerous related sub-paragraph requirements that make up the engine airworthiness design standard.

### 3.5.3 Fuels

The certification of aviation fuels is a complex area, noting that the specification for AVGAS has evolved over many years to provide a safe and reliable aviation fuel.

The specification for AVGAS is ASTM Standard, D910-11 (2011). This specification ensures that AVGAS provided worldwide is a “good” fuel for all stakeholders including the producers, engine manufacturers, airframe manufacturers, component manufacturers, and the users of the fuel. Therefore, a change in fuel specification will impact aircraft performance, fuel consumption, operating instructions, maintenance requirements and instrument markings when compared to the original certification basis of the aircraft and engine. The impact of certification of an alternate aviation fuel is dealt with in the natural gas case study shown in Appendix 1, noting that this activity in itself could be the basis for detailed study.

## 3.6 DEVELOPMENT AND OPERATIONAL ENVIRONMENT

### 3.6.1 Design environment

The current end users associated with aircraft modification development projects fall within the following:

- Design organisation – The organisation which undertakes the modification design activity including development of methodologies to evaluate the design options and solutions (CASA, 2014). This organisation may be the Applicant as described earlier.
- Modification installation organisation – The organisation undertaking the physical installation of the modification.
- Supplier organisation – The organisation providing components and subsystems.
- Client – This may be the operator or owner of the aircraft.
- NAA - The Regulator which is responsible for provision of certification review, advice and approval.

### 3.6.2 Support environment

The support environment associated with aircraft modification projects fall into the following categories:

- Design Organisation - The organisation providing ongoing engineering support to the modification (CASA, 2014).
- Approved Maintenance Organisation – The organisation undertaking

routine aircraft maintenance including systems and subsystems impacted by the modification. This organisation may also support in-service changes or updates to the modification.

- NAA - provision of Airworthiness Directives (AD) and Airworthiness Bulletins (AWBs) as required.

### 3.6.3 Operational environment

#### *3.6.3.1 Commuter aircraft charter*

Typical charter operations conducted by a commuter category aircraft are based on the business of renting an entire aircraft as opposed to purchasing individual aircraft seats (BITRE, 2017). These charter operations involve operations which are flown to the passengers' itinerary, in day or night, and in Visual Flight Rules (VFR) or Instrument Flight Rules (IFR) conditions.

A typical charter operation involves passengers arriving at the airport 30 minutes prior to scheduled departure, especially if the charter itinerary is time critical. The aircraft is usually fuelled prior to the passenger's arrival. Passenger and baggage weights are processed and are loaded according to the aircraft weight and balance system. This step takes less than 10 minutes as aircraft weight limitations sometimes impose a single baggage item for each passenger (Altitude Aviation, 2015). Given aircraft weight restrictions, it is sometimes necessary that refuelling is required at some intermediate airport enroute to the final destination. This refuelling stop may about 30 minutes depending on operational factors. Typically, intermediate stops would be made into regional airports with AVGAS self-service bowsers or fuel tankers. Note that other factors may also require enroute refuelling such as stronger than planned headwinds or other operational constraints. In this case flight planning would consider availability of fuel at these intermediate stops.

Typically charter operations have the advantage that the itinerary can be developed in accordance with passenger needs as outlined by the National Air Transportation Association (2012). In addition, the itinerary can take flights to airports which are not normally serviced by Regular Public Transport (RPT). It also follows that these charter aircraft can be operated from airports with shorter unsealed airstrips, which provides significant flexibility over larger RPT aircraft, where sealed runways are generally required.

#### *3.6.3.2 Skydiving aircraft operational scenario*

The Cessna 182 has been used by the skydiving community since the early days of skydiving. This aircraft can carry a pilot, three (3) to four (4) skydivers to an altitude of 14,000 feet, which usually takes about half an hour with a full payload to climb to the jump altitude (Glesk, 2018, pers. comm., 8 October).

Correspondence with Glesk (2018, pers. comm., 8 October), highlighted that a typical skydiving mission would comprise a payload of skydivers and pilot plus minimum fuel required with fixed reserves. These flights would attempt to achieve a duty cycle of one (1) to two (2) loads per hour, depending on a range of operational factors and skydiver demand. Refuelling would occur between these flights depending on the skydiver loadings.

# Chapter 4. Formulation

*Without doubt, weight and weight distribution, or balance, are of more importance in airplane design than in any other branch of engineering.*

*T.P. Wright (1999)*

## 4.1 OVERVIEW

### 4.1.1 Conceptual design methodologies

Design methodologies as defined by Pahl et al. (2007) “is a course of action for the design of technical systems that derives its knowledge from design science, cognitive psychology and practical experiences gained from different domains”. These design methodologies make it easier to reapply and establish solutions from earlier projects, and to use technical databases or common structures to apply to design modification projects. Indeed, the establishment of common structures or design catalogues is a pre-requisite for computer applications and support of the design process using simplified mathematical relationships representing performance attributes of the system. As stated by Pahl et al. (2007) systematic design methodologies make the task easier to divide between the designer and the computer, thus providing efficiencies to the modification project.

Conceptual design as described by Blanchard and Fabrycki (1998) is a process that evolves from a need, to the definition of the requirements in functional terms through establishment of the design metrics, and preparation of a system development specification. This introduction of a design methodology within conceptual design therefore provides a framework that defines a course of action within the early design lifecycle phases of a project. The accomplishment of a feasibility analysis or a trade study is a major step within conceptual design that involves three main steps. These steps being:

1. identification of possible design approaches,
2. evaluation of these approaches based on performance, effectiveness, maintenance, logistic support, and cost economics, and
3. a recommendation of the preferred course of action.

In addition, considerations are given to the application of different technologies as part of the design approach. The system requirements analysis steps within this process involve definition of operational requirements, support concepts, the provision of TPMs, functional analysis allocation, synthesis and evaluation. Blanchard and Fabrycki (1998) describe TPMs as the metrics or quantitative factors associated with the system under development.

The most important engineering design document produced during the conceptual design phase is the system specification as described by Blanchard and Fabrycki (1998). This document defines the system functional baseline, including results from the needs analysis, trade-off analysis, operational requirements and maintenance concept, top-level functional analysis, and identifies the TPMs and Design Dependant Parameters (DDPs). This specification may lead into one of more subordinate specifications covering subsystems, support equipment, materials, processes software and other components of the system.

Ullman (2010) describes this conceptual design phase as being primarily concerned with the generation and evaluation of concepts. Generation of concepts is described by Ullman (2010) where customer requirements are utilised to develop a functional model of the system. This functional modelling approach is essential for developing and generating concepts that will eventually lead to a system that is fit for purpose. The evaluation of concepts is a step that compares the concepts generated by the requirements, which is then used to make decisions about selection of the best alternatives. The latter steps of this phase, as shown by Figure 21, involve documenting of the candidate solution, refinement of the project plan, and formal approval of the concepts.

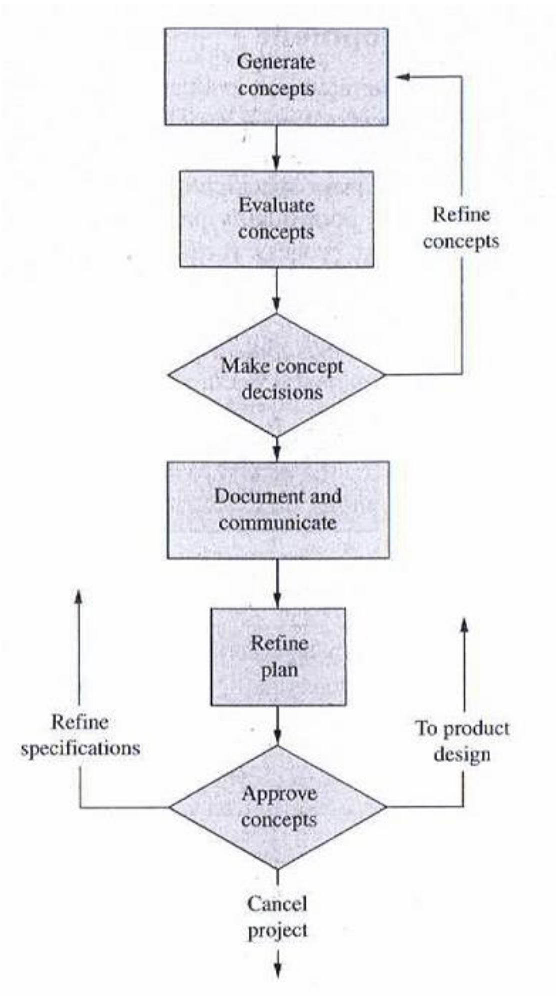


Figure 21. Conceptual design steps of mechanical systems design
Ullman (2010)

### 4.1.2 Conceptual design methodology requirements

Systematic design provides a way to rationalise the design and its associated through life support processes. Structuring the problem and task makes it easier to recognise established solutions from previous projects as stated by Pahl et al. (2007). This stepwise development of established solutions makes it possible to generate, select and evaluate them at an early stage of the design activity and with a reduced level of effort. Furthermore, these systemic processes also make it easier to divide the task between designers and computers, as described earlier. Pahl et al. (2007) states that in order that a design methodology meet these needs it must possess various attributes. These attributes which form the basis of requirements for this conceptual design methodology are quoted as follows:

1. “Allow a problem-directed approach, in that it must be applicable to every type of design activity, no matter the specialist field it involves.

2. Foster inventiveness and understanding in searching for an optimum solution.
3. Be compatible with the concepts, methods and findings of other disciplines.
4. Not rely on finding solutions by chance.
5. Facilitate the application of known solutions to related tasks.
6. Be compatible with electronic data processing.
7. Be easily taught and learned.
8. Reflect the findings of cognitive psychology and modern management science, that is reduce the workload, reduce design time, prevent human error, and help maintain an active interest.
9. Ease the planning and management of teamwork in an integrated and inter-disciplinary product development process.
10. Provide guidance for leaders of product development teams".

### 4.1.3 Formulation approach

The formulation of the conceptual design methodology in this Chapter involves breaking down the process elements from the highest levels, developing new approaches, and adapting existing tools and techniques to provide a multi-step universal framework to apply to aircraft modification programs. This approach embraces systems engineering, product development, and traditional aircraft design methods within a broader framework which formulates the problem in terms of *synthesis*, *evaluation* and *analysis*. It then decomposes these three elements into a unique matrix-based framework by adapting existing tools and techniques or developing new approaches to cater for design modification space. It has adopted a matrix-based method, as it provides a structured and rigorous framework from which to (1) manage requirements, (2) generate and evaluate concepts, (3) validate concept selection decisions, (4) evaluate design change/modification options, (5) evaluate change propagation impacts, (6) manage engineering and certification resources and risks, and (7) analyse performance. Furthermore, it is recognised that the engineering and certification related activities can be managed more effectively and efficiently if structured in a matrix-based framework. Indeed, the aviation industry presents certification information in a tabular format which is sometimes referred to as a compliance summary matrix document. This methodology therefore extends this approach and incorporates and refines the format to encompass the impact of the change propagation resulting from the modification, in addition to providing a structure to manage related resources and costs. Although not presented in this thesis, the design outputs of the conceptual design methodology are structured in such a way that they provide information and data inputs to the necessary design documentation. This approach provides a standardised systems engineering, airworthiness regulation and project management documentation suite. This design information is used throughout the various phases of the design lifecycle, to support further analysis and development effort in refining the modification design.

This chapter therefore details the research theory, techniques, tools and approaches used in formulating this conceptual design methodology.

## 4.2 CONCEPTUAL DESIGN

### 4.2.1 Systems Engineering aspects

Blanchard and Fabrycki (1998) state that conceptual design evolves from an identified need to the definition of the system requirements in functional terms, through establishment of the design metrics, conduct of a feasibility analysis, and lastly the development of a system specification. The accomplishment of a feasibility analysis is a major step within conceptual design that involves three main steps being:

1. identification of possible design approaches,
2. evaluation of these approaches based on performance, effectiveness, maintenance, logistic support, and cost economics, and
3. a recommendation of the preferred course of action.

Also considered are applications of different technologies in combination with the design approach. The system requirements analysis steps within this process involve definition of operational requirements, support concept requirements, the provision of TPMs or metrics, functional analysis allocation and synthesis and evaluation.

Blanchard and Fabrycki (1998) state that the functional analysis allocation process translates system requirements into detailed design criteria or metrics. This process involves abstraction of the needs and then breaking this down to identify requirements for hardware, software, tools, processes, people, facilities, data and the associated combinations. This functional analysis is achieved through use of functional flow block diagrams, which breakdown system high level functions to second level functions and third level functions.

Once the top-level description of the system is defined in functional terms, the next step involves functional allocation. This functional allocation as described by Blanchard and Fabrycki (1998), groups similar functions into logical sub-divisions or groups through identifying major subsystems and lower level elements of the overall system. This structure serves as a framework for preliminary design and evolves from the development of TPMs and Design Dependent Parameters (DDPs) which can be allocated to the appropriate system element.

As the system design develops there are numerous trade-offs involving the evaluation of different technologies, alternative system architectures, manufacturing processes and support strategies. In general, the approach followed in undertaking a trade-off study, or evaluation leads into synthesis which refers to the combination and arrangement of components in such a way as to represent a feasible system solution. The synthesis activity involves the formation of a solution which could be representative of the configuration that the system will eventually take. Synthesis can be undertaken early to develop concepts, then later to define design detail at lower levels.

#### *4.2.1.1 Systems engineering processes*

Faulconbridge and Ryan (2014) suggests that systems engineering processes are built around an iterative application of, synthesis, evaluation and analysis. This iterative approach is fundamental, where initially this occurs at the systems level; then applied to the subsystems level; and then at the various lower levels to components, and so on to the level of detail required in the development process. The analysis, synthesis and evaluation processes are undertaken during conceptual design and involve customer needs in defining requirements. This process is shown conceptually by Figure 22, as an iterative analysis-synthesis-evaluation loop.

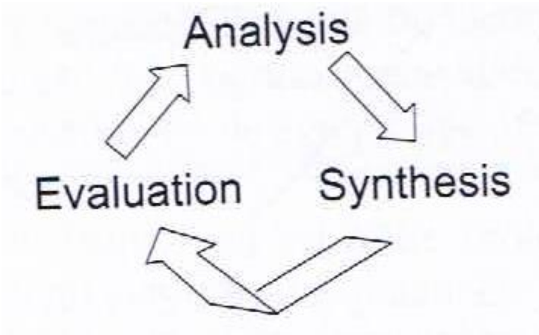


Figure 22. Analyses-synthesis-evaluation iterative process
Faulconbridge and Ryan (2014)

The Analysis process as described by Faulconbridge and Ryan (2014) starts with establishing the project needs. As part of conceptual design, the analysis process investigates these needs and identifies those essential requirements of the system. Requirements analysis activities continue throughout the development life-cycle to develop lower level requirements associated with the functional and physical attributes of the system design. The allocation of these requirements forms a description of the system elements and architecture, and therefore assists in the next process of synthesis.

Synthesis is the integrated process of creativity and technology adoption combined to create a design that meets the system requirements. Faulconbridge and Ryan (2014) state that synthesis is a more appropriate description of this process as it hints to an evolutionary nature of design. In the early phases of conceptual design, synthesis is limited to defining the logical design or architecting the system, and then considering the viable technical solutions using the results of the requirements analysis activity. Later in this process, the selected design architecture is further synthesised until the complete design is finalised to the appropriate level.

Evaluation is the process of investigating and comparing trade-offs based on design requirements, and making the necessary decisions to enable selection of solutions. The process of evaluation continues throughout all stages of the system development life-cycle, determining whether the system satisfies the needs and requirements, and if so, to what level. Faulconbridge and Ryan (2014) state that a trade-off analysis is one of the methods available to undertake this evaluation, with several steps involving:

- Definition of requirements
- Identification of alternative solutions

- Nomination of selection criteria such as metrics
- Determination of criteria weighting
- Definition of scoring functions
- Evaluation of alternatives, and
- Sensitivity studies

The outcome of evaluation is the confirmation of the best candidate solution. Shortfalls that might be identified may result in further analysis and synthesis, with this applied iteratively throughout the life-cycle.

#### *4.2.1.2 The cost of changes*

Cost is an important attribute of systems design and Faulconbridge and Ryan (2014) state that the principal causes of cost and schedule overruns on large scale complex systems engineering development projects can be traced to combinations of numerous factors. These factors could include overambitious support, selection of immature technology, lack of corporate strategic guidance, requirements instability or uncertainty, unrealistic project baselines, inexperienced project staff, and more generally, inadequately applied systems engineering processes. These scheduling problems are often a result of poor requirements engineering practices, where poor requirements cannot be rectified by design. Faulconbridge and Ryan (2014) indicate that the SE has its greatest impact through rigorous application of processes and methodologies during the earliest phases of the project where the ease of change and cost of modification is the lowest. Consequently, SE provides the ideal opportunity to have the greatest impact on a project at a time when these changes are easiest and inexpensive to make. Figure 23 shows the effect where during conceptual and preliminary design, the costs associated with making changes are very low, with these design changes being much easier to incorporate and implement.

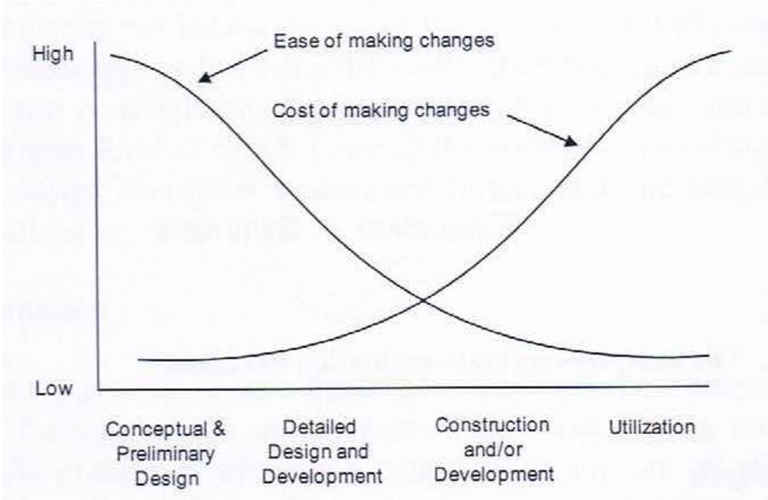


Figure 23. Ease of cost of making design changes throughout the system life-cycle
Faulconbridge and Ryan (2014)

### 4.2.2 Product design

Pahl et al. (2007) describes the product conceptual design phase which is similar to the systems engineering life-cycle process. Numerous texts, such as those by Blanchard and Fabrycki (1998), Kossiakoff and Sweet (2003), and Faulconbridge and Ryan (2014) also describe this conceptual design phase. This product conceptual design phase is concerned with determining the most viable or preferred solution. This is achieved through development of requirements, abstracting the essential problem, establishing functional structures, searching for suitable working principles, and then combining those principles into a viable solution. Pahl et al. (2007) also states that this conceptual design phase results in the specification of a principal solution concept. These product conceptual design steps as described by Pahl et al. (2007) are illustrated in Figure 24.

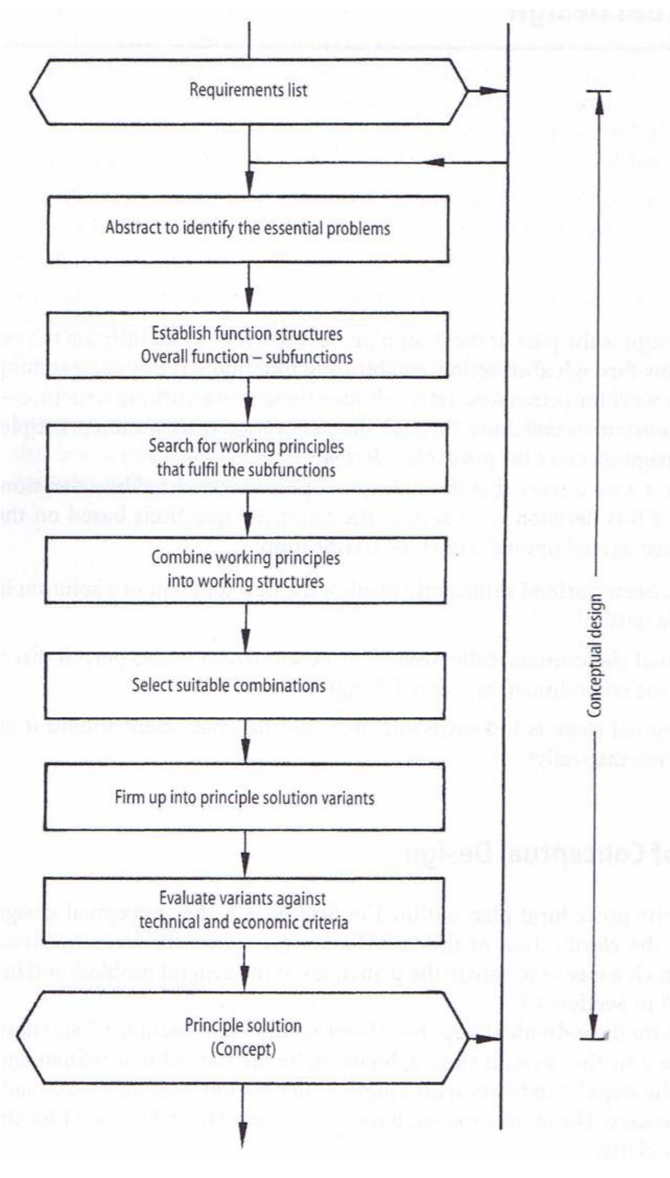


Figure 24. Steps in the conceptual design process
Pahl et al. (2007)

Pahl et al. (2007) also notes that optimisation of the principle solution also occurs in the concept design phase which is achieved through sub-processes which involve (1) searching for working principles, (2) combining working principles, (3) selection of suitable combinations, (4) firming up into principle solution variants, and (5) evaluation against technical and economic criteria. In describing these sub-processes, Pahl et al. (2007) outlines the use of morphological matrices as one means of generating and systematically combining solutions into a working structure. However, Pahl et al. (2007) also discusses the problems associated with this combinatorial approach which relate to ensuring the geometric and physical compatibility of the combined principles. That is, these functional elements are to be combined ensuring smooth flow of signals, energy, and materials. Pahl et al. (2007) also mentions an additional problem with this morphological technique being the evaluation of the large number of theoretically possible combinations and ensuring technical compatibility and economic viability.

The evaluation of solution variants, as described by Pahl et al. (2007) involves several steps as follows:

- Identification of evaluation criteria
- Weighting of evaluation criteria
- Compiling parameters
- Assessing values
- Determining overall value
- Comparing concept variants
- Estimating evaluation uncertainties
- Searching for weak spots

### 4.2.3 General aircraft design

Aircraft design is an example of a highly complex product, with Chapter 2 providing detail in relation to aircraft configuration design processes and the conceptual design phase. These aircraft design processes are encapsulated in various aircraft design texts by Raymer (2012), Torenbeek, (2013, 1982), Gudmundsson (2014), and Roskam (2002).

Raymer (2012), states that in the aeronautical context, aircraft design is about establishing the configuration through an iterative process moving to more sophisticated solutions using more sophisticated methods of analysis as the design progresses. Apart from the requirements, which should be independent of the design process, bi-directional interactions occur at all levels of the design process. It is noted that aircraft design methodologies are applicable also for design changes or modifications, where the same processes could be utilised.

#### *4.2.3.1 Conceptual design*

As stated in Chapter 2, aircraft conceptual design starts with the requirements. However, Raymer (2012) states that early aircraft conceptual design usually starts with a sketch of the aircraft configuration, which provides an approximation of the design layout including wing and tail geometries, fuselage shape and locations of major internal components. This process is called initial “sizing”. Optimisation techniques are then used to determine the lightest and lowest cost solution that will perform the mission and meet all performance requirements. The process then develops a revised layout and following a more detailed analysis and refined sizing and optimisation.

Torenbeek (1982), provides a generalised iterative design process as described shown by Figure 6 in Chapter 2. The principal phases are initiated by the requirements, formulation of a trial configuration, conduct of analyses, requirements comparison, configuration changes, and design optimisation. This general design process includes a convergence test to indicate those situations or cases where no configuration solution satisfies all requirements simultaneously. This convergence test evaluates certain requirements in the specification, constraints being contradictory or too extreme with respect to current technologies.

#### *4.2.3.2 Integrated approaches*

Price et al. (2006) describes the typical design process which follows a linear progression from requirements through conceptual design, preliminary design, and finally detailed design before the manufacturing phase is initiated. This manufacturing phase is seen as the receiver of the design activity as a deliverable. Therefore, the three central design phases dominate the technical design of a system, and these design phases being consistent with the description provided by Blanchard and Fabrycki (1998).

Price et al. (2006) describes a system view of the design process as shown by Figure 25 where the system is evolved in detail and complexity from initial requirements through to eventual production. Analysis is used in the design synthesis loop supporting systems development and there is continuous feedback to function and requirement process. This concept is a classical SE design evolutionary cycle.

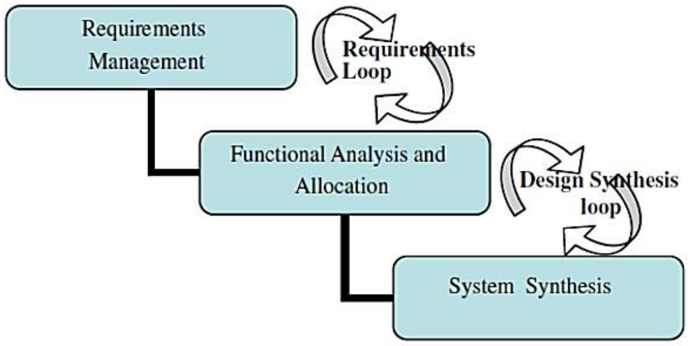


Figure 25. Systems engineering process model
Price et al. (2006)

Price et al. (2006) indicates that one of the challenges associated with SE theory is the identification of interfaces between subsystems and the modelling of the result of interactions between these systems. Aircraft are complex systems combining many subsystems at both physical and functional levels, and the identification and evaluation of such interactions is difficult. These subsystem interactions can be indirect and hidden from initial view, and the associated analysis capability may be insufficient to identify the behaviour accurately. This can lead to undesired or unexpected emergent behaviour of the system. The aim of the engineering analysis therefore is to develop a methodology that identifies the interactions of the subsystems by evaluating the performance of these subsystems in relation to inputs and outputs. Price et al. (2006) describes four key issues which were considered by the paper as being fundamental to addressing integrated design, these being:

1. reductionist versus holistic system design - The reductionist approach to SE involves decomposing the system into its subsystems, components and individual parts.
2. analysis fidelity - The issue of fidelity determined the appropriate models/methods to be utilised at each stage of the process.
3. system characteristics - A complex system is composed of multiple subsystems, with each subsystem having the required analyses carried out in order to derive their attributes.
4. simulation driven design environments - A series of simulation tools and models need to be made available to the user to ensure that a framework for a given product is easily and effectively developed.

### 4.2.4 Systems engineering lifecycle aspects

Blanchard and Fabrycki (1998) describe the engineering life-cycle of systems used to bring systems into being. This process begins with the definition of customer needs, extending this through requirements analysis, functional analysis and allocation, design synthesis, design evaluation, and system validation. This is achieved through an iterative process involving steps of analysis, evaluation, feedback, and modification.

Blanchard and Fabrycki, (1998) describe this system life-cycle by two major program phases, being acquisition and utilisation as shown by Figure 26. The acquisition phase involves three sub-phases covering conceptual-preliminary design, detail design and development, and lastly production and/or construction. It should be noted that under this model, the three main life-cycle design activities involve conceptual design, preliminary design and detail design, with the latter program phase involving utilisation which covers product use and disposal.

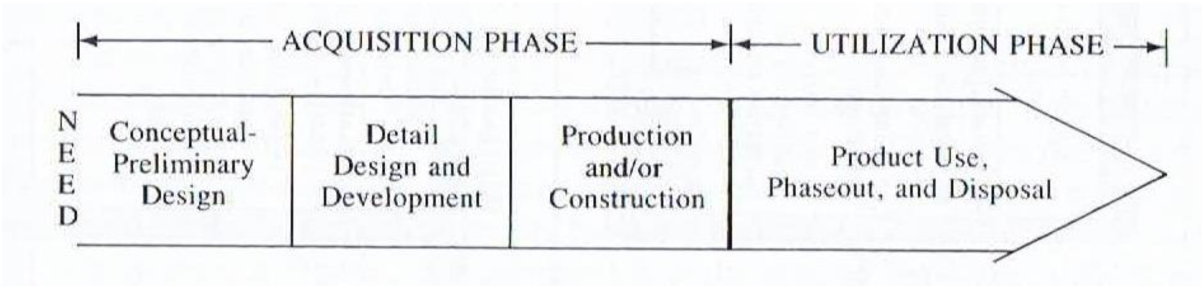


Figure 26. System life-cycle
Blanchard and Fabrycki (1998)

## 4.3 DESIGN SYNTHESIS

### 4.3.1 Needs and requirements

One method for facilitating early consumer-producer communications is QFD techniques, which involves construction of one or more matrices that describe requirements in terms of importance, technical solutions, and inter-relationships of attributes. The latter allows comparison of alternatives and for planning. This QFD technique described by Blanchard and Fabrycki (1998) is achieved by constructing a House of Quality (HOQ) diagram. An example showing the implementation of the QFD/HOQ approach is illustrated in Appendices 1 and 3 as part of the respective case studies. Blanchard and Fabrycki (1998) state that the functional analysis allocation process translates system requirements into detailed design criteria or metrics. This process involves abstraction of the needs and then breaking these down to identify requirements as described earlier.

### 4.3.2 Synthesis

As described earlier, Faulconbridge and Ryan (2014) state that synthesis or design is the process of creativity and technology adoption combined to produce a design that best meets the system requirements. Synthesis is an appropriate term used to describe this process as it hints to the evolutionary nature of design and development. In the early phases of systems engineering, given by conceptual design, synthesis is limited to defining completely the logical design or architecting the system, and then considering all possible technical approaches as described earlier. In this case, synthesis of the alternate fuel system or aircraft propulsion system modification is constrained by the architecture of the existing aircraft subsystems. Functionally the alternate fuel systems or propulsion system modifications are similar as a result of the common physical and functional attributes. An example of this synthesis process is shown for each case study in Appendix 1 and Appendix 3 respectively, where the formulation of morphological matrices is based on functional and physical characteristics of each alternate fuel/propulsion system modification.

### 4.3.3 Generating concepts

The next step in conceptual design is concept generation. Concept generation, according to Ullman (2010) can be achieved in many ways, and most commonly this occurs at the engineering requirements development stage, with a single concept, which is then developed and refined to a product. However, it is acknowledged by Ullman (2010) that this tends to be a deficient methodology, as it omits other potentially better concepts. Ullman (2010) discusses functional decomposition with concept generation along with concept variation techniques, with these techniques supporting a divergent-convergent design philosophy as also described by Ulrich & Eppinger (2012). In this context, Ullman (2010) provides a multi-step functional design process that decomposes the function, sub-functions, ordering of sub-functions and refining of sub-functions. Ullman (2010) also outlines practical methods or approaches to facilitate generation of concepts through various techniques such as Method 6-3-5 (a group-structured brainstorming technique), use of design analogies, brainstorming, patents, contradictions and the Theory of Inventive Machines (TRIZ). TRIZ is based on a problem-solving analysis technique derived from invention patterns in patent literature (Sheng & Kok-Soo 2010).

Pahl et al. (2007) outlines a systematic approach that is described by Ritchey (1998) as the morphological matrix, which is particularly useful in generating system solution concepts. This morphological matrix technique is applied in the case studies shown in Appendix 1 and Appendix 3 to generate alternate fuel systems or propulsion system modifications concepts. In these cases, the sub-functions are usually limited to the main functions only, and appropriate solutions are entered in the rows of the scheme. Pahl et al. (2007) suggests that if this approach is used for the generation of the overall solutions, then at least one solution principle must be chosen for each sub-function. That is a solution must be chosen in each row. To provide an overall solution, these sub-solutions must be then combined systematically into an overall solution.

If there are $m_1$ solution principles for the sub-function $F_1$, $m_2$ for the sub-function $F_2$, and so on, then after the completed combination there is $N = m_1.m_2.m_3...m_n$ theoretically possible overall solutions. Pahl et al. (2007) states that the main problem with this technique is the determination of compatibility of the solutions to ensure that the search field is narrowed down. In the case of those alternate fuel system modifications presented as case studies certain configurations were omitted as they adversely impacted certain design requirements or were discounted on account that they did not meet certain mission or role requirements. Pahl et al. (2007) further reinforces this by emphasising that:

- Only compatible solutions are combined.
- Solutions should be pursued only if they meet the requirements list and fall within the available resources.
- Promising combinations should be adopted with details provided to justify selection over other concepts

### 4.3.4 Technical Performance Measures and Metrics

The basis of this research is dependent on the formulation of appropriate performance, costing and sustainability metrics to support the development of a methodology applicable to alternate fuel and propulsion concepts. Implicit is the underpinning requirement to reduce fuel consumption and hence harmful environmental emissions, and to maintain or improve aircraft performance. These metrics are intrinsically linked. In general, most papers cited in Chapter 2 have studied the impact of various concepts and technologies using these metrics as the basis of an integrated analysis approach. However, this research sets out to determine the appropriate measures from which to undertake this life-cycle based evaluation, and indeed numerous papers Markish & Willcox (2002), Marx, Mavris & Schrage (1999), Mavris et al. (1998), Mavris & Kirby (1999), Ross et al. (2010), Schwartz & Kroo (2009) and Willcox (2005) have adopted a value-based approach to costing, albeit applied to specific aircraft design projects rather than modifications to existing aircraft.

The determination of metrics is therefore linked to requirements. In this case these requirements can be expressed as functional, performance and regulatory. For example, metrics aligning with aircraft and ground infrastructure requirements may address:

- Aircraft performance – Range, payload capability and cruise speed.
- Aircraft costs – Engineering, certification and operations.
- Aircraft emissions.
- Ground infrastructure performance – Fuel storage hold time, charging time and fill time.
- Ground infrastructure costs - Fuel or charging station cost and fuel delivery and/or supply.

### 4.3.5 Data collection

The data developed for this research project is associated with that used to validate the methodology. Data will be gathered for the case study pertaining to the aircraft and the associated modification subsystems and components. This data is derived from various sources including the authorised flight manual data and design literature sources in case of fuel system components. This data is used in simple models and parametric form using linear regression analysis based on existing designs. It is also used in custom developed morphological, change engineering and change certification matrices, along with costing and analysis tools associated with the selection and evaluation of concepts. This technical data is in the form of aircraft performance characteristics (e.g. cruise speed, range, fuel consumption), weight and balance data, operating limitations, general systems description and operating procedures. As stated above, this technical data is typically found in the approved (certificated) sections of the respective aircraft flight manuals. Other technical information relating to aircraft certification is found in the relevant Type Certification Data Sheet (TCDS). Data relevant to natural gas fuel tanks such weight, size, geometry and cost is available from product manuals and technical maintenance procedures and catalogues. Other data is available from the engine manufacturer, where powerplant maintenance manuals form the set of approved data for the aircraft type. Data relevant to electric aircraft propulsion systems is available from Original Equipment Manufacturers (OEMs) in the form of weights, power, efficiency and electrical capacity, supplemented with data reported in scientific and engineering journals.

As stated in the literature review there are a number of parametric costing methodologies which can be applied to validate the costing models developed for various life-cycle phases. Aircraft geometry, weight data and operating parameters derived from aircraft technical documentation as described above, can be used as an input to these parametric models.

## 4.4 EVALUATION OF CONCEPT ALTERNATIVES

### 4.4.1 Overview

Faulconbridge and Ryan (2014) state that “evaluation is the process of investigating trade-offs between requirements and design, considering the design alternatives, and making the necessary decisions to enable selection of solutions”. The process of evaluation is conducted throughout all stages of the system development life-cycle, determining whether the system satisfies the needs and requirements, and if so, to what level. Faulconbridge and Ryan (2014) state that a trade-off analysis is one of the techniques available to undertake this evaluation, with several steps involving:

- Definition of requirements,
- Identification of alternative solutions,
- Nomination of selection criteria,
- Determination of criteria weighting,
- Definition of scoring functions,

- Evaluation of alternatives, and
- Sensitivity studies.

The end result of evaluation is the selection of the desired candidate solution. Shortfalls that might be identified may result in further analysis and synthesis, with this applied iteratively throughout the design life-cycle.

### 4.4.2 Concept selection

Concept selection is a fundamental part of the product design and development process, where concepts are evaluated with respect to customer needs, requirements and other criteria. Ulrich & Eppinger (2012), describes this process as a comparison of the relative merits and disadvantages of the various concepts, and selecting one or two candidates for further investigation, testing or development. According to Ulrich & Eppinger (2012) the concept selection process is iterative and closely related to concept generation and testing. Concept generation can be applied by screening and scoring methods, which may include weighting of metrics or critical parameters. This concept generation and selection process is shown conceptually by Figure 27. What is inferred from this diagram is that time is portrayed horizontally and the number of concepts vertically. Therefore, as the conceptual design phase progresses, relatively few concepts are considered, increasing to many, where at some later, screening and scoring is used to reduce the concept number to one or two of the best candidates.

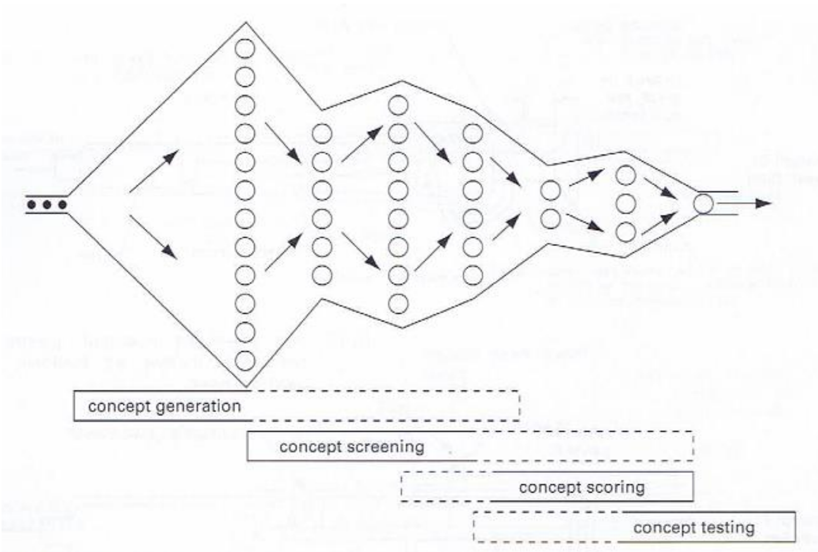


Figure 27. Concept generation and selection process
Ulrich & Eppinger (2012)

### 4.4.3 Pugh concept selection

The concept screening process is based on a methodology often called *Pugh concept selection,* first formulated by Pugh (1990). The purpose of this stage is to narrow the number of concepts quickly and efficiently and to improve the concepts if possible. The concept screening matrix rates the concepts against a reference concept using a simple code, which applies a “+” for better than, “0” for the same as, and “-“ for worse than in order to identify concepts for further consideration.

Concept scoring is applied when an increase resolution is required to better differentiate among the competing candidate solutions. In this stage, the relative importance of the selection criteria or technical performance measures are incorporated and refined comparisons are carried out with respect to these criteria. The concept scores are determined by weighted sums of the ratings using the following relationship:

$$S_j = \sum_{i=1}^{n} r_{ij} w_i \qquad \text{Equation 1}$$

Where $r_{ij}$ = raw rating of the concept $j$ for the $i$th criterion

$w_i$ = weighting of the $i$th criterion

$n$ = number of criteria

$S_j$ = total score of the concept

Ulrich & Eppinger (2012) states that the application of these concept screening and scoring stages reveals small impacts on results, and hence the techniques should be used with caution. For example, the concept selection method utilises selection criteria which is evaluated within an independent framework, although concept quality is the sum of the collective qualities relative to each criterion. Some concepts cannot be broken down easily into a set of independent criteria, therefore limiting the effectiveness of the method. Ulrich & Eppinger (2012) indicate that simply selecting a concept based on the sum of the performance attributes relative to each criterion may fail to capture emergent qualities. Furthermore, some selection criteria, such as those related to system aesthetics, are highly subjective, and decisions made between these alternatives should be made cautiously. Ulrich & Eppinger (2012) state that cost is an extremely important factor in choosing a concept and impacts the economic success of the project. For this reason, it is recommended that a cost metric be included when evaluating concepts, even though the costs may not be directly associated with customer needs and requirements. Similarly, there may be needs of other stakeholders that were not expressed by actual customers that are important and need to be included, such as those costs associated with regulatory compliance, and third-party infrastructure and facilities.

Ullman (2010) describes the process of evaluation as a means to refine a number of concepts before committing to one concept. In this context, Ullman (2010), uses the term evaluation which implies a comparison between competing concepts relative to the requirements they must meet. The results of evaluation provide the necessary data and information from which concept decisions can be made.

### 4.4.4 Change propagation impacts

In a complex system where all parts or subsystems are closely linked, modifications or changes to one part of the system are highly likely to result in a change to another part of subsystem. This in turn can propagate further throughout the system. Eckert et al. (2004) states that the greater the connectivity between subsystems or parts, the greater the likelihood that a change or modification to one subsystem leads to a change in other subsystems. In complex systems, a change rarely occurs without an impact to other systems, or subsystems. Furthermore, multiple changes interact with other systems or subsystems. This makes managing changes to complex systems a challenging conceptual design problem. It is only when the impact of the change has been fully predicted and understood, can resources be allocated to undertake the change as proposed. It should be noted also that conventional change analysis usually applies systems boundaries around the system and does not consider other external impacts. For example, changes to complex systems can also impact facilities requirements, logistics support, personnel and training requirements. These life-cycle impacts are often overlooked in treatment of change propagation effects.

#### *4.4.4.1 Change process*

Eckert et al. (2004) states that change processes should be considered in a broader context of three areas of research which address:

- Design studies, which is concerned with those areas associated with the design of new products as described by Cross (1994) and Pahl et al. (2007).

- Design reuse, which is concerned about the use of pre-existing designs, established design ideas or component parts. Reuse is well represented in research literature where it is treated differently in different fields of design, with this described by Eckert et al. (2000).
- Configuration management, which is concerned with managing changes on the level of subsystems or components, ensuring that their function and interfaces are maintained consistent over the type basis.

Eckert et al. (2004) examines further the inherent problems associated highly interconnected systems and the processes associated with changing them. Furthermore Eckert et al. (2004) extended this study to describe case study into helicopter modifications and the characteristics of changes observed. Two types of changes were distinguished with different causes and similar processes. These changes were (1) an initiated change, which arises from new customer requirements, and (2) an emergent change, which responds to deficiencies in the product.

Eckert et al. (2004) interviewed several engineers within a helicopter manufacturing organisation and established that system complexity, in terms of the number of parts and relationships between them, was determined as a major source of emergent problems. It was apparent that no one person had a detailed overview of all the systems in the helicopter, such that they could assess the impact of proposed changes and its likely cost and consequences. Furthermore Eckert et al. (2004) interviewed designers, and found that they typically expect up to four subsequent changes arising from each initiated change. Therefore, it is necessary to be aware not only of individual change but also chains of complex networks as illustrated by Figure 28.

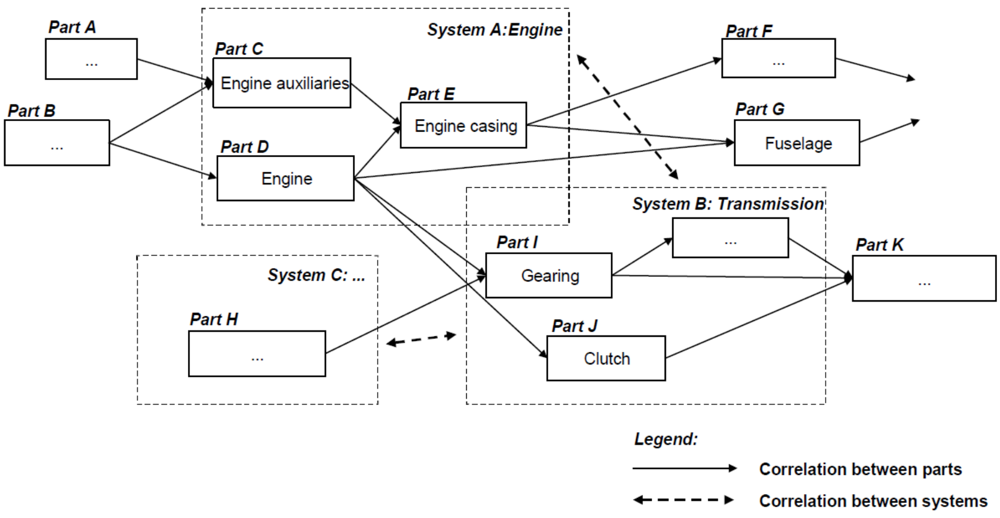


Figure 28. Change networks through change propagation
Eckert et al. (2004)

Eckert et al. (2004) states that these changes can be expressed more concisely in the form of a change matrix, which is similar to a Design Structure Matrix (DSM). Figure 29 shows an example of a part of a change matrix showing the likelihood of change of one system as a consequence of a change to another. For example, Eckert et al. (2004) states that when helicopter engines are changed, there are inevitably changes required to the fuselage, the gearbox, avionics and engine auxiliaries (each with a likelihood value of 1, and many other systems not shown. However, a change in the gearbox is very unlikely to result in a change to the engine (likelihood value of 0.05). These likelihood values may be determined from previous design changes and from the experience of senior designers. For this reason, the change propagation methods described here are best conducted as a design team activity.

| | Bare fuselage | Engines | Transmission | Avionics | Engine auxiliaries | ... |
|---|---|---|---|---|---|---|
| Bare fuselage | | 1 | 1 | 0.02 | 0.05 | |
| Engines | 0 | | 0.05 | 0 | 0.5 | |
| Transmission | 0 | 1 | | 0 | 0.01 | |
| Avionics | 0 | 1 | 0.02 | | 0.01 | |
| Engine auxiliaries | 0 | 1 | 0 | 0 | | |
| ... | | | | | | |

Figure 29. Likelihood of change between various helicopter systems
Eckert et al. (2004)

Figure 30 further shows some of the interaction relationships in more detail for this example. A more powerful engine, which is usually heavier, might require more structural reinforcing to the fuselage. It might also require a new cowling because of the engines larger physical size, which is an interaction between the system and subsystem. Furthermore, this increase in size impacts overall drag on the bare fuselage affecting performance. And a larger more powerful engine may require a larger transmission to transmit the increased power, and further structural reinforcement to the bare fuselage. The impact of structural reinforcement to the bare fuselage will increase weight and therefore impact the payload potential of the helicopter. This example illustrates the cascading effect of a modification on what is a simplistic model of a helicopter. The same can be said for aircraft as another example of a highly coupled complex system.

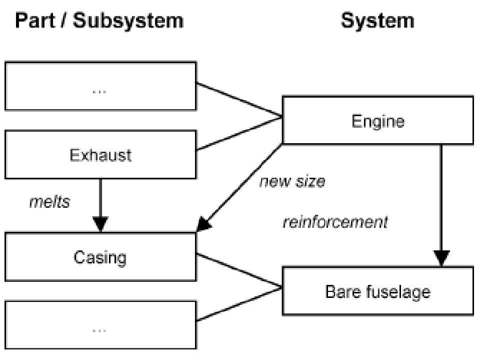


Figure 30. Relationship between components and systems
Eckert et al. (2004)

#### 4.4.4.2 Linking parameters

As indicated by Eckert et al. (2004) the functional subsystems in engineering systems create, transmit and transform both intended and required quantities, such as fuel, and unintended quantities such as vibration and heat. Complex systems can be thought of as comprising three types of flows. These flows as proposed by Eckert et al. (2004) are “flows of matter, flows of energy and flows of information and data”. These flows show that linking between parts and systems includes geometry, force, torque, temperature, heat transfer, mechanical vibrations, electromagnetic radiation and material parameters. Eckert et al. (2004) states that these linking parameters can invoke further changes, and may change themselves during the change process. This can be illustrated by the simple example described earlier, where a larger more powerful engine is installed in a helicopter. Therefore, increasing engine power = torque x angular velocity results in changes of physical parameters. This might require an increase in output shaft diameter (geometry change), and hence a new engine housing (geometry change).

However, it is not possible to make predictions of change behaviour of a subsystem or component based solely on the parameters. Eckert et al. (2004) states that it is necessary to investigate the characteristics of the system itself in terms of the properties associated with the way changes are absorbed, carried forward or multiplied as quoted below.

- *Absorbers* – These subsystems or components have properties that can absorb more change than they introduce. Typically, a very small number of subsystems are total absorbers. These absorbers potentially reduce the overall complexity of the change problem within a system.
- *Carriers* – These subsystems have properties that take the same amount of change as they introduce themselves, and they do not increase the complexity of the overall change problem. Simple geometric components fall within this category such a bracket that remains physically the same as a result of the introduced change.
- *Multipliers* – These subsystems have properties that generate more changes than they introduce. Change propagation becomes more

complex through multipliers and unexpected change avalanches can arise.

Eckert et al. (2001) notes that change propagation behaviour is not a static characteristic of a system, and it depends on the design state. A change absorber can become a multiplier if the change is too large to absorb. For example, a helicopter engine may be able to absorb a certain increase in gross weight. However, if the weight increase is significant enough, then the engine must be modified, or an engine with increased power be selected.

The key therefore to change propagation prediction within complex systems is understanding the tolerance and margins of key parameters. Eckert et al. (2001) states that in practice, these tolerances and margins are often not known. They are often documented in design reports when the design is undertaken and are kept within the design organisation as proprietary data. In reality, the real margins are not known, because the initial design decisions are often based on previous experience and the application of design manual factors and practises. Furthermore, certification often requires testing to show compliance, rather than to determine performance exceedance. Testing components and subsystems to limits is expensive, while computer analysis methods such as Finite Element Method (FEM) have improved, the models still do not have fidelity to predict interconnected properties of complex systems. For example, in helicopter development vibration problems are anticipated by computer modelling but are often eliminated through later developmental testing. The design planning effort therefore reflects this approach and allocates resources to vibration problem and fault-finding during the prototype testing and development phase.

#### *4.4.4.3 Macro level views*

Eckert el al. (2001) provides a macro-level account of the change process which illustrates the extent that a change to a single component can impact many parts and subsystems of a system. A study conducted on change highlighted different change processes types depending on the number of impacted components with a single change process. It is therefore critical that changes be managed in such a way that the change effort be completed within the required timeframe. Eckert el al. (2001) describes changes that are completed on time can be further divided into two broad groups as follows:

1. Change *ripples* and change *blossoms* result from changes that generally decrease the change effort as time progresses. Change *blossoms* result in in a change effort which increases rapidly before decreasing to finish on time. An example of a change *ripple* may be modifications to aircraft cabling and wiring. Whereas change *blossoms* may be a number of changes that are ended within expected time limits. An example may be the routine modification to the fuselage.
2. Change *avalanches* are processes that extend beyond the project time limits, where the volume of changes, and level of change effort increases steadily. A change avalanche may be that associated with a major change that gives rise to equally major changes. An example may be the installation of crashworthy troop seating in the cabin of a helicopter, where structural reinforcing is required to the fuselage frames and beams, re-routing of electrical cables, and revised structural attachment points for role equipment such as guns and tie-down equipment etc.

These types of changes are illustrated in Figure 31.

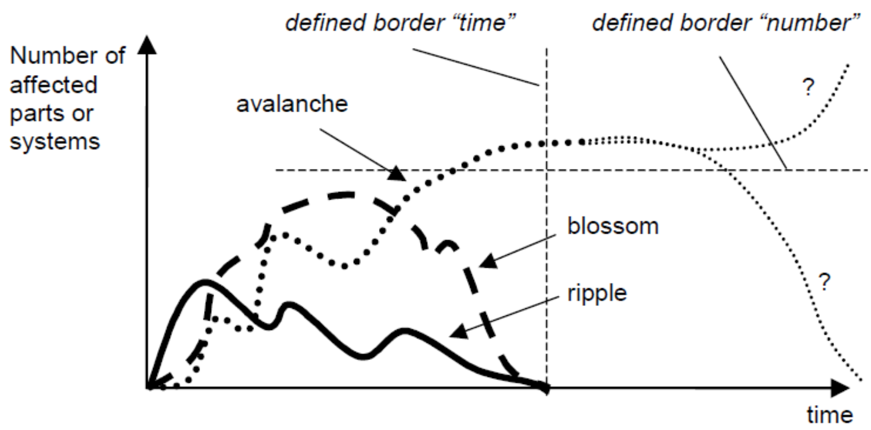


Figure 31. Macro view on change
Eckert et al. (2001)

### 4.4.5 Design evaluation

Design evaluation is an essential step within the evaluation process of design alternatives. Blanchard and Fabrycki (1998) states that a design alternative is a projection of what could be and how well the design alternative might be if chosen for further development. Design evaluation is preceded by system analysis, which in turn is preceded by synthesis in an iterative process.

Blanchard and Fabrycki (1998) states that the first step in an evaluation activity is to establish a baseline against which a design candidate can be compared. This baseline is derived via an iterative process of requirements analysis, with the functions that the system must perform described, along with the technical performance measures. Both the operational (airborne segment) and maintenance and support functions (such as ground segments) must be described at the top level. As part of this process, it is necessary to establish systems metrics that describe performance, cost, effectiveness and other such quantitative factors as required to ensure that the customer needs are met. Some of these metrics are considered to be more important than others by the customer, which will in turn influence the design process in placing different weightings on the selection of design criteria. The result is the identification of TPMs for the system overall. In the case of aircraft alternate fuel system modifications, these TPMs can be classified as metrics dealing with performance (range, payload and cruise speed), structural weight, fuel weight, operational and procurement costs, and emissions.

With the applicable TPMs defined at the system level, Blanchard and Fabrycki (1998) states that the next step is to determine the specific properties that must be merged into the design itself. As stated earlier, the functional decomposition of an alternate fuel system modifications or propulsion system modifications is already defined by similar architectures. DDPs are identified, analysis and trade-studies are conducted by considering various design concepts, design synthesis is undertaken, and the iterative process of design evaluation takes place. This process flows down to the appropriate system level to ensure that the system configuration meets customer needs and requirements.

Blanchard and Fabrycki (1998) states that these TPMs can be prioritised at the top level to reflect overall performance characteristics in relation to the mission objectives with an example design consideration hierarchy shown by Figure 32.

System value is shown as a first order consideration, with economic factors and technical factors comprising second order considerations. Technical factors may be expressed in terms of systems effectiveness, whilst economic factors may be broken down into revenues and life-cycle costs as shown in Figure 32. Systems effectiveness leads to such third order considerations which are a function of performance, availability, supportability etc. Assuming that performance represents a high priority in design, such features as size and weight should be stressed in the design. Conversely, if life-cycle costs such as operational costs or procurement costs represent high priority, or are representative of costs of the system, then these should also be stressed in the design. Thus, the criteria for design and the associated DDPs (such as aerodynamic drag) may be established early in the conceptual design process and carried through the entire design cycle.

It is important to note that this design evaluation activity is iterative and continues through system-level, subsystem level down to component level as stated by Blanchard and Fabrycki (1998). However, in the case of aircraft design modifications this process can be abbreviated as described in this thesis, as design synthesis provided by functional and requirements analysis are established by pre-existing fuel system architecture and common aircraft systems interfaces.

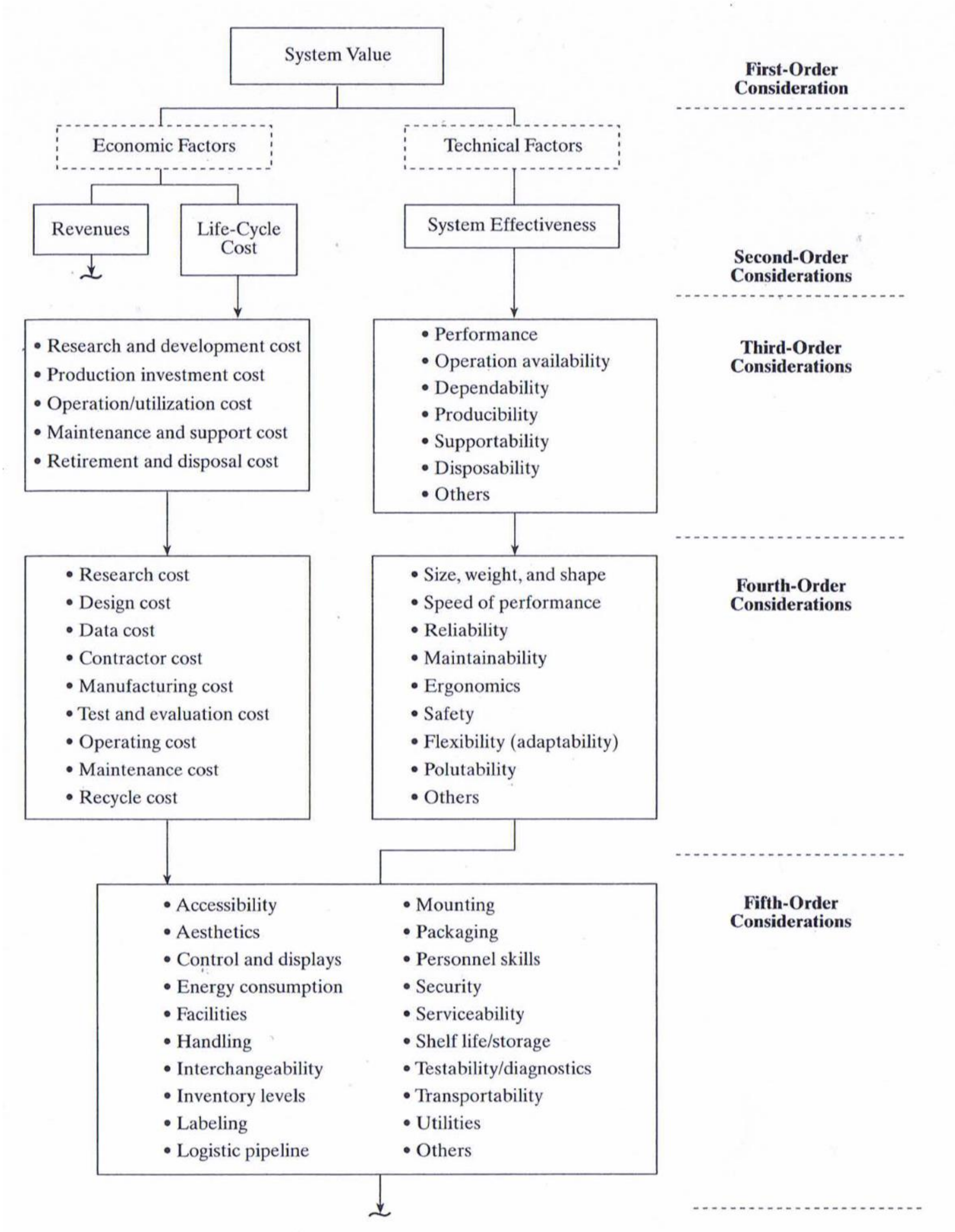


Figure 32. Design consideration hierarchy
Blanchard and Fabrycki (1998)

## 4.5 ANALYSIS AND OUTPUTS

### 4.5.1 Requirements

The analysis process as described by Faulconbridge and Ryan (2014) starts with establishing the business and project needs. Within conceptual design, the analysis process characterises these needs, and identifies those essential requirements of the system. Requirements analysis activities continue throughout the development life-cycle to develop lower level requirements associated with the physical and functional attributes of the system design. The allocation of requirements forms an essential description of the system elements and architecture, and therefore supports the synthesis process.

### 4.5.2 Systems analysis process

As the system design develops, there are numerous trade-offs involving the evaluation of different technologies, alternative system schemes, manufacturing processes and logistic support strategies. Blanchard and Fabrycki (1998) state that the trade-off studies lead into synthesis which refers to the combination of subsystems and components in such a way as to represent a feasible system solution.

Blanchard and Fabrycki (1998) states that generic systems analysis within conceptual design comprises processes that involve trade-offs, break-even points, sensitivity studies (inclusive of risks and uncertainty) and subsequent recommendations. These processes utilise technical performance measures or metrics derived from the requirements and in the case of those alternate fuel modifications these metrics will relate to aircraft performance, costs and sustainability.

### 4.5.3 Outputs

#### *4.5.3.1 Safety*

Because safety is a very important aspect of any product, military operations have adopted a standardised approach to systems safety. The military standard MIL-STD-882 – Standard Practice for Systems Safety (2012) focuses specifically on safety of military equipment and hardware, including aircraft and related systems. This standard provides a simple method in dealing robustly with any hazard during design through to operation and support, including health hazards. MIL-STD-882 characterises a hazard by the combination of likelihood or probability of occurrence, and the consequence if that hazard eventuates.

The US Federal Airworthiness Regulations (FAR) have similar structures for ensuring safety equipment installations on civil aircraft. These regulations, such as FAR Part 23 (and other FARs), provide airworthiness standards for small aircraft. For example, FAR Part 23 has a requirement §23.1309, which deals with systems, equipment, and installations. This requirement has a corresponding Advisory Circular (AC) 23.1309-1E (2011) which described the acceptable means of showing compliance §23.1309, for equipment, systems, and installations in FAR Part 23 aircraft. Although this AC is not mandatory, and is issued for guidance only is does present a method of compliance which deals with system safety analysis and assessment of aircraft and equipment installations.

#### *4.5.3.2 Risks*

Project risk is the next area of risk that relates to schedule, budget, or other factors adversely affecting the project progress. Project risks may be also related to uncertainties associated with technology readiness; material processes not being available at the price, or behind schedule; inadequate or resources becoming unavailable; and vendor supply issues or shortfalls.

Decision risk as stated by Ullman (2010) are those chances that choices may not eventuate or turn out as expected. These decisions are calls to action which commits to resources, and it is only later that a decision can be determined to be good or bad. Again, the decision-making risk is a probability that a poor decision has been made, in combination with the consequences of the outcome. However, the important difference in this risk assessment is that there is no real measure of uncertainty in the Decision matrix.

## 4.6 MATRIX-BASED DESIGN METHODOLOGY

### 4.6.1 Overview

In order to combine the elements of design synthesis, evaluation of concept alternatives and analysis of outputs, this thesis has formulated a matrix-based conceptual design methodology. This methodology implements the concepts of design synthesis, evaluation and analysis as an iterative process, building and linking together existing techniques such as Quality Function Deployment (QFD) matrices, Gavel et al. (2006, 2007) and Ölvander et al. (2009) Quantified Morphological Matrix (QMM), Pugh's decision matrices, Koh et al. (2012) Change options Multiple Domain Matrix, and a simplification of Clarkson et al. (2001) Change Propagation Method (CPM) matrices. The methodology is extended however to develop Engineering and Certification DMM techniques, based on Design Structure Matrices (DSM) to evaluate the impact of design modification changes on engineering and certification risks and costs.

The outputs of this process are those artefacts that are used as inputs to the next preliminary design phase of the systems design life-cycle. In this instance, requirements analysis and functional decomposition steps are significantly simplified as result of a defined alternate fuel system or propulsion system modification. That is, this aspect of the methodology uses systems functions as defined by the pre-existing aircraft system being modified, as well as the supporting infrastructure, and hence the focus is to ensure that user needs and requirements are reflected and flowed through this methodology. This is captured in the early steps through application of QFD and Pugh pairwise comparison techniques. Generation of systems concepts is achieved through application of morphological matrices where the system requirements analysis and functional decomposition steps are imbedded in the formulation of this matrix. This morphological technique has the benefit that the complete design space is explored with all combinations of systems/subsystems functions considered. The challenge is to consider all potential solutions, which could be a sizable set, and reduce this down to a manageable subset for further consideration as discussed by Ulrich & Eppinger (2012). This is partially achieved through quantisation of the morphological matrix as described by Ölvander et al. (2009), Gavel et al. (2006, 2007) and others. In this instance, this Quantified Morphological Matrix (QMM) approach was applied using simple relationships representing TPMs (or metrics) and then rating these as metrics to reduce the solution space.

One focus of this conceptual design methodology is the evaluation and assessment of the impact of changes and how they propagate as a result of alternate fuel system or propulsion system modification. There are two aspects to this evaluation, being an assessment of change options as applied through Koh et al. (2012) MDM techniques, and an assessment of change propagation impacts to aircraft systems, subsystems and components as described by Clarkson et al. (2001) and others. The Koh et al. (2012) change options technique is applied early in the methodology to assess and feedback requirements that are of the most importance to the design. The change propagation methods as described by Clarkson et al. (2001) are implemented to assess the impact of changes brought about by the modification, and is a simplified representation of the Change Prediction Method (CPM) to highlight risks. These changes are assessed using change propagation matrices which capture change severity risks at subsystem, component and design code levels. These risks are recorded in a traditional risk matrix and are visualised via change propagation tree and case risk plots, in order to inform the preliminary design phase.

The latter steps of this design methodology apply DMM-based techniques, as described by Koh et al. (2012), to evaluate engineering and certification impacts of changes resulting from the alternate fuel system or propulsion system modification. In this instance change propagation techniques as described by Clarkson et al. (2001) and others are applied to these engineering and certification DMMs to determine the impact of the modification. The engineering DMM assesses these changes at the system requirements, subsystems and component level, using as a basis, the requirements and design changes documented in the QFD matrix and Change options MDM, and the results of the CPM. The certification DMM follows a similar approach, with the main difference being that change impacts are evaluated against the respective airworthiness design standard whether it be aircraft, engine or any other applicable standard or code. The main benefits of such an approach is that all airworthiness requirements are assessed for the impact of the modification, and change severity risks are determined accordingly.

As described above, the outputs of this conceptual design methodology are artefacts that are used as inputs to the preliminary design phase. It is important to note that this methodology does not extend into the regulatory domain past the draft Certification Program Plan (CPP) document as provided by the certification DMM, nor does this methodology extend into project management artefacts, which would traditionally accompany such conceptual design phases. However, it must be said, that that outputs of the methodology could be extended to provide inputs into development of initial Work Breakdown Structures (WBS) and schedules if required.

The following sections provide an outline of theory and structures underpinning this conceptual design methodology following a general systems synthesis, evaluation and analysis structure.

### 4.6.2 Abstraction

The process associated with the development of this conceptual design methodology follows a system engineering life-cycle approach as described previously. At each step, formulation of approaches and analyses shall be undertaken, each with specific outcomes. However, it is important to note that the procedure is iterative, which is reflected in the formulated methodology.

It is also important to note that the intent here is to determine concepts and solutions early in the design life-cycle to enable a relatively detailed evaluation of design performance, design change risks and associated costs. That is, the methodology must be able to generate system solutions and then reduce these to a viable and usable solution set that can be evaluated for any particular solution configuration then down-selected to a particular solution. In this way, use of the optimisation methodologies are minimised and involve the identification of the best solution using TPMs (or metrics) and constraints determined by applicable requirements. One such approach is the adoption of general morphological analysis (GMA) techniques. The adoption of such GMA techniques is an ideal tool to generate concept options and can be further extended to include evaluation of these options via quantisation as described later in this section.

The general process followed is similar to the Technology, Identification Evaluation and Selection (TIES) methodology as described by Kirby (2001), noting that as indicated above, the similarity, and the effect of technology will necessitate provision of cost and value estimates. The TIES methodology as described by Kirby (2001) focuses on the application of a set of technologies for a single vehicle concept and the identification of the highest value technology combinations. The method is a nine-step process shown in Figure 7. The process begins with defining the problem in terms of the customer requirements, to selecting the best family of alternatives (in terms of design attributes and technology sets), in order to best satisfy customer requirements.

This life-cycle based methodology will consider aspects of the fuel system or propulsion system modification including those attributes associated with new support infrastructure development as required. This methodology therefore encompasses the modification development, test, certification, acquisition and operations space. Further, the methodology incorporates tools and/or techniques to undertake decision making, evaluation of risks and costs associated with the integration of new fuel types or propulsion concepts. It is therefore possible that the results of this study could be used to determine the viability and development strategies to establish the “best” (for planet, profit and performance) design solution.

#### 4.6.3 Quality Function Deployment matrix

One method for facilitating early consumer-producer communications is QFD techniques, which involve construction of one or more matrices that describe requirements in terms of importance, technical solutions, and inter-relationships of attributes. This QFD technique described by Blanchard and Fabrycki (1998) and Gudmundsson (2014) is achieved by constructing a House of Quality (HOQ) diagram. Blanchard and Fabrycki (1998) state that the functional analysis allocation process transposes system requirements into detailed design criteria. This process involves abstraction of the needs and then flows this down to identify requirements which is described later in Section 4.6.4. This functional analysis is achieved through use of functional flow block diagrams which breakdown system high level functions, to second level functions and third level functions. However as discussed earlier, details of this functional analysis is not presented in this thesis as this is a standard SE process.

Once the top-level description of the system is defined in functional terms, the next step involves functional allocation. This functional allocation as described by Blanchard and Fabrycki (1998) groups similar functions into logical sub-divisions or groups through identifying major subsystems and lower level components of the overall system. With this approach, the system is broken down into components. This structure serves as a framework for preliminary design and evolves from the development of TPMs and Design Dependent Parameters (DDPs) which can be allocated to the appropriate system element as discussed in the following section.

### 4.6.4 Morphological matrix quantisation

A variation on this morphological technique has been applied to a fuel system configuration design on fighter aircraft as described by Gavel et al. (2006, 2007), Gavel et al. (2008), Ölvander et al. (2009), and Svahn, (2006). In these accounts, matrix-based methods were employed to quantify the morphological matrix providing a solution which is characterised with a set of parameters such as system weight, cost, performance etc. The selection of the candidate solutions are modelled with decision variables and the selection and optimisation problem is formulated within a mathematical framework as described below. The quantification of the morphological matrix provides access to every potential solution which is described as either a physical or statistical model, or a combination of both. Therefore, using this approach the TPMs are quantified as metrics.

In the following example provided by Ölvander et al. (2009) the TPMs provided by Cost (C) and Weight (W) are important to the conceptual design of an aerospace vehicle.

As described by Ölvander et al. (2009) the morphological matrix X can be stated as *n* different functions with M potential solutions for each function, resulting in a matrix X as follows:

$$X = \begin{bmatrix} X_{11} & \cdots & X_{1m} \\ \vdots & \ddots & \vdots \\ X_{n1} & \cdots & X_{nm} \end{bmatrix} \qquad \text{Equation 2}$$

The concept, as reproduced here from Ölvander et al. (2009), relies on determining one solution to fulfil one function only. This can be expressed by letting $x_{ij}$ equal 1, if solution *j* is selected to implement function *i*. Otherwise in the remaining cases, 0 applies. Therefore, in each row there can be only one element different from zero with this relationship implemented. This is shown below by the following:

$$\sum_{j=1}^{m} x_{ij} = 1, \;\; i = 1 \dots . n \qquad \text{Equation 3}$$

$$x_{ij} \;\; \in (0, 1)$$

For this matrix, system weight, $W$, is calculated by summing the weight of each solution as shown below.

$$W = \sum_{i=1}^{n} \sum_{j=1}^{m} w_{ij} x_{ij} \qquad \text{Equation 4}$$

Therefore, the weight of a specific solution, $w_{ij}$ is calculated as function of the specific constraints implied by the system as well as the system specific parameters defined in the vector ***y*** defined below. However, weight is also a function of the chosen concept ***X.*** This allows for dependencies where for example the weight of one solution may be also dependent on other solutions. This is particularly relevant in the case of alternate fuel systems where the weight of the fuel tank may be dependent on the fuel state (liquid or gas) or battery type selected. That is a concept where the selected gaseous fuel may require a heavier and larger fuel tank to achieve a range requirement (which may be represented in ***y***). Therefore, the weight of a particular solution could be determined according to the equation below.

$$w_{ij} = w_{ij}(X, y) \qquad \text{Equation 5}$$

Where $X$ is the chosen concept, and

$y$ is the specific system parameter vector

It should be noted that the above equations yield a non-linear expression for total weight and that the total weight meets the requirements in ***y***.

As indicated by Ölvander et al. (2009), this system weight solution may be minimised as an optimisation problem within the following relationships:

$$min \sum_{i=1}^{n} \sum_{j=1}^{m} w_{ij}\, x_{ij}$$

Such that

$$\sum_{j=1}^{m} x_{ij} = 1,\ i = 1 \ldots . n \qquad \text{Equation 6}$$

$$w_{ij} = w_{ij}(X, y)$$

$$x_{ij} \in (0, 1)$$

It should be noted that this approach provides numerous infeasible solutions, and therefore this method requires the introduction of a set of feasible solutions *S*, in which to explore for feasible solutions. In simple cases $x_{ab}$ is incompatible with $x_{cd}$ could be expressed as:

$$x_{ab} + x_{cd} \leq 1 \qquad \text{Equation 7}$$

The relationship described as $x_{ef}$ requires that both $x_{gh}$ and $x_{ij}$ are within the concept and could be modelled as follows:

$$2x_{ef} - x_{gh} - x_{ij} \leq 0 \qquad \text{Equation 8}$$

If there are only $m_i$ solutions for the function *i*, then the remaining elements $x_{mi+1}$ – $x_m$ should be set to zero as described by the following:

$$x_{ij} = 0, i = 1 \ldots . n, j = m_{i+1 \ldots .} m \qquad \text{Equation 9}$$

Furthermore, there may be many other system attributes that might need to be included when evaluating candidate solution concepts. Ölvander et al. (2009) illustrates this by inclusion of the cost attribute *C*, in the same manner as weight, *W*. This can be expressed as the following function, where $\alpha_1$ and $\alpha_2$ are linear weightings for the objectives $w_{ij}$ and $c_{ij}$.

$$min\ \alpha_1 \sum_{i=1}^{n} \sum_{j=1}^{m} w_{ij}\, x_{ij} + min\ \alpha_2 \sum_{i=1}^{n} \sum_{j=1}^{m} c_{ij}\, x_{ij} \qquad \text{Equation 10}$$

Such that

$$\sum_{j=1}^{m} x_{ij} = 1, \qquad i = 1 \ldots . n$$

$$w_{ij} = w_{ij}(y, X) \qquad (10)$$

$$c_{ij} = c_{ij}(y, X) \qquad (11)$$

$$x_{ij} = 0, i = 1 \ldots . n, j = m_{i+1 \ldots .} m$$

$$x_{ij} \in (0, 1)$$

$$X \in S$$

An alternative formulation is obtained if one decision variable is used for each function. That is for each row in the decision matrix there is one variable that can be optimised and applied to the integer values representing different solutions for that function. Thus, there are $n$ decision variables which can take integer values from 1 to $m_i$, where $m_i$ is the number of solutions for the function $i$, as shown by:

$$x_i = \{1, 2, \ldots., m_i\}, i = 1, 2, \ldots., n \qquad \text{Equation 11}$$

$$X = [x_1, x_2, x_3 \ldots. x_n]^T$$

By adopting this approach, the formulation will have $n$ integer variables instead of $n.m$ binary variables. This is referred to as quantification of the morphological matrix.

The weight $W$ of the concept is therefore calculated by summing up the weights of the functions. However, it can be observed that the weight required to determine function $i$ is obviously a function of the adopted solution. That is $w_i = w_i(x_i)$. It will depend also the solution selections within the concept, that is $w_i = w_i(x)$. Weight also is a function of external requirements $y$, so that $w_i = w_i(x, y)$ as shown below:

$$W = \sum_{i=1}^{n} w_i \qquad \text{Equation 12}$$

$$w_i = w_i(x, y)$$

The lowest possible minimum weight can thus be determined by:

$$min \sum_{i=1}^{n} w_i$$

Such that

$$w_i = w_i(x, y)$$

$$x_i = \{1, 2, \ldots., m_i\}, i = 1, 2, \ldots., n$$

$$X = [x_1, x_2, x_3 \ldots. x_n]^T$$

There may be characteristics of the system that need to be considered when optimising or evaluating the solution concepts. In other design studies, Ölvander et al. (2009) includes other important design parameters such as electrical power consumption and compressed air consumption. Ölvander et al. (2009), indicates that electrical power consumption, $p_{ei}$ and compressed air consumption, $p_{airi}$ may be handled much in the same way as weight. Similarly, in the context of alternate fuel systems, other parameters may be included such as drag increment and cost of the modification.

The objectives can be therefore be aggregated to an overall objective function, as follows, where each objective is weighted by the α

$$min\alpha_1 \sum_{i=1}^{n} w_i\,(x,y) + \alpha_2 \sum_{i=1}^{n} p_{ei}\,(x,y) + \alpha_3 \sum_{i=1}^{n} p_{airi}\,(x,y) + \alpha_4 \frac{1}{\sum_{i=1}^{n} \frac{1}{\lambda_i(x_i)}} + \psi$$

Where ψ is the penalty function which is zero if the concept is within the feasible solution space and >0 if it is not. Ölvander et al. (2009) states that depending on the characteristics of the problem, and the optimisation method employed, then a binary or integer representation may be the best choice. The framework associated with this approach is provided in the next section.

### 4.6.5 Pugh matrix

Pugh's method (Pugh 1990) or decision matrices is a relatively simple and proven approach that can be used for comparing alternative concepts. Pugh's decision matrices as described by Burge, (2009) involves a step by step approach which is analogous to a QFD diagram. Each step fundamentally scores each alternative concept relative to the others by reference to established criteria. One of its key advantages is its ability to handle many different types of decision criteria. The Pugh decision matrix method is iterative in its implementation, and tests the completeness and understanding of the criteria, and identifies or confirms the best candidates. Pugh's method is particularly effective if implemented and conducted independently by each member of a design team. Ullman (2010) provides a comprehensive outline of the steps and supporting information describing the construction of Pugh decision matrices. These main steps are summarised below:

1. Identification and definition of the criteria for selection.
2. Identification of one candidate design option as the baseline and scoring of all requirements against the baseline.
3. Comparison each candidate design option against the baseline, stepping through criteria, and deciding on pair-wise scores.
4. For each candidate design option, determining the total score to identify the best candidate.

5. Having scored each candidate design option consider hybrid options by combining where possible the best from each alternative.
6. Make the decision in relation to the best candidate, and record rationale.

Pugh’s decision matrices can have limitations as described by Burge (2009) which are related to the adequacy of selection criteria. These limitations may include (1) Incorrect, incomplete and inadequate selection criteria, (2) Granularity of pairwise scale, and/or (3) Wrong expertise and insufficient experience in design teams.

### 4.6.6 Change options Multiple-Domain Matrix

The paper by Koh et al. (2012) presents a modelling method that supports the prediction and management of change propagation during the design of complex engineering systems. Like the work undertaken by Clarkson et al. (2001), and Eckert et al. (2001, 2004 & 2006), this thesis builds on the QFD method and the Change Prediction Method (CPM) to model the effects of potential change propagation as a result design changes or modifications. A framework proposed by Koh et al. (2012), extends these methods into different description domains. Hence a Multiple-Domain Matrix (MDM) is proposed by Koh et al. (2012) to better illustrate how dependences between the different domains are modelled. It is noted that this MDM approach is essentially a combination of DSM and Design Mapping Matrices (DMMs). These DMMs are non-square matrices which serve to link related information across different domains. Koh et al. (2012), states that the diagonal of the MDM are DSMs, while the rest of the fields are DMMs. Figure 33 illustrates the modelling method and structure using MDMs, with the various fields denoted as A through E.

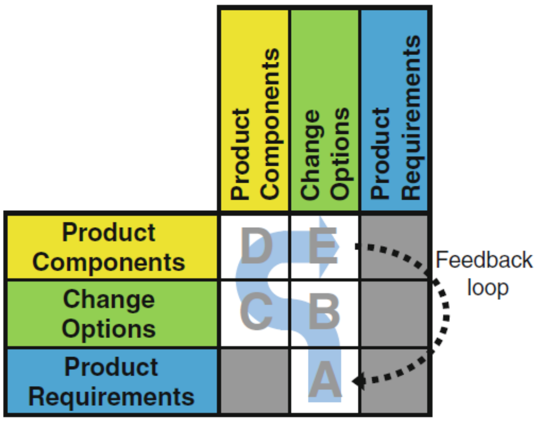


Figure 33. Modelling method using Multiple-Domain Matrices
Koh et al. (2012)

The four steps of this method described by Koh et al. (2012) are summarised in Figure 34, with these steps broadly corresponding to the dependences modelled in each Field labelled as A, B, C, D and E within the Multiple-Domain Matrix (MDM).

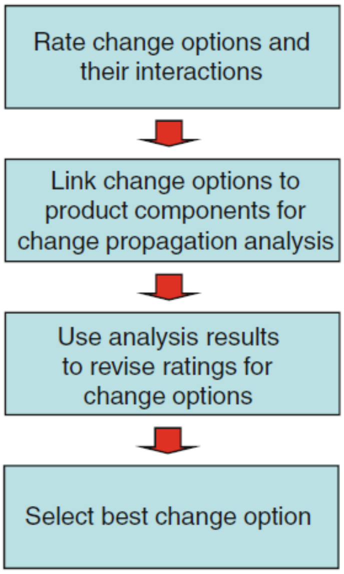


Figure 34. Framework of the change modelling method
Koh et al. (2012)

This change options analysis has been implemented as a specific step in the conceptual design methodology in this this thesis in accordance with the approach described by Koh et al. (2012). Given that the underpinning theoretical background can be found in Koh et al. (2012), full details and descriptions of the method will not be reproduced here. Rather the emphasis is implementation of step 4 involving application of this method to the case studies shown in Appendices 1 and 3.

### 4.6.7 Design Structure Matrices applications to change prediction

Eppinger & Browning (2012), provides an extensive account of Design Structure Matrix (DSM) methods applied to a range of architectures including product, organisational, process and multi-domain applications. The DSM is a network modelling tool which is particularly useful in characterising the interactions of system elements insofar the system architecture or design structure is highlighted for further analysis. The DSM is suited to development of complex, engineered systems and has seen extensive application to engineering management disciplines. However, DSM can also be applied to engineering changes or modifications to a systems architecture and is a useful method highlighting subsystems impacts or interactions. These interactions can be extended into the risk analysis domain to provide a visual indication of modification risks. Its main advantage is in its structured approach where no systems interactions can be overlooked or omitted. The DSM relies on a structured N×N matrix, mapping the interactions of the set of N system elements, where each row or column equate to a functional decomposition of the system.

Compared with other network modelling methods, the primary advantage of DSM is the graphical representation of the matrix format. The matrix provides a compact, scalable, and easily interpreted representation of a system architecture and associated change severity risks as described here.

The DSM is particularly useful in categorising two relationships important in system interaction modelling. These two relationships comprise hierarchical (vertical) and lateral (horizontal) decompositions of the system under consideration. The hierarchical relationships are derived from the decomposition of a system into elements, where this decomposition in large or complex systems, may recur through several levels. The lateral relationships for a system are derived from interactions between elements, given by energy flow, material flow, or information flow at the same level. Although DSM is mainly used to represent the lateral relationships between elements at a particular level, it can also show the locations of the elements in a hierarchy. This is shown by Figure 35 where view (c) shows the lateral relationships among elements at the lowest level of the hierarchy. In contrast view (b) shows only the presence of these relationships between higher level elements in the hierarchy. Note also that views (a) – (c) of Figure 35 are not entirely equivalent because the breakdown structure (a) view does not include the lateral relationships.

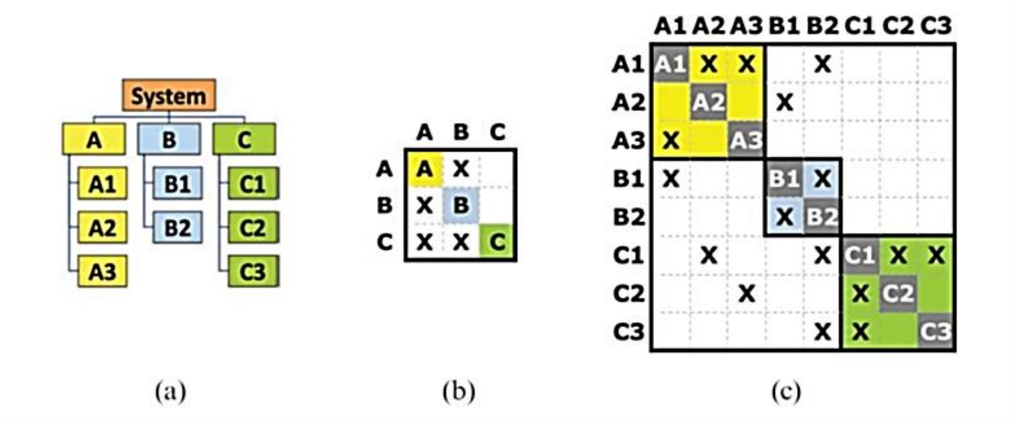


Figure 35. System decomposition through use of DSM hierarchy
Eppinger & Brown (2012)

### *4.6.7.1 Change prediction*

Clarkson et al. (2001) provides an analysis of change propagation behaviour for a helicopter modification, and the development of a mathematical model to predict risk severity of change propagation in terms of likelihood and impact of change. This model of change propagation is incorporated in the Change Prediction Method (CPM) as illustrated in Figure 36.

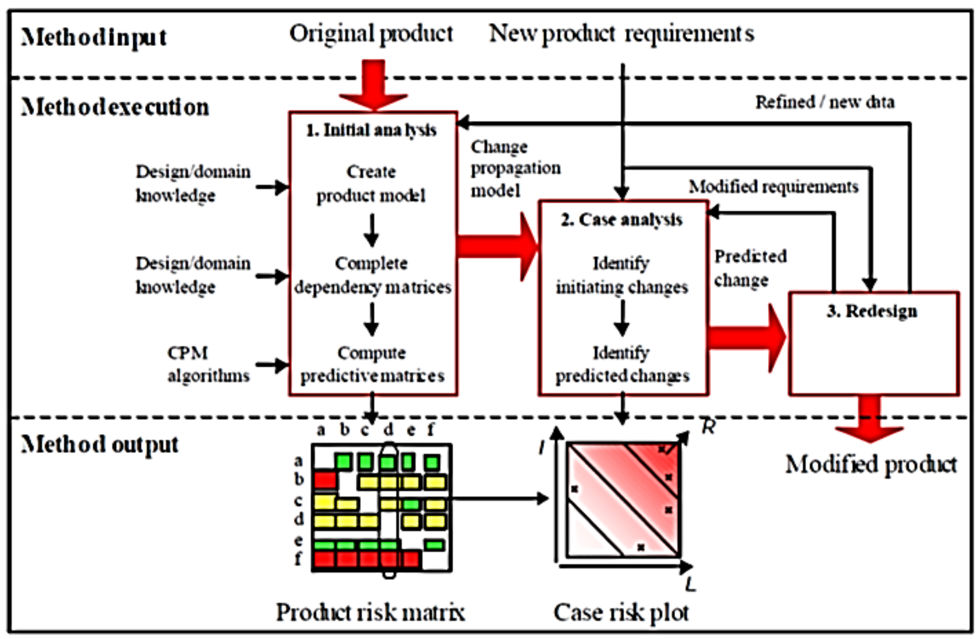


Figure 36. Change prediction method
Clarkson et al. (2001)

This Change Prediction Method (CPM) involves several steps which are summarised below.

1. Initial analysis – This step uses product data and a model of the change propagation to allow preliminary examination of the subsystem or component relationships. It consists of three sub-steps: (1) development of the product model, (2) formulation of the dependency matrices and (3) computing the predictive matrices.

2. Product risk matrix – Once the combined matrices are determined, the resultant risk data is presented in a single matrix using the technique as outlined in Figure 37. In this combined matrix, the likelihood of a change and the impact of this change on a subsystem or component is represented by the combined risk severity matrix. In this combined risk severity matrix, the change propagation between the subsystem represented by the column heading and that represented by the row is assigned the two-dimensional likelihood *l*, and the impact *i* as shown in Figure 37.

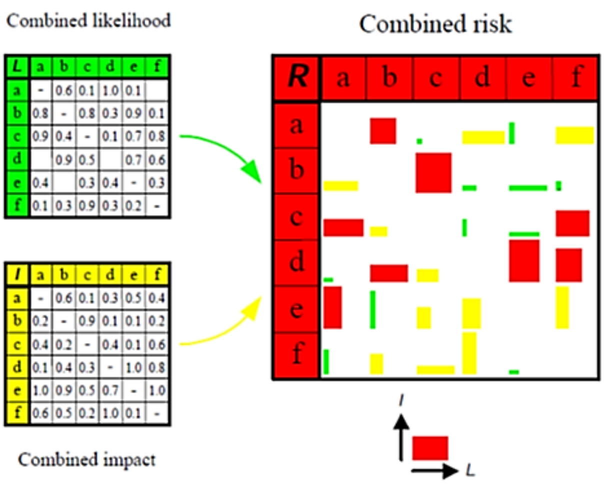


Figure 37. Graphical product risk matrix
Clarkson et al. (2001)

This combined risk severity matrix represents the combined risk of changes propagating between systems/subsystems, both directly and indirectly, with the columns impacting the rows. This combined risk severity matrix is determined by the relationship given by:

$$S_{jk} = i_{jk} \times l_{jk} \qquad \text{Equation 13}$$

Where *S* is the resultant impact severity at the *jk* element of the impact severity matrix.

$i_{jk}$ is the impact at the at the $jk$ element of the direct impact matrix.

$l_{jk}$ is the likelihood at the at the $jk$ element of the direct likelihood matrix.

3. Case analysis – The case by case analysis comprises identification of prospective changes and the presentation of predicted changes. It consists of three sub-steps: (1) Identification of initiating changes, (2) identification of the predicted changes and (3) case risk plot. This is a detailed process which investigates each change based on the new product requirement and associating this requirement with the product subsystems. The case-by-case analysis proceeds to provide $L$ and $I$ values for each instigating subsystem which are mapped to a risk scatter graph giving risk isopleths for immediate comparison of data.
4. Redesign – In this step the subsystems that require additional resources to respond to change are identified. Furthermore, this approach could be used as a measure which could be applied to an options analysis of two modification options, by evaluating the change propagation risk. This step is completed by updating the product model and dependency matrices using the initial analysis. Clarkson et al. (2001) notes that the greater the accuracy of the direct data and care used in selecting the change requirements, the better the resulting modification will be, both in terms of efficiency and functionality.

#### *4.6.7.2 Change representation and visualisation*

One challenge of incorporating design change propagation in complex systems involves the presentation of all the data or information in one display. As described in the previous sections, changes of one subsystem can propagate direct or indirect changes to other parts of the system. Keller et al., (2005), indicates that it is impossible to depict a complex system or product in one graphical representation. To display complex systems effectively, various methods are required to group and filter data so that engineers are not overcome by the complexity and extent of system information. Several means are available to visualise change propagation using a multiple view framework as described by Keller et al., (2005). These are essentially relational models, with each subsystems and associated links represented as the relationships. Each framework is summarised below:

1. Design Structure Matrices (DSM) – The DSM representation has been described in earlier sections, showing how direct linkages between systems can be displayed. However, Keller et al., (2005), states that DSMs are generally inappropriate for displaying indirect linkages. This is particularly problematic as these linkages are sometimes overlooked by designers. This often results in significant adverse impacts to project risks and budgets.

2. Change risk plots – Another way to present change impacts is to use DSM to represent change likelihood and consequence values. Keller et al., (2005), indicates that this matrix shows the combined risk of change to one subsystem given a change to another subsystem. This impact to combined risk is represented by a coloured area within each element of the DSM matrix. The colour of the area within each element represents the likelihood of change and the consequence. Therefore, using this representation of combined risk, one can assign a colour coding to draw attention to various level of risk connections.

3. Change propagation tree – The change propagation tree as described by Keller et al. (2005) is specifically designed to show the different change propagation paths. The change propagation tree is constructed by starting at the root component in the network, with all other subsystems directly connected to this component drawn as children. This is repeated for all the children subsystems until the probability of each branch falls under a user defined threshold value. The advantage of the change propagation tree visualisation is that direct and indirect linages are shown, as well as the propagation paths. However, the disadvantage is that change propagation trees can be complex to construct without dedicated computational tools.

4. Case risk scatter plot – The case risk scatter plot can be used to capture data not presented in the change network or propagation tree depictions. Case risk plots show the likelihood and impact of a resultant change given an initiating change to a subsystem. For the given instigating

subsystem(s), the likelihood *l* and impact *i* values are mapped and plotted (*i* versus *l* for each affected subsystem) on a risk scatter plot

#### *4.6.7.3 Engineering Design Domain Mapping Matrix*

The latter steps of this design methodology apply DSM techniques to evaluate engineering and certification impacts of changes resulting from the alternate fuel system or propulsion system modification. In this instance change propagation techniques as described by Clarkson et al. (2001) and others are applied to develop an engineering DMM to evaluate the impact of the alternate fuel or propulsion system modification.

The engineering DMM assesses these changes at the system requirements, subsystems and component level using as a basis the requirements and design changes documented in the QFD matrix and Change options MDM, and the results of the CPM. This step therefore brings together the earlier Change options and change propagation evaluation to support the design activities and to determine change severity risks and to facilitate mitigation of these risks during this design process. Furthermore, the process is extended into the cost domain where a modified Cost Breakdown Structure (CBS) as described by Fabrycki and Blanchard (1991) can be applied to estimate engineering costs as described in the following sections.

***Engineering management costs***

Fabrycki and Blanchard (1991) describe the cost of specific program management costs within a CBS framework, and this has been adapted here to engineering management activities by the following relationship:

$$C_{RM} = \sum_{i=1}^{N} C_{RM_i} \qquad \text{Equation 14}$$

Where $C_{RM_i}$ – Cost of specific engineering management activity *i*

$N$ – Is the number of engineering management activities

Engineering management costs cover management-oriented activity applicable across the board to design related conceptual/trade-off studies, research, equipment development and support and related data documentation for each modification element.

Such costs cover the engineering manager and administrative staff with these management functions relating to Engineering design ($C_{RE}$) and Engineering development and Support ($C_{RT}$) as described below.

***Engineering design costs***

Fabrycki and Blanchard (1991) describe the cost of specific engineering design activities by the following relationship:

$$C_{RE} = \sum_{i=1}^{N} C_{RE_i} \qquad \text{Equation 15}$$

Where $C_{RE_i}$ – Cost of specific engineering design activity *i*

$N$ – Is the number of engineering design activities

This engineering design cost includes all design activities associated with the development of the aircraft modification. Specific areas include systems engineering; design analysis; engineering drafting; reliability and maintainability studies; human factors analysis; functional analysis; logistics support analysis; installation & test instructions; training, and systems safety.

This cost also includes preparation, printing, and publication of all design data and records associated with $C_{RM}$, $C_{RE}$, and $C_{RT}$ such as reports and plans, test plans, and operational and continuing airworthiness documentation.

***Engineering development and support costs***

Fabrycki and Blanchard (1991) describe the cost of specific engineering development and test activities, and this has been adapted here to include support activities by the following relationship:

$$C_{RT} = \left[C_{RDL} + C_{RDM} + \sum_{i=1}^{N} C_{RDT_i}\right] \qquad \text{Equation 16}$$

Where

$C_{RDL}$ – Cost to fabricate prototype modification and associated assembly labour.

$C_{RDM}$ – Cost of prototype materials and bought in components.

$C_{RDT_i}$ – Cost of prototype support operations (which may include early testing) associated with specific activities *i*

$N$ – Is the number of identified tests

This cost includes the fabrication, assembly, and evaluation of the prototype modification or related subsystems in support of the engineering design activity ($C_{RE}$). Specifically, this comprises support activities the initial design phases involving fabrication and assembly, instrumentation, quality control and inspection, material procurement, logistic support, personnel, spares, support equipment, data collection and evaluation of the prototype modification.

The CBS approach to costing of engineering activities as described above can be incorporated as columns within the Engineering design Domain Mapping Matrix. Application of this costing approach is fully described in the case studies shown in Appendix 1 and Appendix 3. This initial CBS can be then refined and updated in subsequent design phases.

#### *4.6.7.4 Certification Domain Mapping Matrix*

The certification DMM follows a similar approach, with the main difference being that change impacts are evaluated against the respective airworthiness design standard whether it be aircraft, engine or other relevant technical standards or codes. The main benefits of such an approach is that all airworthiness requirements are assessed for the impact of the modification, and change severity risks are assessed accordingly with respect to costs. This is an important step which also incorporates the CBS as described above, thus enabling early estimates of certification risks resulting from changes and their associated costs. A formal evaluation of these certification risks and costs resulting from modification changes is often overlooked in the conceptual design phase. Like the engineering costs these certification costs are estimates using relationships adapted from Fabrycki and Blanchard (1991).

***Certification management costs***

The cost of specific certification management activities is adapted from Fabrycki and Blanchard (1991) using a relationship similar to engineering management:

$$C_{CM} = \sum_{i=1}^{N} C_{CM_i} \qquad \text{Equation 17}$$

Where $C_{CM_i}$ – Cost of specific certification management activity *i*

$N$ – Is the number of engineering management activities

Certification management costs cover management-oriented activity applicable across the board to certification related test, demonstration, analysis and inspection activities including management of equipment/instrumentation development/support, data processing and documentation.

Such costs cover the certification manager and support staff. These certification management functions also indirectly relate to the engineering management functions as described above. Care should be taken to ensure that the costs of the two areas are differentiated when applying these relationships.

This cost includes also preparation, printing, publication and distribution of all data/documentation associated with the certification activity, including test plans and reports, certification plans, airworthiness regulatory submissions and related documentation.

***Airworthiness compliance findings & support costs***

The cost of specific airworthiness compliance finding activities is adapted from Fabrycki and Blanchard (1991) using a relationship similar to engineering design:

$$C_{ACF} = \sum_{i=1}^{N} C_{ACF\,i} \qquad \text{Equation 18}$$

Where

$C_{ACF\,i}$ – Cost of certification compliance finding activities and associated support for specific analyses *i*

*N* – Is the number of identified airworthiness compliance finding activities.

This is the total cost of all activities associated with compliance finding including conformity inspections, compilation of engineering and reports supporting structural substantiation, flight performance and flight characteristics. Compilation and submission of Airplane Flight Manual and Approved Manual Material operating limitations and information. Analysis reports supporting fuel systems certification – fuel tank, fuel system components, fire protection and suppression. Analysis reports supporting engine certification – fuel system, exhaust, induction system, cooling, heat exchangers, fire protection, components.

***Equipment development and instrumentation support costs***

The cost of specific equipment development and instrumentation support activities is adapted from Fabrycki and Blanchard (1991) using a relationship similar to engineering development and support:

$$C_{CD} = C_{CDL} + C_{CDM} \qquad \text{Equation 19}$$

Where

$C_{CDL}$ – Cost of production prototype fabrication (used for certification) and assembly labour.

$C_{CDM}$ – Cost of production prototype material and bought in instrumentation components for certification.

This is the cost of engineering production prototype fabrication, assembly, test and evaluation carried out in support of the certification activity ($C_{CM}$). Specifically, this comprises prototype modification fabrication and assembly, procurement of instrumentation and equipment, quality control and inspections, component procurement, logistic support, personnel, spares, and calibration of instrumentation.

***Test operations costs***

The cost of specific test operations activities is considered separately as it comprises a significant contribution to the certification activity. It is defined by the following relationship using the relationships for compliance findings and support:

$$C_{CT} = \sum_{i=1}^{N} C_{CDT_i} \qquad \text{Equation 20}$$

Where

$C_{CDT_i}$ – Cost of flight-test/ground-test operations and support of specific tests $i$

$N$ – Is the number of identified flight or ground test activities.

This includes all flight test operations to certify the impact of the modification on aircraft performance and flight characteristics, engine operations, icing protection, fuel system operation. It also includes all ground test operations to certify the impact of the modification on aircraft structure such as proof loading, functional testing of the fuel system modification (tank, components, fuel lines, fire protection and suppression systems tests), engine testing (cooling systems, exhaust systems, fire protection, powerplant and fuel system controls), engine testing (calibration, vibration, detonation, endurance, operation, component tests, teardown inspections, block tests), fuel system mod leak check, cold test and pressure tests. This may also include aircraft fuelling/charging operations demonstration.

### 4.6.8 Methodology outputs

As described earlier, the outputs of this conceptual design methodology are artefacts that are used as inputs to the preliminary design phase. Although these outputs are not presented here as a step of this design methodology, the results of the QFD matrices and QMM are used to develop the Design Specification Document (DSD) and associated Requitements Allocation sheet. The results of this conceptual phase analyses, covering selected TPMs such as aircraft performance (range, payload capacity, climb rate and cruise speed etc), ground infrastructure performance, aircraft and ground system costs, emissions and sensitivity studies are also recorded in the DSD. The early design modification cost estimates addressing aircraft and associated ground systems are presented in the initial CBS. The certification DMM is used as the basis for the draft Certification Program Plan (CPP) which is a key regulatory document produced by the modification applicant and submitted to the appropriate National Airworthiness Authority (NAA). It is this document that formally initiates the design change process as a Supplementary Type Certificate (STC) covering the alternate fuel system or propulsion system modification. It is important to note that this methodology does not extend into the regulatory process past the draft compliance summary provided by the certification DMM, nor does this methodology extend into project management structures, which would traditionally accompany such conceptual design phases. However, it must be said that that outputs of the methodology could be extended to provide inputs into development of initial Work Breakdown Structures (WBS) and schedules if so required.

### 4.6.9 Representation of conceptual design methodology

This chapter has described the elements of design synthesis, evaluation of concept alternatives and analysis of outputs, and has formulated a matrix-based conceptual design methodology as shown in Figure 38. This conceptual design methodology implements the concepts of design synthesis, evaluation and analysis as an iterative process, through various steps numbered 1 through 10, building and linking together existing techniques such as Quality Function Deployment (QFD) matrices (Step 1), Gavel et al. (2006, 2007) and Ölvander et al. (2009) Quantified Morphological Matrix (QMM) (Step 2), Pugh's decision matrices (Step 3), Koh et al. (2012) Change options MDM (Step 4), and Clarkson et al. (2001) Change Propagation Matrices (CPM) (Step 5). The methodology is extended to develop Engineering and Certification Domain Mapping Matrix (DMM) techniques (Steps 6 and 7), based on Design Structure Matrices (DSM) to evaluate the impact of design modification changes on engineering and certification risks and costs. This extension incorporates a change propagation evaluation to support the design and certification activities, to determine change severity risks, and to facilitate mitigation of these risks during the next design phase.

The outputs of this conceptual design methodology are those artefacts that are used as inputs to the next preliminary design phase of the systems design process. These artefacts are shown in Figure 38 at Step 9 in the lifecycle. Those engineering artefacts include the Design Specification Document (DSD) and Requirements Allocation sheet. The DSD is the most important engineering design document, defining the system functional baseline and including needs analysis results, performance analysis, operational requirements the maintenance and support concepts, and identifying the critical TPMs. The DSD is a record of the analysis effort undertaken in establishing performance against the project TPMs, or metrics as shown as an output of the analyse results Step 8 in Figure 38.

The risk register shown on the upper right of Figure 38 is the central record of those change severity risks determined from the change propagation matrices, engineering DMM and certification DMM. This risk register will become the central part of the formal risk register for the project in later design phases, with certification-related change severity risks incorporated as part of the draft CPP.

The draft CPP is the only dedicated certification-related output of this design methodology. However, unlike other conceptual design methods, the draft CPP is key to the overall modification effort being the main artefact used by NAAs to determine the adequacy of the proposed approach to the certification of the modification. The key input to this document is the draft compliance summary against the airworthiness requirements as impacted by the proposed modification, which is determined as an output from the Certification DMM.

Finally, the Engineering and Certification DMM incorporates a means to estimate engineering or certification costs based on impacted requirements, functions, subsystems or components. These costs are collated into an initial Cost Breakdown Structure (CBS) which is based on a framework for such structures as described by Fabrycki and Blanchard (1991).

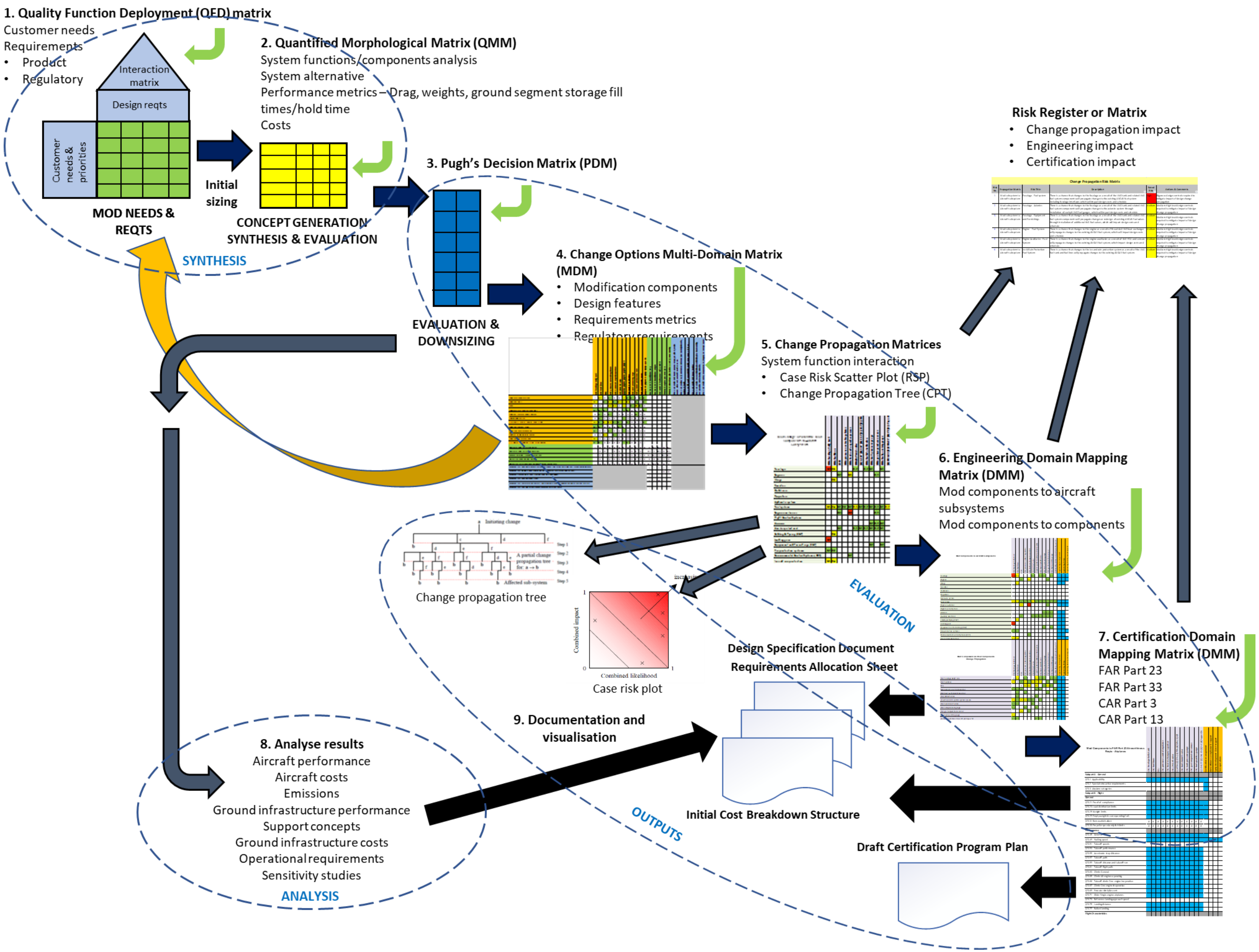


Figure 38. Conceptual design methodology for aircraft alternate propulsion system modifications

# Chapter 5. Implementation and validation

*"Too much detail too soon leads to poor design."*
*David Rendel, past Head of Mechanical Engineering Department, Royal Aircraft Establishment Farnborough*

## 5.1 OVERVIEW

The conceptual design methodology as described in Chapter 4 is based on the concepts of design synthesis, evaluation and analysis as an iterative process, building and linking together existing methods and techniques as well as extending these methods to new design domains. As previously described, it has integrated QFD matrices, QMM by Gavel et al. (2006, 2007) and Ölvander et al. (2009), Pugh's decision matrices, Change options MDM by Koh et al. (2012), and a simplified representation of the CPM by Clarkson et al. (2001). This conceptual design methodology is further extended to develop Engineering and Certification DMM techniques, based on DSM to evaluate the impact of design modification changes on engineering and certification risks and costs. This extension incorporates change propagation evaluation techniques to support the engineering design and certification activities and to determine change severity risks. It also facilitates mitigation of these risks during the future design and certification processes.

This Chapter is presented in two major sections. The first describes the implementation of this conceptual design methodology to two case studies involving a natural gas fuel modification to a commuter category aircraft, and the other relating to an electric propulsion system retrofit to a small 4-seat aircraft used for skydiving. The second section relates to the validation of this conceptual design methodology by verifying and assessing the adequacy of its application through comparative analysis which examines three areas. The first includes an assessment of the design modification space attributes as described in Chapter 3. The second compares the proposed conceptual design methodology with accepted scientific and industry process models and frameworks as presented in Chapter 4. Lastly the validation reaffirms the application of accepted scientific and industry techniques, structures and tools that make up this design methodology.

The implementation of this conceptual design methodology illustrates two aspects that comprise an original contribution to the state of knowledge in this discipline. The first involves the combination of existing and accepted techniques and tools to formulate a matrix-based framework described in this thesis. The second relates to the extension of this methodology into the engineering and certification domain in order to evaluate project costs and risks. Given that existing matrix-based techniques and tools are implemented in this methodology framework, the focus in this Chapter is the engineering and certification domain extensions and those changes or adaptions of existing techniques and tools, noting that Chapter 4, and the associated case studies shown in the Appendices, fully describe the underpinning theory and application.

### 5.1.1 Matrix-based implementation

This conceptual design methodology implemented in Appendices 1 through 3 is matrix-based, thus enabling a structured approach where the outputs of one stage forms the basis of the next. Although not explicitly shown in these Appendices, the design problem is expressed as needs and the design space is characterised as described in Chapter 3. Hence this methodology assumes that these aspects of the initial systems capability needs are fully documented using Systems Engineering techniques, and that this methodology initiates with known needs and requirements. As stated above, this conceptual design process accounts for the modification design lifecycle through (1) development and management of requirements, (2) generation of concepts, (3) concept selection validation, (4) evaluation of design changes, (5) evaluation of design impacts and certification impacts, and (6) analysis of performance estimates. Each of these matrices have outputs which are used as inputs to the next steps, or are used as feedback for preceding steps as shown in Figure 38. These feedback steps are dependent on these outputs and therefore feedback is conducted where necessary to refine the design attributes. The conceptual design documentation outputs are the draft Systems Specification, draft Certification Program Plan (CPP), draft System Evaluation Report (SER), and the project risk matrix.

### 5.1.2 Validation approach

In order to validate this conceptual design methodology, an approach relying on triangulation is implemented. This triangulation comprises two case studies as described above, and these have applied two different sets of data, data collection and analysis methods, and problem sets in order to validate this conceptual design methodology. This is further discussed in Section 5.5.

### 5.1.3 Case studies

The case studies used to triangulate this conceptual design methodology are described in Appendices 1 and 3. Both case studies have implemented the conceptual design methodology process as shown by Figure 38 using a matrix-based framework as described below. Appendix 1 describes the implementation of the Natural Gas fuels case study, with Appendix 2 providing the supporting analysis and information. Appendix 3 describes the implementation of the EP system case study.

The conceptual design methodology is referenced to the numbered process steps shown in Figure 38 with the implementation of these steps described in the following sections and also in Appendix 1 and Appendix 3.

## 5.2 DESIGN SYNTHESIS

### 5.2.1 Requirements – Step 1

Quality Function Deployment (QFD) as described by Blanchard and Fabrycki (1998) and Gudmundsson (2014) is a method intended to help in the design or modification of complex products, by taking various customer requirements into account by using a using a selection matrix. This QFD matrix helps evaluate the impact of the various customer requirements in areas such as engineering development, and achieves this by converting customer requirements into a numeric score that helps define areas for conceptual design. Ultimately, the purpose is to understand which requirements are of greater importance than others and how this complicates the development of the product, in this case, the aircraft modification.

In these case studies a set of requirements are determined from needs as a result of consultation with the customer. Typical customer needs associated with an aircraft alternate fuel modification may include flight performance, flight safety, cost, environmental impact, compatibility, and spaciousness. It is assumed in both case studies that a customer survey has requested that potential customers rate the corresponding requirements using values between 1 (not important) and 5 (very important). The rating of engineering challenges and development of the interaction of these engineering challenges are handled the way described in Appendices 1 and 3 using the definitions provided in the supporting tables. For example, for a natural gas aircraft modification, aircraft flight performance and compatibility (the ability for the modification to be fitted to other aircraft) rate highly followed closely by safety, cost and spaciousness (the ability to carry the required payload without a volume constraint). While for natural gas ground fuelling infrastructure, such requirements as 'fuel quantity storage', 'fill time' and 'costs' are the main areas that receive the greatest attention during conceptual design.

Representative QFD matrices relating to the natural gas fuel system modification are provided in Appendix 1 by Table 10 and the corresponding ground fuelling infrastructure by Table 14. While the representative QFD matrices relating to the EP system modification are provided in Appendix 3 by Table 57 and the ground charging station infrastructure by Table 59.

### 5.2.2 Initial sizing

#### *5.2.2.1 Natural gas modification sizing constraints*

A design modification to an aircraft is constrained by the existing aircraft configuration and geometry. For example, modified natural gas wing tip fuel tanks are generally sized to be the same as existing AVGAS wing tip tanks to minimise aerodynamic and structural impacts to the aircraft. Physical constraints also apply to the under-fuselage natural gas belly tank installation, where ground clearance requirements limit the physical diameter of the tank. Furthermore, the length of this natural gas belly tank is limited by nose landing gear door (forward) and the takeoff rotation ground clearance angle (aft) and the profile of the fairing which is required to reduce aerodynamic drag. These physical constraints limit the sizes and volumes of the fuel tanks as determined in this conceptual design case study. These constraints therefore serve as the basis for initial sizing of the modification, with optimisation of the fuel tank volumes and geometries taking place in the later design phases.

#### *5.2.2.2 EP system performance sizing*

The removal and replacement (retrofit) of an existing aircraft propulsion system along with the requirement to develop the replacement to cater for a specific mission, necessitates a more detailed initial sizing approach. In the case of an EP system modification, this involved the development of an aircraft climb performance model to support the conceptual sizing and quantisation of the morphological matrix supporting the skydiving mission. A two-step process was followed where the outputs of this model were compared with Cessna Aircraft Company – Cessna 182P Pilots Operating Handbook (1976) data at the same climb conditions in order to verify the model. The second step used the same model, modified to exclude altitude density power effects (i.e. accounting for constant power electric motor performance) providing estimates of climb energy. These estimates of climb energy were based on the aircraft climb rate, time of climb performance and engine power. In addition, this climb performance model also estimated reserve energy requirements and accounted for recuperative propeller power regeneration where applicable. These estimates of total energy were used to undertake initial sizing of batteries to achieve the mission. This relatively detailed climb performance model incorporating details of EP system component weights, battery performance, propeller performance and drag impacts, was used as a basis for initial sizing, as well as inputs into the quantised morphological matrix described later.

In summary, both case studies have implemented different approaches to this initial sizing step, with these outputs serving as inputs to morphological matrix quantisation described later. They also serve to triangulate the methodology by demonstrating two different initial sizing approaches.

#### *5.2.2.3 Incompatible concepts*

As described in Appendix 1 the conceptual design methodology eliminated certain configurations on the basis of incompatibilities such as systems safety impacts or technology limitations. In the case of the natural gas fuels modification any configuration that included in-fuselage LNG tank storage location was discounted on account of flight safety. This eliminated those risks and hazards associated with natural gas fuel leakage from fuel tanks or fuel lines within the fuselage which could cause a fire risk to occupants.

Using a similar rationale, the morphological matrix associated with EP systems described in Appendix 3, included Electric concepts only. Although Hybrid Electric configurations comprised possible solutions, the case study analysed EP system concepts only. This was a client design choice which limited the number of configurations as outlined in the morphological matrix.

### 5.2.3 Morphological matrix – Step 2

The structured generation of concepts within the design modification space is a challenge well suited to the morphological methods. As in Chapter 4, morphology is an approach introduced by the astrophysicist Fritz Zwicky as reported by Ritchey (1998). One of the ideas of morphology is to search for a solution in a systematic way by trying out all possible combinations in a matrix.

Both aircraft and ground infrastructure segment morphological matrices are incorporated in one table to show that these two segments are related to the overall systems capability. For the natural gas modification case study, this combined morphological matrix is shown by Table 17. On this basis a segment concept can be generated by selecting one solution for each functional subsystem.

The aircraft segment morphological matrix includes the main functional descriptions which are used to define the segment concept as well as functional descriptions relating to fuel system components. It is important to note that only the main functional descriptions are used to generate a segment concept. Other combinations of subsystem components such as valves, regulators, relief valves, circuit breakers and cockpit instruments/controls are not presented in the morphological matrix, which is consistent with the approach outlined by Pahl et al. (2007).

#### *5.2.3.1 Quantisation of the morphological matrix*

The morphological matrix is one approach of many in the literature that addresses the generation of a large number of possible concepts. This approach extends the framework used by Weiss and Gilboa (2004) into an approach that quantifies the morphological matrix using a methodology which is based on that described by Gavel et al. (2006, 2007), Gavel et al. (2008), Ölvander et al. (2009), and Svahn, (2006). The quantified matrix is a conventional morphological matrix that incorporates mathematical models of the solution elements. In the framework as presented, the focus is on solution elements that can be quantified, such as modification weight, aerodynamic drag, flight performance and cost.

The approach as presented is augmented by quantisation of the matrix using properties of the key technical performance measures (or metrics) where the best solution or smaller set of solutions can be ranked for further analysis or evaluation. In the case studies, concept selection was based on quantisation of the morphological matrix using key metrics as determined in the previous QFD step. This structured approach explored the complete design space, and down-selected the ‘best’ potential solutions using simple relationships based on minimisation and maximising these normalised metrics as a Figure of Merit (*FoM*).

The quantified matrix gives immediate access to approximated solution elements for the complete system, with every potential sub-solution described either by physical or statistical equations, or a combination of these modelling approaches. Thus, useful metrics are quantified accordingly for each solution alternative. This quantification is applied in the case studies presented in Appendix 1 and Appendix 3. A quantified value of the complete modified system can be obtained by combining together the properties for each of the chosen sub-solutions. The mathematical basis underpinning this methodology is described in Chapter 4, where the approach involves identifying the modification solution with the lowest weight ($W$), installation aerodynamic drag ($D$), and cost ($C$), whilst providing the highest possible fuel loading ($F$) or Useful loading solutions.

The supporting analysis for quantisation of the natural gas fuel system modification is provided in Appendix 2. This Appendix provides the underpinning data and information used in Appendix 1, particularly in the quantisation of the aircraft morphological matrix. In this case, metrics are derived for natural gas fuel tank weights and also drag estimates. Fuel tank weight estimates are determined from a literature review of existing CNG and LNG fuel tanks used in the automotive and heavy vehicle transportation industries. In addition, some limited weight data were available on large scale lightweight LNG fuel tank design concepts as proposed by NASA (Carson et al. 1980).

Appendix 2 also provides the underpinning analysis which estimates the drag increment resulting from new natural gas fuel tank installations. These drag estimates were based on data and methods presented in Hoerner (1965) which were derived from wind tunnel test or flight test data. Two sets of drag estimates relating to (1) the wing tip tank installation, and (2) the fuselage belly tank installation, were derived, with these results used to determine the drag metrics shown in the natural gas fuel system modification quantisation matrix.

Representative quantified morphological matrices relating to the natural gas fuel system modification are provided Appendix 1 by Table 18 and Table 19, and the ground fuelling station infrastructure by Table 22. While the representative quantified morphological matrices relating to the EP system modification is provided in Appendix 3 by Table 78 through Table 81, and the ground charging station infrastructure by Table 82.

## 5.3 EVALUATION OF ALTERNATIVES AND CHANGE OPTIONS

### 5.3.1 Pugh's decision matrix – Step 3

The next step in this methodology involves the application of the Pugh Matrix (PM) which is used to validate the candidate concept solution as determined from the previous morphological matrix analysis. As stated by Burge (2009) the PM takes into account multiple factors using a relatively simple approach when reaching a decision. The application of the PM pairwise comparison method provides for a more objective decision when dealing with subjective opinions. It can also accommodate a simple sensitivity analysis, thereby providing some information on the robustness of a the decision. It is an ideal method to employ within a design engineering team, where the independency of the team can be used review and validate the output from the previous morphological analysis. Independent validation of the concept is very important as the next step in this methodology commits further resources and effort in terms of change propagation, engineering and certification analysis.

In Appendix 1, Table 23 shows an example of a PM associated with the six favoured natural gas fuel modification candidate design concepts as determined from the previous morphological analysis step. However, Appendix 3 did not generate a PM as only one compliant candidate design concept was identified. In this case the design methodology progressed to the next step. However, this case study did highlight the sensitivity of the quantified morphological matrix to the derived Technical Performance Measures (metrics) as described below.

#### *5.3.1.1 EP system point of validation*

This case study previously investigated the effect of selecting motor maximum/peak power as the basis of climb performance prediction (Williams, 2018b) in an earlier version of this spreadsheet analysis. The selection of the motor maximum/peak power parameter, in contrast to continuous power, changed the preferred candidate selection. In the previous case the preferred solution was the Emrax 268 motor-based solution, as this motor possessed high specific peak power (i.e. possessed very low weight) in comparison to the other motors. This was closely followed by the Siemens AG SP260D motor and the Emrax 348 motor. However, the preferred candidate was the Siemens AG SP260D motor, which was based on the continuous power metric. This result aligns with motor performance data where the Siemens AG SP260D motor possessed the highest continuous power at high power density. This motor was also flight proven (Siemens AG, 2016). This outcome therefore provides a point of validation for this QMM approach.

### 5.3.2 Change Option Multiple-Domain Matrix – Step 4

The assessment of engineering change propagation effects can be conducted at any stage of systems design lifecycle. However, it has been applied as an intermediate step of this conceptual design methodology. As discussed in Chapter 4 of this thesis, the conduct of this step could occur before the evaluation of the design change impacts. However, it is conducted at this juncture to provide inputs to the engineering assessment step. In reality, these propagation effect design synthesis activities would be conducted as a parallel iterative process.

This assessment of propagation effects and change options are described in the paper by Koh et al. (2012). The underpinning theory associated with this method is described in Chapter 4. Based on this approach, it can be seen that the method needs to deal with information between change options, product requirements and product components (referred to as description domains). Hence, Koh et al. (2012) has applied this method using a Multiple-Domain Matrix (MDM) as shown in schematic form by Figure 33. Basically, this MDM-based method is a combination of Design Structure Matrices (DSMs) and Domain Mapping Matrices (DMMs) as described in Chapter 4.

The four steps of this method described by Koh et al. (2012) broadly correspond to the dependences modelled in each Field labelled as A, B, C, D and E within the Multiple-Domain Matrix (MDM) as shown in Table 24 and Table 83. These fields are used to develop the modification change dependencies MDM as shown in Appendices 1 and 3 for each case study and therefore draw on the earlier steps developed within the earlier QFD matrix.

### 5.3.3 Change Propagation Matrices – Step 5

This analysis of change propagation related to complex systems is based on the work undertaken by various researchers Clarkson et al. (2001), Eckert et al. (2001), Eckert et al. (2009), Eckert et al. (2006), Keller et al. (2005), Koh & Clarkson (2009) and Koh et al. (2012). In this context the changes resulting from aircraft modifications impact not only the interfaces but also to other aircraft systems, subsystems and components. As described by Keller et al. (2005), design complexity has many parts which can be divided into (1) complexity of the design process, (2) organisational complexity, (3) complexity of design description and (4) complexity of the product. This step of the methodology deals with complexity of the product, whereas the broader conceptual design methodology described in this thesis deals with the design process and complexity of the design description.

This change propagation analysis focuses on the methods used to model the dependencies between system, subsystems and components of a particular propulsion system modification solution. This approach can be used to support the risk/impact assessment of the solution which fits into the broader change management process within a design organisation.

Clarkson et al. (2001), outlines a method that predicts change propagation in complex design. The method outlined is referred to as the Change Propagation Method (CPM), which is fully described in Chapter 4. This step illustrates the practical application of this method which has been simplified and adapted to fit within a broader matrix-based conceptual design methodology. It is based on a combination of Design Structure Matrices (DSM) as described by Clarkson et al. (2001), and Multiple-Domain Matrices (MDM) and Domain Mapping Matrices (DMM), as described by Koh et al. (2012). These set of matrices make up the change propagation analysis element of this methodology and examines the impact of change using direct dependency relationships which are then used to derive risks for further consideration by the design team. It is important to note that this CPM as described by Clarkson et al. (2001) has been adapted to consider direct dependencies within the framework of the conceptual design process. As a conceptual design methodology, it is intended that the full application of the CPM shall be applied in the later design phases in order to facilitate the full investigation of change propagation effects on an established configuration baseline.

The first step in this change propagation analysis follows the general process outlined by Clarkson et al. (2001) which is based on a DSM combined with a propagation model. This change propagation model is derived from the product itself in this case being the natural fuel system or the EP system as illustrated in Figure 43 of Appendix 1, and Figure 68 of Appendix 3, respectively. These aircraft systems can be decomposed into subsystems and components and their related dependencies. Furthermore, the existing aircraft fuel system can be decomposed into subsystems and components. In this case, the existing AVGAS fuel system will possess dependencies, although the design requirement for the bi-fuel arrangement will ensure that these AVGAS subsystems and components will not be modified. This design objective will maintain the existing certificated configuration of these fuel subsystems and components, thus negating the requirement for additional recertification activities. Nevertheless, this design objective is testing in certain corner-conditions where bi-fuel fuel valves are required, and changes or modifications are required to engine fuel injection and engine control units.

#### *5.3.3.1 Modification risk elements*

The CPM as outlined by Clarkson et al. (2001), relies on systems decomposition and dependencies as described above, with the dependencies obtained from an analysis of the aircraft subsystems architecture. These are defined in terms of the likelihood that the redesign of the subsystem or component will force the redesign of another and the subsequent impact, or extent, of that redesign. The degree that this interaction occurs is determined by the linking parameters types and their attributes as considered in both case studies shown in Appendices 1 and 3.

As described by Clarkson et al. (2001) the second element of this process is the assessment of the design change and the subsequent impact, or extent, of that redesign. As stated by Clarkson et al. (2001), both the likelihood and impact level are assigned a numerical value between 0 and 1 with this referring to the total change experienced during the redesign process. This aspect is fully described in Chapter 4 and has been implemented in both case studies described in Appendices 1 and 3.

These two relationships are combined to represent a scale of impact severity, where this is defined as the product of the likelihood ($l$) of the change and the consequence/impact ($i$) or cost of the subsequent change. This terminology is borrowed from Department of Defense (2012) systems safety management and simplified in this implementation to facilitate a three-level category which can be colour coded for change severity risk.

The final outcome of this process is to represent impact severity ($s$) of the change in one subsystem or component resulting in a change in an adjacent subsystem or component by propagation over a common interface. The subsequent risk of propagation of these changes to other subsystems or components can then be predicted.

#### *5.3.3.2 Modification dependencies*

The CPM as outlined by Clarkson et al. (2001), has been implemented in Appendices 1 and 3 is based on a common DSM framework which analyses change propagation dependencies as follows:

1. Modification subsystem to aircraft subsystems – Evaluates the impact of design changes to modification *subsystems* on the standardised aircraft *subsystem* architecture.
2. Modification component to aircraft subsystems – Evaluates the impact of design changes to modification *components* on a standardised aircraft *subsystem* architecture.
3. Modification components to modification components – Evaluates the impact of design changes to modification *components* to other modification *components*.

All three change propagation DSM frameworks follow a similar analysis process all of which rely on generation of likelihood and impact dependency matrices based on the respective modification subsystem, aircraft subsystem and modification components.

Note that these change dependency matrices are created by direct subsystem dependencies. This analysis differs from the CPM outlined by Clarkson et al. (2001) where this analysis combined direct and indirect dependencies to develop a change propagation network. The CPM approach allows for a more detailed analysis of change propagation paths via the generation of a change propagation tree. A full network analysis was not undertaken within this conceptual design process, although an example change propagation tree was generated as part of the natural gas fuel modification case study. Rather the approach has been to generate change propagation risks from each respective change severity matrix. This approach ranked change severity identified in each matrix and structured these using a traditional risk parsing format for insertion into a standard project risk matrix.

An extension of the change propagation related DSM work as described by Clarkson et al. (2001) included the provision of an additional column and row providing totals of change severity. The additional column shown on the right-hand side of the change severity matrix shows the change severity total of each aircraft subsystem impacted by the modification subsystems. These totals highlight the degree that aircraft subsystems are affected by the modification. In a similar way, the additional row shown on the lower part of the change severity matrix shows the change severity total of each modification related aircraft subsystem impacted by each aircraft subsystems. These totals highlight modification subsystems that are highly affected by the aircraft subsystems.

The same principles are applied to the remaining change propagation DSM frameworks, where direct likelihood and direct impact matrices are used to generate the resulting change severity risk matrix in both case studies provided in Appendices 1 and 3. Appendix 1 provides additional detail that assists in visualising the change impacts through provision of a partial change propagation tree (as discussed earlier) and a case risk scatter plot. However, given that this conceptual design provides output for latter design phases the approach here has been to record change severity risks in a traditional risk matrix. This risk matrix is used to support conceptual design and inform project management activities and also assist in estimation of development costs.

### 5.3.4 Engineering Domain Mapping Matrices – Step 6

In this step, DMM methods are developed within an engineering domain to evaluate the impact of changes to subsystems and hence impact of risks and costs resulting from the case study modifications. This approach is an extension to existing DSM and DMM methods and uses the results of the change propagation analysis described previously. This DMM extension is also used to evaluate the impact of ground infrastructure segments in the same way.

This distinction between engineering development and certification, as dealt with in the next section, is made here to ensure that the methodology accounts for the discrete activity that involves conceptual design engineering activities and the associated risk mitigation and cost estimates. These engineering activities, like other steps in the conceptual phase are iterative in nature, and the use of engineering DMMs would be undertaken using change propagation analysis techniques as described in the previous step. One output of this DMM-based evaluation is the draft Systems Specification Document detailing the aircraft modification and the ground station infrastructure. The draft system specification documents are based on the format and structure as detailed in the respective DMMs. The other output of this evaluation is a Cost Breakdown Structure (CBS) which estimates engineering costs associated with the aircraft modification and the ground station infrastructure as well as the Requirements Allocation Sheet which documents the connection between allocated function, performance and the physical system.

#### *5.3.4.1 Aircraft design change impacts*

The engineering DMMs corresponding to the aircraft modification case studies are shown in Appendices 1 and 3 respectively. Each aircraft DMM is split into two parts to facilitate presentation. For example, the aircraft natural gas fuel system modification DMM is shown in Appendix 1 by Table 39 and Table 40.

Part 1 of this DMM shown by Table 39 corresponds to the natural gas aircraft modification based on design change propagation data determined from the previous change propagation analysis step. Specifically, this DMM data is shown transposed from the Appendix 1, Table 36 and Table 37 modification change severity/risk results. The colour coding provides a measure of engineering development risk impacting engineering cost. The DMM data is also transposed to provide natural gas modification components as common columns to which cost data estimates can be aligned either from an aircraft system, or as a modification component. The right-hand side of the DMM shown in Table 39 shows engineering costs resulting from natural gas fuel system modification impacts on aircraft subsystems. These costs are broken down into (1) engineering management (2) engineering design, and (3) engineering development and support, using a cost structure as described in Chapter 4. The determination of these engineering costs is not within scope of this thesis. However, it is sufficient in this conceptual design phase to identify those costs impacted by the natural gas fuel system modification by those blue highlighted elements in Table 39.

Part 2 of the engineering DMM shown in Table 40 of Appendix 1 describes the relationship of the natural gas modification components to requirements and design change parameters. This analysis evaluates the impact of the natural gas modification components on requirements and design changes parameters to facilitate a functional view and physical view of the modification. This aspect of the DMM provides a 'reverse' view of the functional analysis step undertaken in Systems Engineering and assigns engineering resources to these activities at a functional level. One output from this process is the provision of a draft Requirements Allocation Sheet which can be used in later design phases of the project. Again, it is not intended to produce this Requirements Allocation Sheet as an output, as this is a standard Systems Engineering activity which is out of scope in this thesis. As stated above, costs are identified by the blue highlighted elements in Table 40. Again, the determination of these costs is not within scope of this thesis.

This engineering DMM, along with the corresponding EP system engineering DMM shown in Appendix 3, provides a framework from which engineering design activities can be evaluated. The basis of this evaluation is risk determined from a change propagation analysis conducted in the previous step which can be then used to estimate the costs. Furthermore, the magnitude of change severity risk can be used as a basis for design controls to be applied to mitigate the impact of these risks.

#### *5.3.4.2 Ground station infrastructure design change impacts*

In the same way, the engineering DMMs corresponding to each ground station infrastructure case studies are shown in Appendices 1 and 3 respectively. For example, the natural gas ground station DMM is shown in Table 41 in Appendix 1.

Again, the same approach is applied to the ground station infrastructure, noting that the main purpose of this DMM is to present engineering costs in a common framework. In this case it is assumed that ground station infrastructure is a non-developmental segment. That is, this ground station segment does not require engineering development and support (i.e. no prototypes or developmental testing) activities with the associated costs. Therefore, these costs are those associated with engineering design activities concerned with the definition and allocation of functional and performance attributes of the ground station infrastructure.

### 5.3.5 Certification Domain Mapping Matrices – Step 7

The impact of alternate fuel system modifications involving new technologies on the aircraft Type Certification Basis (TCB) is often overlooked in early conceptual design studies. The airworthiness design standards may not always cater for these new technologies therefore necessitating design changes to the modification or the formulation of new approaches to demonstrating compliance. Therefore, design changes made necessary by certification may also be accompanied by additional propagation impacts as discussed earlier. This challenge is further compounded insofar that these changes may also be the result of indirect change propagation, which is difficult to predict without a structured and rigorous approach covering all airworthiness design requirements and change propagation models.

In this context a new conceptual framework is developed and implemented via a certification DMM matrix in a similar way to that described for the engineering DMM. Although the focus is airworthiness certification, the technique can also be extended to any system needing to show compliance to a design standard. For example, the DMM technique developed here can be applied to the ground station infrastructure where applicable standards do exist for natural gas production, storage, and handling of LNG. Or alternatively in the case of new technologies, the DMM technique can be adapted to include contemporary consensus standards such as that found in the new CFR 14 FAR Part 23 Amendment 66 airworthiness standards for small aircraft.

The importance of a structured and defensible approach to certification of new alternate fuels and propulsion systems modifications cannot be understated. The presentation by Serra (2018) states that airworthiness certification has many traps and pitfalls which introduce risks and result in increasing costs to an electric aircraft program even after the technology has been successfully demonstrated. In addition, the online article by Thomson (2018) highlights the criticality of engaging early with airworthiness authorities in relation to aircraft electric propulsion technologies. The conclusion was that from a certification standpoint both approaches are important and necessary, and both incorporate varying levels of certification effort. Thomson (2018) states that "retrofitting" via a modification to an existing airframe, simplifies the certification task and makes concrete progress towards certifying a product for operational flight. This would then provide valuable knowledge and experience, including to the airworthiness authorities in support of more disruptive approaches.

It is noted that the impact of changes to an aircraft resulting from a modification is often assessed using a draft compliance summary. For small aircraft this is a framework based on FAR Part 23 design standard requirements. In this thesis a multi-part mathematical process is introduced which derives a number of matrices that equate to the relevant airworthiness design standards. e.g. FAR Part 23 for small aircraft or CAR Part 3 (historical standard for small aircraft); FAR Part 33 for engines or CAR Part 13 (historical standard for engines). The objective is to derive DMMs for each standard and to determine a certification impact matrix which can be used for the next step in this evaluation methodology.

Appendices 1 and 3 provide two case studies which highlight modifications incorporating a natural gas fuel system and EP system into small aircraft. This method relies on using a DMM structure to provide rigor and coverage to the impact of change propagation on aircraft certification requirements and also provides a framework from which to estimate the associated costs noting the change severity risk elements.

In order to evaluate the impact of these modification design changes on certification, it was necessary to develop a DMM structure that reflects the aircraft and engine TCB. This DMM structure needs to also reflect new certification requirements resulting from new technologies or architectures introduced as result of the modification. These are the first steps in defining the proposed TCB or STC to be approved by the NAA.

#### *5.3.5.1 Airworthiness design standards and related codes*

Airworthiness design standards specify regulatory requirements which must be achieved to provide an airworthy and safe type design, with advisory circulars and other guidance documentation providing information in relation to the acceptable means to demonstrate compliance against these design standards. These airworthiness design standards address certification requirements for an aeronautical product and cover the aircraft, engine and propeller. Type certification of the aircraft, engine and propeller implies that aircraft, engine or propeller is manufactured according to the approved design can be issued an Airworthiness Certificate.

#### *5.3.5.2 Aircraft and engine certification standards*

In these case studies the impact of a modification to the aircraft fuel system by addition of a fuel modification will impact the aircraft and engine. These impacts are therefore addressed by airworthiness design standards as described above, with the most common standards applied being FAR Part 23 for aircraft and FAR Part 33 for engines as detailed in Chapter 3, or the older CAR Part 3 for aircraft, and CAR Part 13 for engines, as applicable.

#### *5.3.5.3 Natural gas fuel system components and parts technical standards*

Aviation has a system of Technical Standard Orders (TSOs) which are essentially minimum performance standards for specified materials, parts, and appliances used on civil aircraft. It is understood that no TSOs exist that authorise the use of natural gas fuel related components and equipment for aeronautical applications. One approach may be to adopt the National Fire Protection Association (NFPA) 52 standard requirements which are applicable to vehicular gaseous systems. Indeed, the DMM framework as described in this thesis could be used to assess potential changes required to natural gas fuel system modification components to comply with the aeronautical environment. The outcomes of this approach would also be incorporated into the CPP described earlier, thus forming a component of the certification basis for these natural gas fuel components. The apparent absence of aviation TSOs applicable to natural gas fuel system components introduces additional certification challenges which need to be addressed as part of the broader certification effort. Although the certification DMM can be used to evaluate TSO compliance it will not be dealt with here in this thesis for reasons of brevity.

#### *5.3.5.4 Aircraft alternate fuels certification*

The certification of aviation fuels is a complex area with Ziulkowski (2011) providing a summary on certification aspects of aviation gasoline (AVGAS). The specification for AVGAS is administered by the American Society Testing Materials (ASTM) D910-11 (2011) defining specific types of aviation gasolines for civil use. This specification is provided to ensure that AVGAS provided worldwide is a “good” fuel for all users including the producers, engine manufacturers, airframe manufacturers, component manufacturers, the FAA, and the users of the fuel. Therefore, a change in fuel will have an impact to aircraft performance, fuel consumption, operating instructions, and instrument markings when compared to the original certification basis of the aircraft and engine. Furthermore, a change in AVGAS specification (e.g. removal of TEL) will have an adverse effect also on the certification of parts as a Parts Manufacturer Authorisation (PMA) or a Technical certification Order (TSO) as described above. Therefore, it is apparent that a change of fuel will invalidate previous certification, and with it comes considerable cost, and possible adverse impacts to airworthiness and safety.

In addition, the certification of natural gas fuels is not well defined in the vehicular ground transportation domain as described by Lebrato J. et al. (2013), as it is noted that there are areas within LNG fuel regulations, codes and standards that are yet to be harmonised between countries. Lebrato J. et al. (2013) discusses these areas of non-harmonisation and provides an outline of current plans undertaken to resolve these matters.

On this basis a DMM certification analysis of alternate aviation fuels is not presented here due to its inherent complexity as described by Gillette (2017) and Macnair et al. (2017). Furthermore, the harmonisation issues described by Lebrato J. et al. (2013) do not provide a unified certification basis from which to develop a standard for natural gas aviation fuels at this time. Given the complexity and expense associated with the FAA Piston Aircraft Fuels Initiative (PAFI) program, and issues associated with natural gas aviation fuels certification, future certification of natural gas aviation fuels would present considerable but not insurmountable challenges.

#### *5.3.5.5 DMM structure - aircraft*

Appendices 1 and 3 describe case studies associated with certification DMM implementation on a natural gas fuel system modification and an EP systems modification. Both case studies follow the same certification DMM structure although the modification scope is different in each case study. Rather than describe the details of each case study certification DMM implementation, the natural gas case study shown in Appendix 1 is provided as an example, noting that EP system implementation is similar.

**Certification impact process** - Table 42 and Table 43 of Appendix 1 provides excerpts of the CAR Part 3 and FAR Part 23 aircraft certification DMM which is based on propagation analysis methods using change likelihood and impact matrices. This DMM therefore presents the change severity/risk matrix for the LNG fuel system modification impact on the certification basis as shown. For each applicable LNG fuel modification component, a likelihood and impact assessment is undertaken through reference to the appropriate requirement as defined in CAR Part 3 or FAR Part 23. The process follows the same change propagation analysis as described earlier in this section and is therefore best undertaken by the design team. In addition, Table 43 of Appendix 1 provides an example of this process using a CAR Part 3 requirement relating to sub-paragraph § 3.551 - *Fuel valves*. In this case the proposed automotive LNG valve solution is assessed as non-compliant with the sub-paragraph § 3.551 - *Fuel valves* requirement. In order to comply, it is assumed that redesign of the LNG valve is required. This in turn invokes a change propagation analysis, if not already completed, for this component. This process is shown conceptually in Figure 39, where the changes invoked by certification are assessed within the change propagation matrices. The change severity risks are then determined from this step, and introduced accordingly into the Engineering DMM, as the next step in order to inform risks and to facilitate the engineering redesign assessment activity. Depending on the outcome criticality, a partial change propagation tree may be generated to assist in the understanding change propagation and its effect on engineering redesign and certification activities.

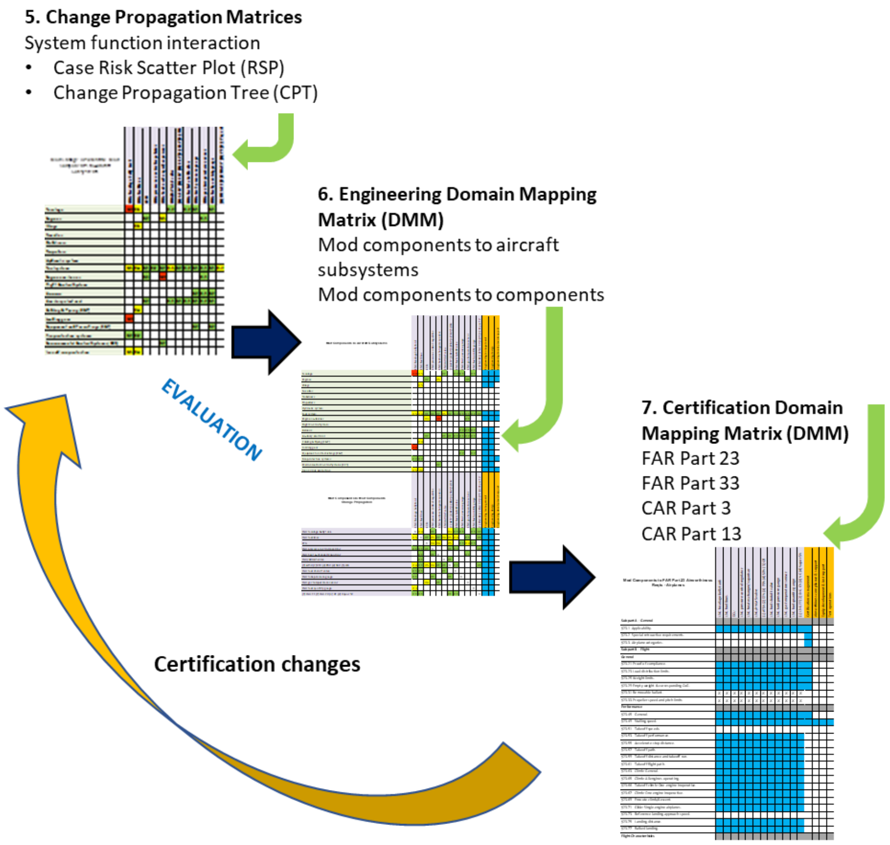


Figure 39. Example of changes invoked by certification

This example illustrates the iterative characteristic of this design methodology, and also highlights that this process can be applied to all other remaining airworthiness requirements. That is, this certification impact process can be applied to certification requirements such as flight manual operating limitations information (i.e. AVGAS/LNG fuel management arrangements), cockpit controls (i.e. location of fuel valve), motion and effect of cockpit controls (i.e. operation of the fuel valve), and control knob shape (i.e. what size and shape of fuel valve).

The application of the certification DMM also acts as a framework from which to judge the airworthiness of a proposed design concept. The excerpt of requirements relating to the fuel system as shown in Table 43 also highlights potential omissions. In this case the architecture of the LNG fuel system is based on a typical heavy vehicle fuel system which does not incorporate a fuel strainer or a fuel drain. However, the CAR Part 3 excerpt shown in Table 43 specifies requirements for §3.552 fuel strainer and §3.553 fuel system drains. This potential non-compliance needs to be addressed with the resulting analysis recorded in the Certification Program Plan (CPP) for NAA consideration, as discussed in Chapter 4. Furthermore, this potential non-compliance may also be incorporated as a requirement in the Design Specification Document (DSD), which would be updated or refined in the next design phase.

It can therefore be seen that incorporation of the certification DMM analysis step is an essential component of this design methodology. For without it, important design considerations may be overlooked, and discovered only in the later phases of the design.

**Certification costs** - The right-hand side of the DMM shown by Table 42 and Table 43 presents the certification costs resulting from LNG fuel system modification as described above. These costs are broken down into (1) certification management (2) airworthiness compliance findings and support, (3) equipment development and instrumentation support, and (4) test operations, again using a similar cost breakdown structure as described in Chapter 4. The determination of these certification costs is not within scope of this thesis. However, the approach here is to use the results of the certification severity/risk matrix to inform the certification cost categories including allowances for severity/risk. For example, green shaded costs are associated with low severity/risk impacts and red shaded areas are associated with high severity/risk impacts. Therefore, these levels of certification severity/risk impact can inform detailed cost estimates in the next phase of the modification design lifecycle.

Furthermore, whilst not shown, the red shaded areas or elements can be further analysed using the change propagation method as described earlier.

#### *5.3.5.6 DMM structure – engines*

As stated above, the same DMM structure can be applied to the certification of engine design changes resulting from the natural gas modification. Table 37 of Appendix 1 provides an excerpt of the CAR Part 13 aircraft engine certification DMM which is based on design change propagation data using change likelihood and impact methods as described earlier. This DMM presents the change severity/risk matrix for the LNG fuel system modification impacts on the engine certification basis. For each applicable LNG fuel modification component, a likelihood and impact assessment of is undertaken through reference to the particular requirement as defined in CAR Part 13.

#### *5.3.5.7 Outputs*

In addition to cost estimates, the method can provide as an output, a draft compliance summary (which comprises the certification DMM) as an important input to the draft CPP described in the following section. This CPP is not presented here in this thesis for reasons of brevity.

## 5.4 ANALYSIS AND OUTPUTS

As shown in Figure 38 the final activities in this conceptual design methodology are concerned with the analysis of performance attributes, preparation of specifications and plans, and an assessment of uncertainties and sensitivities associated with the concept. It should be noted however that all steps in this conceptual design process may feedback into earlier steps as an iterative process, with the main objective of this methodology is to finalise the preferred design concept. This is achieved through establishing the specification and associated plans for the next design phase.

### 5.4.1 Performance metrics and analysis – Step 8

#### *5.4.1.1 Aircraft performance*

Aircraft performance metrics associated with the natural gas and EP system modification case studies are related to the specific mission and role of each aircraft. In the case of the commuter aircraft mission, the natural gas fuel modification range performance was rated as important, via the QFD matrix requirement relating to “low drag and low weight”. In this instance a measure was adopted that encapsulated this performance requirement, which was expressed in a payload-range diagram. The payload-range diagram is a standard means of presenting this performance information and provides a means of comparing payload-range of various aircraft. This payload-range attribute incorporates the weight impact of the modification and fuel load defining payload, and also reflects range performance as a function of the specific fuel consumption and lift to drag ratio. This payload-range diagram was developed in Appendix 2, and presented in Appendix 1 for the natural gas fuel modification case study. Cruise performance was also estimated as a result of the drag impact of the external natural gas fuel tank mounted beneath the fuselage. The estimate of drag and hence impact to aircraft cruise airspeed was developed in Appendix 2, and presented in Appendix 1. It is important to note that the metrics presented in the quantified morphological matrix are reflected either directly or indirectly in the range-payload and cruise speed performance attributes.

In the case of the skydiving aircraft mission, the EP system modification climb speed and useful load performance was rated important via the two QFD matrix requirements relating to “time of climb to altitude” and “useful load to altitude”. Subsequent change options analysis confirmed that the provision of “useful load to altitude” was rated marginally higher than the “time of climb to altitude”, as this useful load attribute encapsulated the value proposition. Given this, climb performance measures were developed and modelled based on maximum useful load to jump altitude within the time of climb requirement. This modelling also supported the sizing of propulsion system to achieve these climb performance requirements. The initial sizing performance modelling also determined battery energy requirements, and hence battery and other system weights which impacted useful load. Climb performance modelling including initial sizing is presented in Appendix 3, via the quantified morphological matrix. This quantified morphological matrix incorporated the weight metrics corresponding to subsystems added using parametric relationships and initial sizing modelling; and predicted climb performance based on initial sizing climb performance modelling.

#### *5.4.1.2 Aircraft costs*

Costs have been incorporated into this methodology as part of the morphological analysis where these costs are based on parametric estimates of hardware costs. In the case of the natural gas fuel modification commuter aircraft mission, cost metrics were based on natural gas fuel tank costs as this was determined to be the most expensive contributor to the natural gas segment costs. Apart from the construction costs associated with ground fuelling infrastructure, aircraft operational costs comprised significant contributor to Life-cycle Costs associated with the natural gas modification. The aircraft annual operating costs were determined using a method described by Gudmundsson (2014), which is based on experiences associated with the actual ownership of a GA aircraft. The primary inputs are flight hours per year, cost of fuel, amount borrowed to fund the aircraft and the natural gas modification, and the associated insurance coverage. As described by Gudmundsson (2014), this can be presented as a flight hour cost as shown in Appendix 1.

In the case of the EP system modification skydiving mission, cost metrics were based on parametric estimates of subsystem costs given by motor, battery and electric controller hardware. Batteries were determined by inspection to be the most expensive contributor to the EP system costs, as this component is likely to be a bespoke solution. The operating costs associated skydiving using an aircraft modified with an electric propulsion system is dependent on several parameters related to energy requirements of the flight, usage profile and the conditions associated with electricity supply. Given that there are currently no electric aircraft used for skydiving at this time, these operating costs were difficult to determine accurately. Given commercial electricity rates and the estimate of the climb energy and reserve energy requirements, an estimate of costs to complete a typical skydiving mission was estimated. However, this estimate did not include battery amortisation costs or demand charges for the reasons as discussed in Appendix 3.

#### *5.4.1.3 Emissions*

The emissions associated with each case study were determined from literature reviews or by inspection. In the case of the natural gas fuel modification commuter aircraft mission, six papers describing the results of tests comparing natural gas emissions with gasoline were analysed to provide the percentage reduction/increase in natural gas fuel emissions, as summarised in Appendix 2.

In the case of the EP system skydiving mission, emissions were not determined directly in the case study. Rather it was observed that an electric aircraft possesses zero emissions at the point of operation.

#### 5.4.1.4 Ground infrastructure performance

The ground infrastructure performance metrics associated with each case study are incorporated into each respective morphological matrix shown in Appendices 1 and 3. In a similar way to aircraft performance measures, the ground infrastructure metrics were dependent on the aircraft mission and role, and the energy type and technologies involved.

In the case of the natural gas fuel modification commuter aircraft mission, the main metrics fell into three categories being cost, fuel storage and fill times. The costs were determined by natural gas fuelling station construction costs comprising buildings, equipment and facilities, and delivery/production costs associated with CNG or LNG fuels, which involved transportation of fuels to site, or the production of fuels onsite. The fuel storage and fill time metrics were determined by the storage time of fuel without degradation or evaporation for a period of 1 week. The fill time metric was based on a refill rate of at least 200 gallons per hour (760 litres per hour).

In the case of the EP system modification skydiving mission, metrics were divided into three broad areas being costs, battery recharge time and flight duty cycle. The cost metric related to the ground charging system equipment costs, and the costs of additional battery units used for exchange. Given that the objective function was to minimise costs, then the number of additional battery sets were minimised in this analysis. The maximum battery recharge time metric was determined for corresponding charging station levels based on an assumption of average recharge times applicable to an average Electric Vehicle (EV), noting that actual recharging time was subject to a combination of factors such State-of-Charge (SOC), Charging Level and battery size. The flight duty cycle metric related to the number of flights achievable during an assumed 8-hour day. This was determined from the recharge time for each system charging level, and the spare battery sets available for exchange after each flight.

#### 5.4.1.5 Uncertainties and sensitivity analysis

Conceptual design, as an early life-cycle activity, is subject to constraints in relation to the fidelity of models and the availability of accurate data. It is accepted that low fidelity models are applied in early design phases with increasing fidelity in later phases. Price et al. (2006) states that early design life-cycle models usually comprise low fidelity simple equations, look up tables with no associated geometry, and data which is subject to some level of uncertainty. The natural gas fuel modification case study undertook a range sensitivity analyses on two parameters based on uncertainties in installed LNG fuel tank drag coefficient ($C_D$) and LNG Specific Fuel Consumption (*SFC*). These uncertainties impacted the Breguet range either directly as a *SFC* term, or indirectly through the Lift/Drag ratio ($C_L/C_D$) term. The sensitivity analysis decreased $C_L/C_D$ ratio by 5% to account for variation in LNG fuel tank installation drag, and increased LNG *SFC* by 10% to account for uncertainties in LNG-related engine performance and *SFC* data. The results are shown in Appendix 2.

In the case of the EP system modification, a sensitivity analysis was undertaken to investigate the impact of battery specific energy density, motor peak specific power and propeller type on useful load and total battery weight. This EP system analysis has focused on useful load and battery weight as these are parameters that determine the viability of the EP system modification as described earlier. The results of this sensitivity analysis is shown in Appendix 3 where useful load and total battery weight as a function of battery specific energy density was determined for the preferred candidate configuration.

This analysis also presented useful load and total battery weight as a function of the motor peak specific power as determined for a range of motors considered in the corresponding morphological matrix. Also shown in Appendix 3 was the relationship of total battery weight with motor peak specific power where the variation in battery weight was proportional to motor peak power.

### 5.4.2 Risks safety and airworthiness

The safety and airworthiness requirements rated as one of the highest for both natural gas fuels and EP systems case studies. Airworthiness certification is dealt with in Step 7 of this design methodology providing a means of documenting certification risks as well as estimating costs. The certification risks were combined with change propagation severity risks to provide a consolidated project risk register as shown by Step 9.

### 5.4.3 Methodology outputs – Step 9

The conceptual design methodology as formulated in this thesis provides as outputs several documents comprising specifications, plans and registers. The development of these documents is dependent on the data and information provided at various steps in the methodology. As described in Chapter 4 and illustrated in Figure 38, these documents comprise the:

- System Design Specification Document (DSD) – This specification document includes the technical, performance, operational and support details for the modified system. It also includes the allocation of functional requirements, and it defines the various functional interfaces as described in the engineering DMM.
- Initial Cost Breakdown Structure (CBS) – This CBS collates the estimated certification and engineering costs determined from the respective DMMs into a single document. This cost breakdown structure is described by Fabrycki and Blanchard (1991), as a means to facilitate the initial allocation of costs on a functional basis using a bottom up approach.
- Draft Certification Program Plan (CPP) – The draft CPP is the only dedicated certification-related output of this design methodology. However, unlike other conceptual design methods, this draft CPP is key to the overall modification effort being the main artefact used by NAAs to determine the adequacy of the proposed approach to the certification of the modification. The key input to this document is the draft compliance summary, which is determined from the Certification DMM.
- Project risk register/matrix – The project risk register is the central record of those change severity risks determined from the change propagation matrices, engineering DMM and certification DMM. This risk matrix will become the central part of the formal risk register for the project in the subsequent design phases.

In order to maintain brevity, Step 9 design outputs are not formally presented or discussed in Appendix 1 or Appendix 3.

## 5.5 METHODOLOGY VALIDATION

This section describes the validation of the conceptual design methodology by verifying and assessing the adequacy of its application through an analysis which examines three characteristic areas. These areas include (1) the coverage of the design space attributes, (2) validation of the process against accepted scientific and industry frameworks and knowledge, and (3) confirmation of accepted techniques, structures and tools within the methodology.

### 5.5.1 Coverage of design space attributes

#### *5.5.1.1 Role and Mission type coverage*

The roles and missions relating to these case studies were very different. The mission profile flown by a natural gas modified commuter aircraft was characterised by a transport mission profile with an enroute climb to a medium altitude and then transit at this altitude for typically 2-5 hours. In contrast, the mission profile of a skydiving aircraft was typically a maximum rate climb flight profile to jump altitude, then a rapid descent with a typical mission duration of 20-25 minutes. The mission profile of a commuter aircraft can be considered to be a point A to B transit profile, whereas a skydiving mission was point A to point A return to departure profile.

#### *5.5.1.2 Scope of systems and subsystems integration*

The case studies were characterised by differences in the scope of subsystems integration. For example, the natural gas fuel systems case study was bounded by modifications to the aircraft fuel subsystems. Whereas, the EP system case study replaced the entire powerplant and selected fuel systems as a retrofit modification. These differences in scope extended to ground segment infrastructure as well as changes to the analysis approach in the areas of initial sizing and flight performance modelling.

#### *5.5.1.3 Data set sources*

The two case studies were differentiated by the data sets which were characterised by the aircraft configuration (twin engine vs single engine), aircraft mission and role, operating environment, technologies, scope of airframe integration, interfaces, and performance metrics. These differences are described in Chapter 3 and are referred to as the design modification space which also includes the ground segment infrastructure, support environment and operating environment.

The data required to support the two case studies were derived from different sources according to the developmental status of the technologies involved. As an aircraft modification, approved data derived from Pilot Operating Handbooks, Flight

Manuals and Maintenance Manuals was used to support development of the modifications. As the technologies underpinning both modifications were developmental, data was drawn from various sources in related air or ground transportation disciplines.

In the case of natural gas fuel systems data was sourced from studies previously conducted on aviation applications of natural gas fuels, or existing ground transportation implementation of natural gas fuel systems technologies. In the latter there was considerable overlap with existing data for ground transportation vehicles using natural gas fuels (fuel tanks, valves, regulators etc.) and refuelling infrastructure, where parametric relationships could be derived, or where extensive data existed.

In the case of EP systems, data was determined from original equipment manufacturers of motors, batteries, electric controllers and ground charging stations, with parametric relationships applied where possible. However, the developmental status of such EP systems meant that data was limited to the few manufacturers in this industry, and the hence the methodology was adapted to incorporate initial sizing techniques and quantisation models that was limited to this data.

Note here that in early conceptual design phases it is not uncommon to derive data from alternate sources for projects that are developmental. In larger projects this may be supported with early concept exploration modelling, simulation and prototyping activities to provide adequate design data in order to progress the design activity.

### 5.5.2 Methodology validation

The validation of the conceptual design methodology is undertaken in this section by assessing the adequacy of its application through analysis of requirements and comparison with accepted scientific and industry frameworks and knowledge.

#### *5.5.2.1 Design methodology requirements*

Pahl et al. (2007) states that in order that a design methodology meet its needs and requirements it must possess various attributes. These attributes which have been detailed and discussed in Chapter 4 form the basis of requirements for this design methodology. In keeping with the matrix-based conceptual design methodology. Table 5 shows an assessment of this conceptual design methodology against the needs and requirements as outlined by Pahl et al. (2007). The assessment is therefore a statement of compliance that this methodology fulfils the general requirements for a conceptual design methodology.

Table 5. Pahl et al. (2007) – Methodology needs and requirements

| Pahl et al. (2007) Methodology needs and requirements | Conceptual design methodology compliance |
|---|---|
| 1. Allow a problem-directed approach, in that it must be applicable to every type of design activity, no matter the specialist field it involves. | This conceptual design methodology is applicable to alternate fuel system and propulsion systems modifications on any small aircraft and related ground infrastructure as demonstrated by case studies shown in Appendices 1 and 3. |
| 2. Foster inventiveness and understanding in searching for an optimum solution. | The conceptual design methodology promotes generation of ideas and concepts through application of morphological matrices to explore the design space for the best solution (refer Fig. 39 Step 2). |
| 3. Be compatible with the concepts, methods and findings of other disciplines. | The conceptual design methodology has adopted and/or adapted accepted scientific principles, industry frameworks and processes as presented and discussed in this Chapter. |
| 4. Not rely on finding solutions by chance. | The conceptual design methodology has adopted an approach that explores the design space through a repeatable and structured process to generate viable solution concepts (refer Fig. 39 Step 2 ). |
| 5. Facilitate the application of known solutions to related tasks. | The conceptual design methodology relies on industry accepted techniques and tools such as QFD, morphological matrices, Pugh's matrices, change option MDMs, CPM, and DSMs techniques applicable to engineering and certification domains (refer Fig 39). |
| 6. Be compatible with electronic data processing. | The conceptual design methodology is implemented in two spreadsheets supporting this thesis (Williams, 2018a, 2018b). This spreadsheet implementation can be developed as a dedicated software tool with the appropriate graphical user interfaces and functions to facilitate ease of use. |
| 7. Be easily taught and learned. | The conceptual design methodology is provided within a matrix-based framework and has adopted techniques and tools which have widespread usage within industry as described in this Chapter. This widespread usage facilitates ease of teaching and learning of the methodology within design teams by using a matrix-based approach shown in Fig. 39. |
| 8. Reflect the findings of cognitive psychology and modern management science, that is reduce the workload, reduce design time, prevent human error, and help maintain an active interest. | The conceptual design methodology aligns with modern management science where it is intended to reduce workload, prevent errors and oversights through implementation of a matrix-based structure. It maintains interest of the design team through consultation with discipline specialists and subject matter experts throughout all design steps. |
| 9. Ease the planning and management of teamwork in an integrated and inter-disciplinary product development process. | The conceptual design methodology promotes teamwork through consultation with team members and experts throughout the design lifecycle as noted in the case studies presented in Appendices 1 and 3. |

| Pahl et al. (2007) Methodology needs and requirements | Conceptual design methodology compliance |
|---|---|
| 10. Provide guidance for leaders of product development teams. | The conceptual design methodology provides a matrix-based framework to which leaders of product teams can refer during each step of the design lifecycle (refer Fig 39). |

#### *5.5.2.2 Synthesis-evaluation-analyses process by Faulconbridge and Ryan (2014)*

Faulconbridge and Ryan (2014) suggest that Systems Engineering processes are developed on the basis of an iterative application of analysis, synthesis and evaluation. The conceptual design methodology as presented in this thesis has adopted this *synthesis-evaluation-analysis* approach, which initially occurs at the systems level, followed by application to the subsystems level, and then at the various lower component levels. This synthesis, evaluation and analysis framework is shown at the top-level by Figure 38, with this thesis structured following these principal elements.

#### *5.5.2.3 Mechanical systems conceptual design steps by Ullman (2010)*

Ullman (2010) describes this conceptual design phase as being primarily concerned with the generation and evaluation of concepts with a functional modelling approach essential for developing concepts that will eventually lead to modified system that is fit for purpose. In this context the mechanical systems conceptual design steps described by Ullman (2010) are reflected in the conceptual design methodology as assessed in Table 6.

Table 6. Ullman (2010) – Mechanical systems conceptual design steps

| Ullman (2010) Mechanical systems conceptual design steps | Conceptual design methodology compliance |
|---|---|
| 1. Generation of concepts. | This is achieved through application of morphological matrices used to generate system concepts applicable to the requirements (refer Fig. 39 Step 2). |
| 2. Evaluation of concepts. | This is achieved through application of the quantised morphological matrices, CPM techniques, the engineering and certification DMMs, and concept evaluation against technical performance measures (refer Fig. 39 Steps 2 through 7). |
| 3. Making concept decisions. | This is achieved through application of the quantised morphological matrices, change options MDM, and PM techniques (refer Fig. 39 Steps 2, 3 and 4). |
| 4. Documenting and communicating. | This is achieved through development of the design methodology outputs as provided by the DSD, CBS, CPP and Project risk register/matrix (refer Fig. 39 Steps 8 and 9) |

| Ullman (2010) Mechanical systems conceptual design steps | Conceptual design methodology compliance |
|---|---|
| 5. Redefining plans. | This is achieved by the design methodology outputs as described above (refer Fig. 39 Step 8). |
| 6. Approving concepts. | The main output of the methodology is the DSD, which is the document that describes the modification technical, performance, operational and support characteristics for use in the next phase in the system design life-cycle. |

#### *5.5.2.4 Trade-off analysis by Faulconbridge and Ryan (2014)*

Faulconbridge and Ryan (2014) state that a trade-off analysis is one of the tools available to undertake evaluations within a conceptual design framework. In this context, the steps of a trade-off analysis described by Faulconbridge and Ryan (2014) comprise several steps which are also incorporated in the conceptual design methodology as assessed in Table 7.

Table 7. Faulconbridge and Ryan (2014) – Trade-off analysis

| Faulconbridge and Ryan (2014) Trade-off analysis | Conceptual design methodology compliance |
|---|---|
| 1. Definition of requirements. | The QFD matrix is used to define, record and analyse the modification requirements as a step within the conceptual design methodology (refer Fig. 39 Step 1). |
| 2. Identification of alternative solutions. | The quantified morphological matrices are used to generate system concepts applicable to the requirements (refer Fig. 39 Step 2). |
| 3. Nomination of selection criteria such as metrics. | The QFD matrix along with the quantified morphological matrices are used to nominate selection criteria and metrics (refer Fig. 39 Steps 1 and 2). |
| 4. Determination of criteria weighting. | Criteria weightings are incorporated into the quantised morphological matrices and Pugh matrices (refer Fig. 39 Steps 2 and 3). |
| 5. Definition of scoring functions. | Scoring functions are incorporated into the quantised morphological matrix through normalising scores and determination of a *FoM* for compatible solutions (refer Fig. 39 Step 2). |
| 6. Evaluation of alternatives. | Several techniques and tools are used including quantisation of the morphological matrices, CPM, engineering and certification DMMs, and concept evaluation against technical performance measures (refer Fig. 39 Steps 2 through 7). |

| Faulconbridge and Ryan (2014) Trade-off analysis | Conceptual design methodology compliance |
| --- | --- |
| 7. Sensitivity studies | Sensitivity studies are conducted within the design methodology and are based on metric uncertainties, key technical performance measures and derived parameters (refer Fig. 39 Step 8). |

#### *5.5.2.5 Feasibility analysis and trades studies by Blanchard and Fabrycki (1998)*

Blanchard and Fabrycki (1998) state that the accomplishment of a feasibility analysis or a trade study is a major step within conceptual design that involves three main steps. These three main steps are inherent in this conceptual design methodology as assessed in Table 8.

Table 8. Blanchard and Fabrycki (1998) – Feasibility analysis

| Blanchard and Fabrycki (1998) Feasibility analysis | Conceptual design methodology compliance |
|---|---|
| 1. Identification of possible design approaches. | Morphological matrices are used to generate and identify system concepts and design approaches applicable to the requirements (refer Fig. 39 Step 2). |
| 2. Evaluation of these approaches based on performance, effectiveness, maintenance, logistic support, and cost economics. | Evaluation of design approaches are achieved by the application of quantised morphological matrices, CPM, engineering and certification DMMs, and concept evaluation against technical performance measures for both aircraft and ground infrastructure segments (refer Fig. 39 Steps 2 through 7). |
| 3. A recommendation of the preferred course of action. | The design methodology outputs are provided by the DSD, CBS, CPP documents and the Project risk register/matrix. Recommendations are provided in the DSD as the key document recording this design process. |

#### *5.5.2.6 Evaluation of solution variants by Pahl et al. (2007)*

The evaluation of solution variants, as described by Pahl et al. (2007) involves several steps. These steps are inherent in this conceptual design methodology as assessed in Table 9.

Table 9. Pahl et al. (2007) – Evaluation of solution variants

| Pahl et al. (2007) - Evaluation of solution variants | Conceptual design methodology compliance |
|---|---|
| 1. Identification of evaluation criteria. | The QFD matrix along with the quantified morphological matrices are used to identify evaluation criteria and metrics (refer Fig. 39 Steps 1 and 2). |
| 2. Weighting of evaluation criteria. | Evaluation criteria weightings are incorporated into the quantised morphological matrices and the PM (refer Fig. 39 Steps 2 and 3). |
| 3. Compiling parameters. | Several techniques and tools are used including quantisation of the morphological matrices, PM, and engineering and certification DMMs (refer Fig. 39 Steps 2 through 7). |
| 4. Assessing values. | Assessment of design value is achieved by the application of quantised morphological matrices, CPM, engineering and certification DMMs, and concept evaluation against technical performance measures for both aircraft and ground segments (refer Fig. 39 Steps 2 through 8). |
| 5. Determining overall value. | Determination of overall value is achieved by application of the of quantised morphological matrices where all metrics are incorporated (refer Fig. 39 Step 2). |
| 6. Comparing concept variants. | Several techniques and tools are used including quantisation of the morphological matrices, CPM, engineering and certification DMMs, and concept evaluation against technical performance measures (refer Fig. 39 Steps 2 through 8). |
| 7. Estimating evaluation uncertainties. | Sensitivity studies are conducted within the design methodology and are based on metric uncertainties, key technical performance measures and derived parameters (refer Fig. 39 Step 8). |
| 8. Searching for weak spots. | The design methodology requires design advice from discipline specialists and subject matter experts at selected steps in the process as described in Appendices 1 and 3. These design team inputs are particularly important in identifying impacts of change propagation on engineering design and certification activities. |

### 5.5.3 Application of accepted tools, methods and techniques

As described in Chapter 4, this conceptual design methodology has adopted and adapted scientific and industry tools, methods and techniques within a matrix-based framework. These tools, methods and techniques are described in detail in Chapter 4 and are implemented in the two case studies shown in Appendices 1 and 3. They are summarised below for completeness and to establish the veracity of this methodology:

- QFD matrix – Also known as the House of Quality - Blanchard and Fabrycki (1998) and Gudmundsson (2014).

- Quantified Morphological Matrix (QMM) - Gavel et al. (2006, 2007), Gavel et al. (2008), Ölvander et al. (2009), and Svahn, (2006).
- Pugh's pairwise comparison matrix (PM) – Burge (2009)
- Change options MDM – Koh et al. (2012).
- Change Propagation Method (CPM)– Clarkson et al. (2001), Eckert et al. (2001), Eckert et al. (2009), Eckert et al. (2006), Keller et al. (2005), Koh & Clarkson (2009) and Koh et al. (2012).
- Engineering and Certification DMMs - Developed for this methodology and is based on Design Structure Matrix methods as described by Cross (1994).

# Chapter 6. Conclusions

## 6.1 SUMMARY

This thesis has presented a conceptual design methodology that provides a framework from which to evaluate modifications on small civil aircraft. This design methodology has been formulated in response to problems associated with the development of design modification concepts and predicting the impact of these changes as they propagate through the various subsystems and certification requirements. It also provides a means to estimate project costs, determine and document risks and to provide project documentation supporting the next design phase. In order to triangulate this methodology, two case studies were selected involving two different alternate fuel system technologies which are to be integrated or retrofitted on two aircraft of different types, operating in two different roles. The first case study involved a modification to a small civil commuter aircraft to incorporate a natural gas fuel system with cleaner emissions as compared to conventional AVGAS fuels. The second case study involved a modification which retrofits an electric propulsion system to a small 4-seat light aircraft for skydiving.

This conceptual design methodology has been formulated within a framework described by Faulconbridge and Ryan (2014), where the processes are built around an iterative application of *analysis*, *synthesis* and *evaluation* at the top level. At the lower levels, the methodology has used this framework to link together existing techniques and tools such as Quality Function Deployment matrices, Gavel et al. (2006, 2007) and Ölvander et al. (2009) Quantified Morphological Matrix, Pugh’s decision matrices, Koh et al. (2012) Change options Multiple-Domain Matrix, and Clarkson et al. (2001) Change Propagation Method matrices. This methodology is extended however to develop Engineering and Certification Domain Mapping Matrix techniques, based on Design Structure Matrices to evaluate the impact of design modification changes on engineering and certification risks and costs. This thesis has formulated a matrix-based conceptual design methodology which encapsulated each step as a matrix where the information and data is used to evaluation the performance, cost and risk severity attributes of the design modification.

This matrix-based methodology enables a structured approach, where the outputs of one stage forms the basis of the next, accounting for the aircraft system modification design lifecycle through (1) development and management of requirements, (2) generation of concepts, (3) concept selection validation, (4) evaluation of design changes, (5) evaluation of design impacts and certification impacts, and (6) evaluation of concept performance. Each of these matrices have outputs which are either used as feedback for earlier steps or are used as inputs to the next steps.

The requirements analysis and functional decomposition steps are significantly simplified as the result of a defined aircraft system modification, where the modification adopted system functions as defined by the pre-existing aircraft, as well as the supporting infrastructure. This is captured in the early steps through application of Quality Function Deployment matrices and Pugh pairwise comparison techniques. Generation of systems concepts is achieved through application of the quantified morphological matrices where the system requirements analysis and functional decomposition steps are imbedded in the formulation of this matrix. This morphological technique has the benefit that the complete design space is explored with all combinations of systems/subsystems functions considered, as described by Ölvander et al. (2009), Gavel et al. (2006, 2007) and others. In this instance, this quantified morphological matrix approach was applied using simple relationships representing Technical Performance Measures (TPMs) (or metrics) and then rating and combining these metrics to identify a candidate solution.

One focus of this conceptual design methodology was to incorporate a step to evaluate and assess the impact of changes propagation throughout the system as a result of the modification. There were two aspects to this evaluation being an assessment of change options as described Koh et al. (2012) and an assessment of change propagation impacts to aircraft systems, subsystems and components as described by Clarkson et al. (2001) and others. The change options technique as described by Koh et al. (2012) was applied early in the methodology to assess and feedback those requirements that are of the most importance to the design. The change propagation method as described by Clarkson et al. (2001) assessed the impact of changes brought about by the modification, and was a simplified representation of the Change Prediction Method (CPM) to highlight change severity risks. These changes were assessed using change severity matrices which captured the change severity risks at system, subsystem, and component levels. The risks were recorded in a traditional risk matrix and were visualised via change propagation tree and case risk plots, in order to inform further analysis in later design phases.

The latter steps of this design methodology applied Design Structure Matrix (DSM) techniques to evaluate engineering and certification impacts of changes resulting from the alternate fuel system or propulsion system modification. In this instance, change propagation techniques as described by Clarkson et al. (2001) and others were adapted to these engineering and certification Domain Mapping Matrices (DMMs) to determine the impact of the modification. The engineering DMM assessed these changes at the system requirements, subsystems and component level using as a basis the requirements and design changes documented in the QFD matrix, the Change options Multiple-Domain Matrix (MDM), and the results of the CPM. The certification DMM followed a similar approach, with the main difference being that change impacts were evaluated against the respective airworthiness design standard whether it be aircraft, engine or any other applicable standard or code. The main benefits of such an approach is that all airworthiness requirements are assessed for the impact of the modification and change severity risks are determined accordingly.

The conceptual design methodology as formulated in this thesis provided as outputs, several documents comprising project specifications, plans and registers. The development of these documents was dependent on the data and information provided at various steps in the methodology, and comprised the System Design Specification Document (DSD), the Initial Cost Breakdown Structure (CBS), the Draft Certification Program Plan (CPP) and the Project risk register/matrix. This methodology did not extend into the regulatory domain past the draft Certification Program Plan (CPP) document as provided by the certification DMM, nor does this methodology extend into the draft project management artefacts, which would traditionally accompany such conceptual design phases.

This conceptual design methodology has introduced two original contributions to the state of knowledge in this discipline which are described in the following sections. The first involves the combination of existing and accepted techniques and tools to formulate a matrix-based framework as described in this thesis. The second relates to the extension of this methodology into engineering and certification domains in order to conduct a robust evaluation of project costs and risks.

## 6.2 CONTRIBUTION TO FIELD OF CONCEPTUAL DESIGN METHODOLOGIES

This methodology as illustrated in Figure 38, encapsulates *synthesis-evaluation-analysis* processes within a matrix-based framework using existing techniques and tools, and has been triangulated by two case studies as presented in this thesis. These case studies were selected to provide a diverse design modification space to trial the methodology.

The validation of this conceptual design methodology was achieved by verifying and assessing the adequacy of its application through an analysis which examined three areas. These areas included coverage of the design space attributes; validation of the methodology against accepted scientific and industry frameworks; and confirmation of accepted techniques, structures and tools within the methodology.

1. Coverage of the design space was achieved through selection of case studies encompassing differing aircraft roles and mission types, differing technologies and subsystems integration scope, and different data sources/ collection and analysis methods.
2. Validation of this conceptual design methodology was achieved in two parts. This first demonstrated compliance against the needs and requirements of a design methodology as stated by Pahl et al. (2007). The second compared and evaluated the conceptual design methodology against processes utilised in mechanical design, product design, trade studies, feasibility analysis and systems engineering.
3. Lastly the conceptual design methodology has adopted and adapted existing scientific and industry tools, methods and techniques within a matrix-based framework as described in this thesis.

## 6.3 CONTRIBUTION TO EVALUATION OF DESIGN CHANGE IMPACTS ON ENGINEERING AND CERTIFICATION

Traditionally conceptual design methodologies embrace existing techniques and tools within a framework that focus on requirements, design synthesis and generation of concepts, ranking of concepts and then analysis which assesses whether the candidate solution meets requirements. The extension of this methodology into engineering and certification domain was undertaken to ensure that important modification-related risks and costs were incorporated into the early stages of design. The extension adopts existing DSM and DMM-based techniques and tools to evaluate the impact of changes to subsystems and hence impact of risks and costs resulting from the case study modifications. This approach also applies the change propagation method analysis techniques as described in this thesis.

### 6.3.1 Engineering DMM

The Engineering DMM accounts for the discrete engineering activity as part of the conceptual design phase by providing a means to evaluate and mitigate risk and to estimate development cost. These engineering activities, like other steps in the conceptual phase are iterative in nature, with the engineering DMM developed to incorporate change propagation method analysis techniques in a matrix-based framework utilised in other parts of the methodology. There were two main outputs of this DMM-based evaluation. The first was the draft Systems Specification Document detailing the aircraft modification and the ground infrastructure. The second was the input into the initial CBS which estimated engineering costs associated with the aircraft modification and the ground station infrastructure. Both of these documents are prepared as inputs to the next modification design phase.

### 6.3.2 Certification DMM

It was noted that the impact of new technologies on the aircraft Type Certification Basis (TCB) was often overlooked in early conceptual design studies. In some cases, airworthiness design standards may not always cater for new technologies therefore necessitating design changes to the modification or the formulation of changes to the design to ensure that compliance could be achieved. Therefore, design changes made necessary by certification may also be accompanied by additional propagation impacts.

In this context, a new framework was developed and implemented via a certification DMM matrix in a similar way to that described for the engineering DMM. Although the focus is airworthiness certification, the technique can also be extended to any system needing to show compliance to a design standard or code.

It is noted that the impact of changes to an aircraft resulting from a modification is often assessed using a draft compliance summary. For small aircraft this is a framework based on FAR Part 23 or CAR 3 design standard requirements. In this thesis a multi-part mathematical process is introduced which derives a number of matrices that equate to the relevant airworthiness design standards. The key output of the certification DMM was the draft CPP, which was the only dedicated certification-related output of this design methodology. This draft CPP is a key document supporting the overall modification effort being the main artefact used by NAAs to determine the adequacy of the proposed approach to the certification of the modification. Again, this draft CPP is prepared as an input to the next design phase.

## 6.4 LIMITATIONS

This thesis has been limited insofar that the case studies have not evaluated all aircraft modification lifecycle costs. Rather the approach has been to present the bottom-up engineering and certification cost estimation methods activities which form part of the extension of new knowledge in this area. As described in the case studies, production and through life support costs have not been incorporated as part of the conceptual design phase. Rather these costs are to be evaluated as part of the next design phase. Notwithstanding this, the conceptual design methodology makes provision for cost estimation via an initial cost breakdown structure for engineering and certification activities and operating costs associated with the aircraft and ground infrastructure. However, actual costings have not been developed in the case studies for reasons as outlined above, and it is anticipated that further research may be required to augment this initial approach with causal techniques, parametric methods, or an extension to the bottom up approach.

Although this research has implemented existing techniques and tools to evaluate alternate fuel system modifications on small aircraft, the application of some techniques was simplified to facilitate implementation in a conceptual design phase. One such technique was the Change Propagation Method by Clarkson et al. (2001) where only direct change impacts were evaluated, with a full analysis involving indirect change impacts to be undertaken in the following design phase where required.

## 6.5 EXTENSIONS TO METHODOLOGY

### 6.5.1 Other design space applications

Although this conceptual design methodology has demonstrated its validity through application to two case studies involving small aircraft alternate fuel modifications, it is highly likely that it can be applied to problems involving other complex aeronautical system modifications such as role equipment installation on military fixed wing and rotary wing aircraft. As well as manned aviation, the method could also be employed to unmanned aerial systems (UAS) modifications where design change propagation has a significant impact to these highly integrated systems.

As demonstrated in this thesis, the methodology caters for various civil certification domains including those airworthiness standards defining aircraft and engine products. Likewise, it can cater equally well for other standards, codes and regulations related to aviation ground infrastructure and equipment, with no limit to this decomposition. The combination of airworthiness certification standards, related ground infrastructure codes into one large DMM, can facilitate a broader analysis that can sort and categorise certification dependencies on a systems level. This would then become a useful tool that can be used to establish the *system certification basis*. This would differ from current approaches which consider the certification each product separately, not as an integrated system. A large *system certification* DMM would present challenges due to its increased size, necessitating a solution using dedicated software, rather than spreadsheets.

Given that this design methodology incorporates ground-based infrastructure, it can be easily seen that it could be applied to other infrastructure where changes are to be incorporated to a complex system. For example, this methodology could be applied to manage early life-cycle design processes involving modifications to power generation assets, rail transportation hardware and networks, and mining infrastructure and equipment.

### 6.5.2 Project management related outputs

This thesis has been limited to those conceptual design outputs concerned with specifying the initial design and ensuring that adequate plans are provided to support the next design phase effort. However, it must be said that that outputs of the methodology could be extended to provide documentation that traditionally falls within the project management discipline. For example, with some modifications to the engineering and certification DMMs, the same information could be used to develop an initial Work Breakdown Structures (WBS) and a draft Project Management Plan (PMP) which would draw on the information from all steps of the methodology. This would therefore provide a set of engineering conceptual design and project management related documentation which can be used as the basis for the next phase of the design lifecycle.

## 6.6 RECOMMENDATIONS

It is therefore recommended that two areas could be pursued in relation to the development and refinement of this conceptual design methodology. Further research could be undertaken to address the limitations of this conceptual design methodology in the following areas:

1. Provision of actual costings of case studies in relation to engineering and certification and operation costs. Considerations should be given to determination of actual costings for all lifecycle phases including production and manufacturing in order to evaluate the true cost of alternate fuel systems.
2. Extension of the Change Propagation Method by Clarkson et al. (2004) to include direct and indirect change evaluation and also inclusion of a means to automate the production of partial change propagation trees, risk scatter plots and network dependency diagrams to help visualise change impacts.

Lastly, this methodology should be trialled within other disciplines involving complex systems modifications. Whilst this thesis has focussed on innovative propulsion technologies applied as modifications to small aircraft, the methodology may have broader applications to other disciplines such as modifications to road and rail transportation assets, power infrastructure, and mining equipment.

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# Appendix 1. NG Fuels – Case study

## 1.1 BACKGROUND

This Appendix details the matrix-based framework associated with the conceptual design methodology applied to a case study involving modification of a commuter aircraft to utilise natural gas fuels. The underpinning theory associated with this design framework has been detailed in Chapter 4. This case study illustrates the implementation of this methodology as described in Chapter 5 from the requirements management stage through several steps to development of the initial systems specification. As stated in Chapter 5, this methodology is matrix-based, therefore enabling a structured approach where the outputs of one stage forms the input for the next. Each of these matrices have outputs which are either used as feedback for earlier steps, or are used as inputs to the next steps.

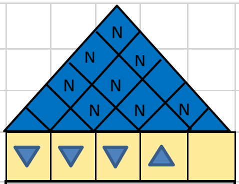
## 1.2 REQUIREMENTS – STEP 1

Blanchard and Fabrycki (1998) describe Quality Function Deployment (QFD) (also known as the House of Quality) as a method intended to help in the design of complex products, by taking various customer requirements into account by using a selection matrix that helps evaluate the impact of these requirements on areas such as the engineering development. The QFD matrix is a specialised matrix, depicting a sketch of a house, designed to convert customer requirements into a numeric score that helps define areas for conceptual design.

### 1.2.1 Quality Function Deployment matrices

The preparation of a QFD matrix is best explained through this case study as follows. Generally, the QFD matrix consists of several matrices that focus on different aspects of the development of a product as shown in Table 10.

Table 10. Aircraft natural gas modification – QFD matrix




| Customer needs | Aircraft Mod Spec Requirements | Importance 1 (not) - 5 (very) | | Modification engineering challenges: Drag impact | Weight impact | LCC impact | Power impact | Size impact | |
|---|---|---|---|---|---|---|---|---|---|
| High performance - Low drag and low weight | The modification shall minimise the impacts to range, payload, & cruise speed performance. | 4 | 17.4% | 9 | 9 | 9 | 9 | 3 | |
| Safety & airworthiness | The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 5 | 21.7% | 1 | 1 | 3 | 1 | 1 | |
| Low cost | The modification lifecycle costs (LCC) shall be kept to a minimum | 3 | 13.0% | 3 | 9 | 9 | 3 | 3 | |
| Environmentally friendly - Low emissions | The modification shall minimise emissions. | 3 | 13.0% | 3 | 1 | 1 | 1 | 1 | |
| Compatibility | The modification shall be compatible with a range of commuter aircraft types. | 4 | 17.4% | 1 | 1 | 1 | 1 | 1 | |
| Spacious | The modification shall not impact passenger, baggage or cargo space. | 4 | 17.4% | 3 | 1 | 3 | 3 | 9 | |
| | SUM = | 23 | 100.0% | 3.26 | 3.43 | 4.22 | 3.0 | 3.0 | 16.91 |
| | | | | 19.3 | 20.3 | 24.9 | 17.7 | 17.7 | |

It has been assumed that customer surveys have collected data and information supporting the development of the natural gas fuel modification with needs established along with desired requirements. In this case a simplified set of requirements have been determined which are consistent with those described in Gudmundsson, (2014). It should be noted that an actual QFD matrix would have more than five requirements, but is limited here in this case study for brevity reasons. In this case study the needs are for a high performance, safe, inexpensive, environmentally friendly and spacious (does not impact passenger or cargo space) natural gas fuel system modification which is compatible with commuter aircraft. It is assumed here that the customer survey has requested that potential customers rate the corresponding requirements using values between 1 (not important) and 5 (very important). This is placed in a matrix which is shown on the left-hand side of the QFD matrix in Table 10. In this context, 'high performance' has a rating of 4.0 (important), 'safe' has a rating of 5.0 (very important), and so on. These ratings are then added and the sum 23 is entered as shown. The column to the right shows the percentages of the ratings referenced to the overall score total of 23.

The next step requires the design team to list the number of modification engineering challenges that relate to the customer requirements. For instance, the requirement for 'high performance' calls for special attention to the drag and weight characteristics of the aircraft. These engineering modification challenges are shown on

Table 10 as drag impact, weight impact, Life Cycle Cost (LCC) impact, Power impact and Size impact. These engineering challenges are defined in Table 11 and are revisited later in this section.

Table 11. Aircraft natural gas engineering challenge definitions

| | Engineering challenges definitions |
|---|---|
| Drag impact | This relates to the impact of the modification on overall aircraft drag and performance |
| Weight impact | This relates to the impact of the modification on empty weight and hence payload capability |
| Life Cycle Cost impact | This relates to the impact of the modification on aircraft LCC |
| Power impact | This relates to the impact of the modification on propulsion system power output and/or related changes |
| Size impact | This relates to the impact of the modification on aircraft volume and/or space affecting payload carriage capability |

The ‘roof’ shown on Table 10 is the triangular region above the engineering challenges matrix. This roof is used to indicate interrelationships between the various engineering challenges. The roof consists of two parts: the ‘roof’ itself, and the ‘fascia’ as described by Gudmundsson, (2014). The ‘fascia’ is used to indicate whether the engineering challenge as listed (e.g. ‘drag’ or ‘weight’) has a favourable effect on the product. The ‘roof’ itself, is depicted by the triangular region shown as the diagonal lines in Table 10. It is used to indicate positive and negative relationships between the challenges. These are typically denoted with symbols but are represented using the same terminology as shown in Gudmundsson, (2014). This rating scale terminology along with the definitions is shown in Table 12. It should be noted that the build-up of these relationships is highly dependent on interpretation, requiring the design team to reach consensus. This design team activity would follow the example approach as described in Gudmundsson, (2014). Once complete, the example letter combinations are entered as shown in Table 12.

Table 12. Rating scale definitions

| | |
|---|---|
| NN | Means that there is a strong negative relationship between two engineering challenges |
| N | Means that there is a negative relationship |
| P | Means that there is a positive relationship |
| PP | Means that there is a strong positive relationship between the two engineering challenges |

The next step is assigning weightings to the engineering challenges as they relate to the customer requirements. This is accomplished using the interrelationship matrix as shown in the main body of the QFD matrix. Table 13 shows these rating scales as devised by Gudmunsson (2014) and used in the examples of this method. Again, this process should be achieved through design team consultation. Nevertheless, the guidance as provided in Gudmunsson, (2014) is used as the basis for this scoring with some changes to account for the impact of the modification rather than the holistic aircraft design.

Table 13. Rating scales for interrelationship matrix

| | |
|---|---|
| 9 | Means that the customer requirement has great influence |
| 3 | Means that the customer requirement has moderate influence |
| 1 | Means that the customer requirement has weak influence |

The target matrix sits below the interrelationship matrix and represents the results of a cross-multiplication and summation that is used to determine where to place the most effort during the development of the product. Consider the percentage column of the customer requirements matrix as shown in Table 10 (17.4%, 21.7%, 13.0%, 13.0% etc.) and the first column of the technical requirements column ('Drag impact', 9, 1, 3, 3, etc.). These are multiplied and summed as follows:

$$0.174\times9 + 0.217\times1+ 0.13\times3+ 0.13\times3+0.174\times1+0.174\times3 = \mathbf{3.26}$$

The remaining columns are multiplied in this fashion, always using the percentage column of the customer requirements, yielding 3.43, 4.22, 3.0, and 3.0, and so on. The next step converts the results into percentages. Firstly, all the results given by 3.26, 3.43, 4.22, 3.0, and 3.0 are added to total 16.91. Secondly, for the first column, the percentage of the total is calculated thus 100% × 3.26/16.91 = 19.3%, and so forth for the remaining columns. These numbers are the most important part of the QFD matrix, as the highest value indicates where most of the development effort should be spent. In this QFD analysis the 'LCC and 'weight' and 'drag' are the three areas that should receive the greatest attention during the conceptual design phase. It should be noted also that there is very little difference between 'weight' and 'drag' percentages, so these modification engineering parameters should be given equal focus behind LCC.

As stated earlier, the provision of the natural gas fuel system modification installed on a commuter aircraft not only affects the aircraft segment, but also the ground segment. The ground segment comprises the natural gas fuelling station infrastructure which provides the natural gas fuelling capability for the modified aircraft. The ground segment is therefore an important part of the overall systems capability, and hence costs associated with the construction, certification and operation of this segment can have a major impact on the viability of this concept. Therefore, the ground segment QFD matrix is analysed in the same way as the aircraft segment described above.

Table 14 shows the QFD matrix for the natural gas fuelling station segment, with the same sub-matrices as described above. In this particular case, the 'fuel quantity storage', 'fill time' and 'costs' are the three areas that should receive the greatest attention during the conceptual design phase.

Table 14. Natural gas fuel station – QFD matrix

| | | | | Engineering challenges | | | | |
|---|---|---|---|---|---|---|---|---|
| | | | | ▲ | ▲ | | ▲ | |
| **Customer needs** | **Ground refuelling infrastructure requirements** | **Importance 1 (not) - 5 (very)** | | **Fill time** | **Fuel quantity storage** | **Station location** | **Low running costs** | |
| Fuel storage time | The ground refuelling station shall be able to store fuel without degradation or evaporation for a period of 1 week. | 4 | 20.0% | 1 | 9 | 3 | 3 | |
| Refuelling fill time | The ground refuelling station shall provide a refill rate of at least 200 gallons per hour (GGE). | 4 | 20.0% | 3 | 3 | 1 | 3 | |
| Fuel storage quantity | The ground refuelling station shall provide sufficient fuel for 10 aircraft at 100 US gallons per day (GGE) with a holding capacity for 10 days. | 4 | 20.0% | 3 | 9 | 9 | 3 | |
| Safety | The ground refuelling station shall comply with CNG and LNG filling station safety standards (NFPA 52, the Vehicular Gaseous Fuel Systems Code, CFR Title 49, Part 193, NFPA 59A, Standard for the Production, Storage, and Handling of Liquefied Natural Gas (LNG)) | 5 | 25.0% | 9 | 3 | 1 | 3 | |
| Low cost | The ground refuelling station shall be designed to minimise capital and operating costs. | 3 | 15.0% | 9 | 3 | 3 | 9 | |
| | SUM = | 20 | 100.0% | 5 | 5.4 | 3.3 | 3.9 | 17.6 |
| | | | | 28.4 | 30.7 | 18.8 | 22.2 | |

Like the aircraft segment QFD analysis, the definitions of these engineering challenges are provided in Table 15. It can be seen that the assumptions implicit in this analysis place priority on fuel quantity storage and usage rate for a stated period.

Table 15. Natural gas fuel station engineering challenge definitions

| | Engineering challenges definitions |
|---|---|
| Fill time | This relates to achieving the required fill time comparable to AVGAS refuelling operations. |
| Fuel quantity storage | This relates to sizing the fill station to support the fuel quantity usage for the stated period of time. |
| Siting | This relates to siting of the fill station to minimise truck delivery distances or provide access to NG pipeline. |
| Low running costs | This relates to the design of the fill station to minimise operation costs and maintenance costs. |

### 1.2.2 Initial sizing - Fuel tank and location constraints

It should be noted that this analysis has been confined to natural gas fuel tank sizes (and volumes) which are constrained by the existing aircraft configuration and geometry. For example, natural gas wing tip fuel tanks will be sized to approximate the existing AVGAS wing tip tanks. However, some variations will be necessary to provide a cylindrical cross section pressure vessel as opposed to the aerodynamically profiled AVGAS fuel tank. Physical constraints also apply to the under-fuselage natural gas belly tank where ground clearance requirements limit the physical diameter of the tank. Furthermore, the length of this natural gas belly tank is limited by nose landing gear door (forward), the takeoff rotation ground clearance (aft), and the profile of the fairing which is required to reduce aerodynamic drag. These physical constraints limit the sizes and volumes of these fuel tanks using in this conceptual design case study. Optimisation of these fuel tank geometries would therefore take place in the later design phases.

It was decided that the conceptual design would eliminate any configurations that included in-fuselage LNG tank storage. This design decision eliminates those fire risks and hazards associated with natural gas fuel leakage from fuel tanks or fuel lines within the fuselage.

## 1.3 CONCEPT GENERATION – STEP 2

### 1.3.1 Overview

The structured generation of concepts within the design space is a challenge well suited to the morphological approach. This approach can be augmented by quantisation of the matrix using properties of the key technical performance measures (or metrics) where the best solution or smaller set of solutions can be ranked for further analysis or evaluation. In this case study, concept selection is based on quantisation of the morphological matrix based on key metrics as determined in the previous QFD step. This structured approach explores the complete design space, and down-selects the ‘best’ potential solutions using simple relationships based on minimisation and maximising these metrics as a Figure of Merit (*FoM*).

### 1.3.2 Morphological matrix

The morphological matrix is a technique that supports design synthesis through assisting the design team to identify and generate combinations of systems, sub-systems or components as described in Chapter 5.

#### *1.3.2.1 Natural gas solution space*

There are two natural gas fuel tank solutions considered in the case study, being those associated with CNG or LNG fuel tank storage. The natural gas fuel tanks can be installed in two possible locations by either replacing the existing wing tip tanks, or by the installation of a new fuselage belly tank installation, as shown in Figure 40. It should be noted that three solutions for fuel tank locations exist, being the wing tip tank replacement, the new fuselage belly tank, and both locations combined. In addition, bi-fuel solutions are possible where CNG/AVGAS or LNG/AVGAS bi-fuel combinations being available. Bi-fuel arrangements are common in ground transportation applications. Indeed, a prototype bi-fuel CNG/AVGAS modified light aircraft is described in Chapter 3 as an example of this configuration. A major advantage of a bi-fuel configuration is that it can operate on AVGAS fuels for takeoff and landing phases of the flight, and then switch to natural gas fuels during the cruise or descent phases of the flight. This option would provide maximum operational flexibility through use of two fuel type options as well as providing potentially lower operating costs, and reduced emissions.

It should be noted that this concept relies on a bi-fuel arrangement that can provide engine operation on AVGAS, or natural gas, via a valve to switch between the two fuels. This arrangement is consistent with a bi-fuel system architecture of most automotive and heavy vehicles. However, there are dual-fuel arrangements where both fuel and natural gas are premixed and then injected into the engine. This arrangement is common in diesel engines used in trucks and heavy vehicles. This arrangement is not considered in this thesis.

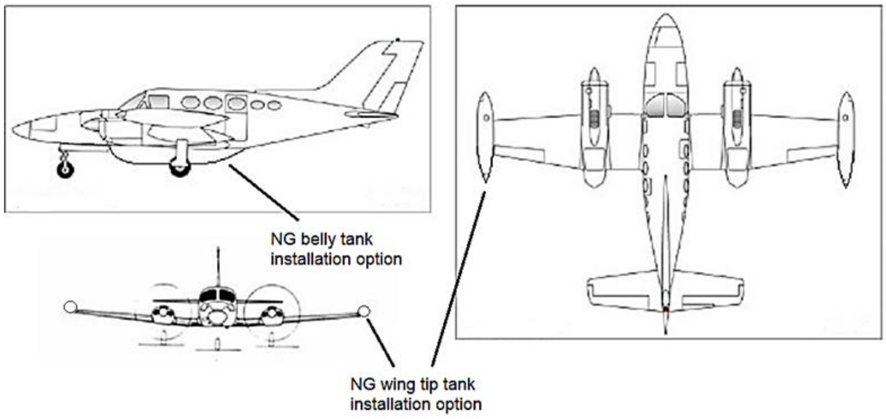



Figure 40. Cessna 421B – Examples of natural gas fuel tank configuration concepts

As stated in Chapter 3, the Cessna 421B possesses AVGAS integral wing tanks (one in each wing) and wing tip fuel tanks (on each wing tip). The integral wing tanks are not replaced or removed as part of this natural gas modification. However, the wing tip tanks may be replaced with natural gas wing tip tanks depending on modification options selected. Therefore, the combinations of potential solutions are constrained to using particular AVGAS fuel tanks when bi-fuel natural gas options are selected. The logical condition set for bi-fuel solutions is summarised in Table 16, and this is reflected in the morphological matrix at Table 17.

Table 16. Allowable tank combinations for NG/AVGAS bi-fuel arrangements

| NG tank tank(s) | Allowable AVGAS loading combination |
|---|---|
| Belly tank | • Main wing fuel tanks<br>• Wing tip fuel tanks<br>• Main wing & wing tip fuel tanks |
| Belly tank & wing tip tanks | • Main wing fuel tanks only |
| Wing tip tanks | • Main wing fuel tanks only |

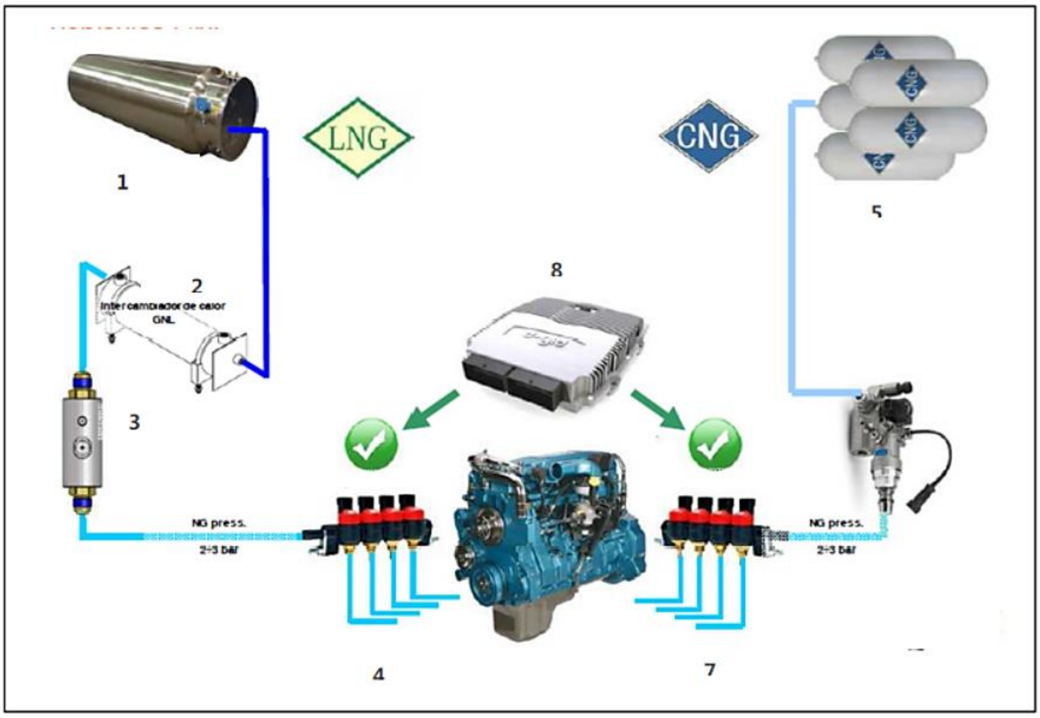
Table 17 shows the morphological matrices for both the aircraft segment and ground infrastructure segment relating to the natural gas fuel modification. Both aircraft and ground segment morphological matrices are incorporated in one table to show that these two segments are related to the overall systems capability.

On this basis a segment concept can be generated by selecting one solution for each functional sub-system as indicated by the dashed lines in Table 17. These lines are shown for both the aircraft segment and ground infrastructure segment.

The aircraft segment natural gas morphological matrix includes the main functional descriptions which are used to define the segment concept as well as functional descriptions relating to natural gas fuel system components. Only the main functional descriptions are used to generate a segment concept with these highlighted in Table 17. Functional descriptions relating to other lower level natural gas fuel system components are presented for completeness to illustrate similarities between the two fuel system types.

#### *1.3.2.2 Commonality of architecture*

Both CNG and LNG fuel components have common functions and hence similar fuel system architectures. Engines installed in both CNG and LNG vehicles are fed by natural gas in gaseous state. Once the fuel leaves the cryogenic storage tank (on demand by the engine), it enters the heat exchanger/vaporiser and then flows to the pressure regulator, where its pressure conditions are adapted to those required by the engine inlet. Figure 41 describes in general terms the main devices involved in both technologies, with LNG technology on the left side and CNG technology in the right side. This aspect becomes important, for example, when rationalising the fuel tank (cylinder) weight contribution to total system installed weight for a particular fuel system (CNG or LNG) configuration. Given the similarity of lower level component weights and fuel system architectures, it can be seen that the fuel tank (cylinder) comprises the major differentiator to natural gas fuel modification weight. Therefore, natural gas tank (cylinder) weight is used as a metric used in the quantisation analysis as described later.




1. Cryogenic tank. -120ºC at 11 bar.
2. Heat exchanger LNG (in some cases it is inside the tank).
3. Pressure regulator. Output at 5-8 bar.
4. NG injectors.
5. High pressure tank. CNG at 200 bar.
6. Pressure regulator. Input at 200 bar. Output at 5-8 bar.
7. NG injectors.
8. ECU.

Figure 41. Comparison of LNG and CNG fuel system components
Lebrato J. et al. (2013) LNG Blue Corridors

As stated above, Table 17 also includes the morphological matrix for the natural gas ground infrastructure segment. This natural gas ground infrastructure can be configured in several ways, but fundamentally relies on an external source of natural gas provided to the fuelling station. The general process relies on natural gas being converted and stored onsite from a natural gas pipeline or provided from an external source, and then transported to the fuelling station for storage and subsequent use. The most common means of discharging natural gas is via a bowser or dispenser in much the same way that other liquid fuels are dispensed. There are several variations on this theme which is fully documented in the related quantisation analysis as described later in this Section.

Table 17. Natural gas system morphological matrix

| | Modification function | Selected solution | Alternative solutions | | | |
|---|---|---|---|---|---|---|
| Aircraft segment | NG energy state | LNG tank(s) | CNG tank(s) | LNG tank(s) | 0 | 0 |
| | NG energy storage location | NG fuselage belly tank | NG wing tip tanks | NG fuselage belly tank | NG wing tip and fuselage belly tanks | 0 |
| | AVGAS energy storage location | AVGAS wing tip and wing tanks | AVGAS wing tip tanks | AVGAS wing tanks | AVGAS wing tip and wing tanks | 0 |
| | Energy transfer type | LNG lines | CNG lines | LNG lines | 0 | 0 |
| | Propulsion system control | ECU update | ECU update | No change | 0 | 0 |
| | Energy regulation & control | LNG pressure control regulator | CNG regulators | LNG pressure control regulator | 0 | 0 |
| | Heat exchange | LNG heat exchanger/vaporisor | CNG regulator heater | LNG heat exchanger/vaporisor | 0 | 0 |
| | Energy selection mode | LNG/AVGAS valve | CNG/AVGAS valve | LNG/AVGAS valve | CNG valve | LNG valve |
| | Energy safety system | LNG vent/relief/auto cutoff valves | CNG safety valve | LNG vent/relief/auto cutoff valves | 0 | 0 |
| | Energy enable/disable | LNG valve(s) | CNG valve(s) | LNG valve(s) | 0 | 0 |
| | Energy storage state monitoring | LNG tank pressure gauge | CNG pressure gauge(s) | LNG tank pressure gauge | 0 | 0 |
| | Energy state monitoring | LNG gas temp sensor | CNG flowrate gauge(s) | LNG gas temp sensor | 0 | 0 |
| | Energy quantity monitoring | LNG quantity gauge(s) | CNG quantity gauge(s) | LNG quantity gauge(s) | 0 | 0 |
| | Energy charge/discharge interface | LNG fuel fitting/check valve/vapor shutoff | CNG fuelling valve | LNG fuel fitting/check valve/vapor shutoff | 0 | 0 |
| | | | | | | |
| Ground infrastructure and delivery segment | NG energy supply type | High N2 CNG mains supply | CNG mains supply | High N2 CNG mains supply | Biogas supply | 0 |
| | NG energy input or state conversion | CNG-LNG liquefaction | CNG compressor | LNG-CNG converter | CNG-LNG liquefaction | Cleanup and CNG-LNG liquefaction |
| | NG transportation | LNG tanker | LNG tanker | Mobile on-site CNG tanker | None | 0 |
| | NG energy storage type | LNG tank | CNG tank | LNG tank | No storage | None |
| | NG energy transfer/charge | Station bowser from tank | Station bowser from tank | Mobile on-site CNG tanker | 0 | 0 |

### 1.3.3 Quantisation of the aircraft morphological matrix

The major shortfall of the morphological matrix technique is the potentially large number of candidate solutions. The relatively small matrix in Table 17 already gives 2×3×3 = 18 possible concept combinations. Other configurations may be possible however this analysis has been restricted to natural gas fuel tank (cylinder) locations that do not impact existing payload volume, or do not require extensive modifications to wing structure. This is a design choice which limits the number of configurations in accordance with customer needs outlined in the QFD matrix.

A conventional morphological matrix that incorporates mathematical models of each solution element is referred to as a quantified morphological matrix. In this matrix framework the focus is on properties that can be quantified, such as fuel loading, fuel quantity, tank weight, aerodynamic drag. and tank cost. A quantified value of the complete product can be obtained through aggregating the attributes for each respective sub-solution. The mathematical basis underpinning this methodology is described in Chapter 4, where the approach involves identifying the concept solution with the lowest tank weight (*W*), tank installation drag (*D*), and tank cost (*C*), whilst providing the highest possible fuel loading (*F*) (i.e. weight and quantity) solution. In this case the model used is one of simple summation to provide a Figure of Merit and also a parametric Payload-Range score as described below.

Table 18 and Table 19 show the respective quantified morphological matrices relating to the CNG and LNG fuel system modification concepts. Table 18 shows the CNG and CNG/AVGAS bi-fuel concepts and Table 19 shows the LNG and LNG/AVGAS bi-fuel concepts. Although it is usual to present this information in one matrix, it has been separated here to facilitate presentation in this thesis. The upper rows of the matrices provide configuration details of each concept which are determined from the morphological matrices as described earlier. These rows present each energy storage location corresponding to the configuration concept along with an accompanying diagram which represents the utilisation and location of each tank on the aircraft. In this case the yellow highlighted regions under the fuselage on these aircraft diagrams depict utilisation of existing AVGAS fuel tanks. While the black highlighted regions depict the natural gas fuel tanks forming part of the modification. These diagrams are intended to provide a quick visual means to define each configuration concept.

#### 1.3.3.1 Metrics

Rather than attempting to represent each metric as an objective function comprising all system components as described Gavel et al. (2008) the metrics as described in Table 18 and Table 19 have been simplified based on QFD requirements as described in Section 1.2. Here each requirement attribute is represented as a metric and is modelled to approximate the properties of the complete aircraft segment. These metrics equate to the natural gas modification weight, aerodynamic drag and cost impacts, with these quantities either directly or indirectly derived from the QFD matrix. In this instance weight is further broken down into fuel quantity and weight, in addition to the structural weight of the natural gas fuel tank installation. It should be noted that the QFD engineering attributes comprising *Power impact* and *Size Impact* are implicit in the design concept selection. That is *Power impact* will be a function of the implementation of Engine Control Unit (ECU) and changes to the engine such as ignition timing or turbocharger boost, as/if required. Size Impact is also dependent on conceptual design selection where potential natural gas fuel tanks are located to ensure no impact on passenger, baggage, or cargo space. Therefore, given that these attributes are implicit in concept selection these are not quantified here as metrics.

#### 1.3.3.2 Normalisation of metrics

This quantisation is based on the metrics as shown in the left most column of Table 18 and Table 19. These metrics are grouped into blocks which presents the data relevant to each metric such as itemised weights or fuel quantities, weight removed/added, drag increments or costs. The final score relating to each metric, shown in **<u>bold underline</u>**, is then determined and shows the change impact or increment. This score is expressed as a ratio in case of fuel weight or quantity or is normalised to the configuration baseline. The configuration baseline used to normalise tank weight, installed drag and tank cost metrics is Configuration 3, shown in Table 18, which equates to a natural gas wing tip fuel only concept. This same Configuration 3 baseline is used also for LNG configuration concepts shown in Table 19. Therefore, all scores are normalised to this baseline.

#### 1.3.3.3 Metric weighting

Each metric is weighted to account for importance in accordance with the weightings derived in the QFD matrix shown in Table 10. These weightings are shown on the left-hand side of the matrices shown in Table 18 and Table 19 and have been assigned to fuel weight, fuel quantity, tank weight, installed drag and tank cost respectively. Note that the weighting applied to the weight metric has been split equally between fuel weight and tank weight, with 10% assigned to each respectively. It should be noted that this weighting has focussed on the three highest rated engineering challenges as shown in the QFD matrix as described earlier.

#### *1.3.3.4 Fuel weight*

The total fuel weight metric is determined by a simple fuel accounting procedure that compares the resultant fuel loading with the existing aircraft flight manual published limits. It is important to note that this modification therefore aims to ensure fuel loading is maintained within existing published aircraft flight manual limits. For example, the total fuel weight comprising natural gas and AVGAS is determined for each concept and compared with the maximum allowable fuel weight as published in the flight manual which is 1020 lbs (463 kg). This metric is then expressed as a fuel weight ratio.

#### *1.3.3.5 Fuel quantity*

The total fuel quantity metric is determined by a simple fuel accounting procedure that compares the resultant with the maximum allowable fuel quantity. It should be noted however that the resultant fuel loading may vary considerably for each configuration as a function of AVGAS fuel tank utilisation as seen in Table 18 and Table 19. For example, the fuel loading of CNG only and LNG only solutions (i.e. non-bi-fuel) are limited without the utilisation of existing AVGAS tanks. This in turn limits the range performance of these concepts.

#### *1.3.3.6 Tank weight*

In the case of weight, the representative weights of natural gas fuel tanks (otherwise known as cylinders) are determined using the empirical relationships described in Chapter Appendix 2. The impact of change in tank weight for a particular tank configuration is accounted for using traditional Weight and Balance (W&B) approaches. For example, two main configurations are considered as part of this modification being the wing tip tank replacement and/or the addition of the fuselage belly tank. These tank weight metrics are accounted for as a normalised score compared to the baseline as described above.

#### 1.3.3.7 Tank installed drag

The impact of additional drag resulting from the modification is determined by the drag contribution of the natural gas fuel tank. Fuel tank drag is determined by inspection to be the main contributor to drag for this modification. The fuel tank drag contributions are determined from an analysis which accounts for fuel tank location (wing tip mounted or under fuselage mounted) using the approximations as described in Chapter Appendix 2. Again, the drag contribution metric is accounted for as a normalised score compared to the baseline as described above.

#### 1.3.3.8 Tank costs

Installation cost is represented by the cost of the natural gas fuel tank. Fuel tank cost is determined by inspection to be the most expensive contributor to the natural gas segment costs, as this component is likely to be a custom developed solution. This is in contrast to other fuel system components which will be variant designs (valves, regulators, fuel lines, heat exchangers) or standard aeronautical solutions (instrument displays). Fuel tank cost is determined using the empirical relationships as described in Appendix 2. Again, the cost contribution metric is accounted for as a normalised score compared to the baseline as described above.

It should be also noted that optimisation of natural gas fuel tank sizing is not undertaken in this case study. Rather fuel tank sizing has been determined by either matching the size of the existing wing tip tanks, or by sizing of the belly tank to provide the required ground clearance and undercarriage door clearance. This approach is adequate for conceptual design as tank sizing optimisation is traditionally an early preliminary design activity.

#### 1.3.3.9 Payload-range score

In order to encapsulate flight performance associated with this quantisation analysis, an additional measure has been formulated that combines payload and range characteristics. The range parameter is determined from the combination of normalised scores for fuel weight and quantity, which is analogous to range capability. The payload parameter is determined from simple accounting of aircraft takeoff weight less empty weight, fuel weight and the natural gas tank weight. This payload parameter is then normalised using the Configuration 3 baseline as described above. It is important to note that the payload range measure does not incorporate cost. Therefore, the payload range measure will not directly align with the Figure of Merit results described below.

#### *1.3.3.10 Figure of Merit*

As described above, these metrics are quantised by use of simple mathematical models allowing an approximation of the complete aircraft segment modification. These metrics can then be evaluated by a Figure of Merit (*FoM*) which is established on a relationship that accounts for a requirement to minimise or maximise these quantities respectively. This approach involves identifying the concept solution with the lowest tank weight, tank installation drag, and tank cost, whilst providing the highest possible fuel loading (i.e. fuel weight and quantity) solution.  This *FoM* is formulated to provide a ranking score using these requirements and is reflected in Table 18 and Table 19 as follows:

$$FoM = \frac{W_{fuel}}{W_{score}+D_{score}+C_{score}} \qquad \text{Equation 21}$$

Where

$W_{fuel}$ – Normalised score - fuel weight of each concept solution.

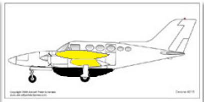
$W_{score}$ – Normalised score - NG fuel tank weight for each concept solution.

$D_{score}$ – Normalised score - NG fuel tank drag for each concept solution.

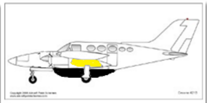
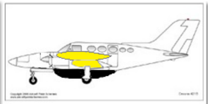
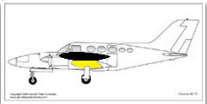
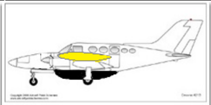
$C_{score}$ – Normalised score - NG fuel tank cost for each concept solution.

Based on this *FoM* relationship the highest-ranking solution can be determined and compared for over the aircraft segment solution space. The subsequent approach therefore can select the best solution or solutions to in order to conduct a Pugh pairwise comparison analysis used to confirm the results of this morphological analysis.

## Table 18. CNG & CNG/AVGAS bi-fuel quantisation

| | | CNG and CNG/AVGAS bi-fuel options | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|
| **Sub -system function** | **Weighting** | **Config 1** | **Config 2** | **Config 3** | **Config 4** | **Config 5** | **Config 6** | **Config 7** | **Config 8** |
| Energy storage location 1 | | CNG wing tip tanks | CNG wing tip tanks | CNG wing tip tanks | None | None | None | CNG wing tip tanks | None |
| Energy storage location 2 | | CNG fuselage belly tank | CNG fuselage belly tank | None | CNG fuselage belly tank | CNG fuselage belly tank | CNG fuselage belly tank | None | CNG fuselage belly tank |
| Existing energy storage 1 AVGAS tanks | | AVGAS wing tank | None | None | None | AVGAS wing tank | AVGAS wing tank | AVGAS wing tank | None |
| Existing energy storage 2 AVGAS tanks | | None | None | None | None | 0.679 | AVGAS wing tip tank | None | AVGAS wing tip tank |
| | |  |  |  |  |  |  |  |  |
| | | **BI-FUEL** | **CNG** | **CNG** | **CNG** | **BI-FUEL** | **BI-FUEL** | **BI-FUEL** | **BI-FUEL** |
| **Fuel weight** | | | | | | | | | |
| Wing tip tank fuel (lbs) | | 308.3 | 308.3 | 308.3 | 0.0 | 0.0 | 600.0 | 308.3 | 600.0 |
| Fuselage belly tank fuel (lbs) | | 117.3 | 117.3 | 0.0 | 117.3 | 117.3 | 117.3 | 0.0 | 117.3 |
| AVGAS wing tanks (lbs) | | 420.0 | 0.0 | 0.0 | 0.0 | 420.0 | 300.0 | 420.0 | 0.0 |
| Total fuel load (lbs) | | **845.6** | **425.6** | **308.3** | **117.3** | **537.3** | **1017.3** | **728.3** | **717.3** |
| Aircraft max AVGAS fuel load (lbs) | | 1020.0 | 1020.0 | 1020.0 | 1020.0 | 1020.0 | 1020.0 | 1020.0 | 1020.0 |
| Normalised score - fuel weight | 1.10 | **0.91** | **0.46** | **0.33** | **0.13** | **0.58** | **1.10** | **0.79** | **0.77** |
| **Tank weight** | | | | | | | | | |
| Wing tip tank (lbs) - Added | | 385.9 | 385.9 | 385.9 | 0.0 | 0.0 | 0.0 | 385.9 | 0.0 |
| Wing tip tank (lbs) - Removed | | -73.0 | -73.0 | -73.0 | 0.0 | 0.0 | 0.0 | -73.0 | 0.0 |
| Wing tip tank AVGAS fuel load + tank struct (lbs) | | 673.0 | 673.0 | 673.0 | 0.0 | 0.0 | 0.0 | 673.0 | 0.0 |
| Wing tip tank NG fuel load + tank struct (lbs) | | 694.2 | 694.2 | 694.2 | 0.0 | 0.0 | 0.0 | 694.2 | 0.0 |
| Net wing tip weight (lbs) - (2 tanks) | | **21.2** | **21.2** | **21.2** | **0.0** | **0.0** | **0.0** | **21.2** | **0.0** |
| Fuselage belly tank (lbs) | | 146.8 | 146.8 | 0.0 | 146.8 | 146.8 | 146.8 | 0.0 | 146.8 |
| Net structural weight added (lbs) | | **459.7** | **459.7** | **312.9** | **146.8** | **146.8** | **146.8** | **312.9** | **146.8** |
| Normalised score - tank structural weight | 1.10 | **1.62** | **1.62** | **1.10** | **0.52** | **0.52** | **0.52** | **1.10** | **0.52** |
| **Installed drag** | | | | | | | | | |
| Wing tip fuel tank drag coefficient - Added | | 0.0026 | 0.0026 | 0.0026 | 0.0000 | 0.0000 | 0.0000 | 0.0026 | 0.0000 |
| Wing tip fuel tank drag coefficient - Removed | | -0.0005 | -0.0005 | -0.0005 | 0.0000 | 0.0000 | 0.0000 | -0.0005 | 0.0000 |
| Fuselage belly tank drag coefficient - Added | | 0.0012 | 0.0012 | 0.0000 | 0.0012 | 0.0012 | 0.0012 | 0.0000 | 0.0012 |
| Net drag count (based on wing ref area) | | **3.2** | **3.2** | **2.1** | **1.2** | **1.2** | **1.2** | **2.1** | **1.2** |
| Normalised score - installed drag | 1.19 | **1.88** | **1.88** | **1.19** | **0.69** | **0.69** | **0.69** | **1.19** | **0.69** |
| **Tank cost estimate** | | | | | | | | | |
| Wing tip tanks ($) | | $10,418 | $10,418 | $10,418 | $0 | $0 | $0 | $10,418 | $0 |
| Fuselage belly tank ($) | | $3,963 | $3,963 | $0 | $3,963 | $3,963 | $3,963 | $0 | $3,963 |
| Net cost (tank hardware only) | | **$14,381** | **$14,381** | **$10,418** | **$3,963** | **$3,963** | **$3,963** | **$10,418** | **$3,963** |
| Normalised score - tank hardware cost | 1.25 | **1.73** | **1.73** | **1.25** | **0.48** | **0.48** | **0.48** | **1.25** | **0.48** |
| **Normalised Payload-range** | | | | | | | | | |
| Payload (lbs) | | 1735 | 2155 | 2272 | 2485 | 2065 | 1585 | 1852 | 1885 |
| Payload normalised score | | 0.76 | 0.95 | 1.00 | 1.09 | 0.91 | 0.70 | 0.82 | 0.83 |
| Fuel - range normalised score | | 0.91 | 0.46 | 0.33 | 0.13 | 0.58 | 1.10 | 0.79 | 0.77 |
| **Payload-range score** | | **0.696** | 0.435 | 0.333 | 0.138 | 0.526 | **0.765** | 0.640 | **0.642** |
| | | | | | | | | | |
| **Figure of Merit** | | 0.175 | 0.088 | 0.094 | 0.075 | **0.344** | **0.651** | 0.222 | **0.459** |

## Table 19. LNG & LNG/AVGAS bi-fuel quantisation

| | | LNG and LNG/AVGAS bi-fuel options | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|
| **Sub -system function** | **Weighting** | **Config 9** | **Config 10** | **Config 11** | **Config 12** | **Config 13** | **Config 14** | **Config 15** | **Config 16** |
| Energy storage location 1 | | LNG wing tip tanks | LNG wing tip tanks | LNG wing tip tanks | None | None | None | LNG wing tip tanks | None |
| Energy storage location 2 | | LNG fuselage belly tank | LNG fuselage belly tank | None | LNG fuselage belly tank | LNG fuselage belly tank | LNG fuselage belly tank | None | LNG fuselage belly tank |
| Existing energy storage 1 AVGAS tanks | | AVGAS wing tank | None | None | None | AVGAS wing tank | AVGAS wing tank | AVGAS wing tank | None |
| Existing energy storage 2 AVGAS tanks | | None | None | None | None | None | AVGAS wing tip tank | None | AVGAS wing tip tank |
| | |  |  |  |  |  |  |  |  |
| | | **BI-FUEL** | **LNG** | **LNG** | **LNG** | **BI-FUEL** | **BI-FUEL** | **BI-FUEL** | **BI-FUEL** |
| **Fuel weight** | | | | | | | | | |
| Wing tip tank fuel (lbs) | | 561.8 | 561.8 | 561.8 | 0.0 | 0.0 | 600.0 | 561.8 | 600.0 |
| Fuselage belly tank fuel (lbs) | | 230.3 | 230.3 | 0.0 | 230.3 | 230.3 | 230.3 | 0.0 | 230.3 |
| AVGAS wing tanks (lbs) | | 228.0 | 0.0 | 0.0 | 0.0 | 420.0 | 190.0 | 420.0 | 0.0 |
| Total fuel load (lbs) | | **1020.1** | **792.1** | **561.8** | **230.3** | **650.3** | **1020.3** | **981.8** | **830.3** |
| Aircraft max AVGAS fuel load (lbs) | | 1020 | 1020 | 1020 | 1020 | 1020 | 1020 | 1020 | 1020 |
| Normalised score - fuel weight | 1.10 | **1.10** | **0.85** | **0.61** | **0.25** | **0.70** | **1.10** | **1.06** | **0.90** |
| **Tank weight** | | | | | | | | | |
| Wing tip tank (lbs) - Added | | 159.6 | 159.6 | 159.6 | 0.0 | 0.0 | 0.0 | 159.6 | 0.0 |
| Wing tip tank (lbs) - Removed | | -73.0 | -73.0 | -73.0 | 0.0 | 0.0 | 0.0 | -73.0 | 0.0 |
| Wing tip tank AVGAS fuel load + tank struct (lbs) | | 673.0 | 673.0 | 673.0 | 0.0 | 0.0 | 0.0 | 673.0 | 0.0 |
| Wing tip tank NG fuel load + tank struct (lbs) | | 721.4 | 721.4 | 721.4 | 0.0 | 0.0 | 0.0 | 721.4 | 0.0 |
| Net wing tip weight (lbs) - (2 tanks) | | **48.4** | **48.4** | **48.4** | **0.0** | **0.0** | **0.0** | **48.4** | **0.0** |
| Fuselage belly tank (lbs) | | 65.4 | 65.4 | 0.0 | 65.4 | 65.4 | 65.4 | 0.0 | 65.4 |
| Net structural weight added (lbs) | | 1594.9 | 1594.9 | 1529.5 | 65.4 | 65.4 | 65.4 | 1529.5 | 65.4 |
| Normalised score - tank structural weight | 1.10 | **5.61** | **5.61** | **5.38** | **0.23** | **0.23** | **0.23** | **5.38** | **0.23** |
| **Installed drag** | | | | | | | | | |
| Wing tip fuel tank drag coefficient - Added | | 0.0028 | 0.0028 | 0.0028 | 0.0000 | 0.0000 | 0.0000 | 0.0028 | 0.0000 |
| Wing tip fuel tank drag coefficient - Removed | | -0.0005 | -0.0005 | -0.0005 | 0.0000 | 0.0000 | 0.0000 | -0.0005 | 0.0000 |
| Fuselage belly tank drag coefficient - Added | | 0.0012 | 0.0012 | 0.0000 | 0.0012 | 0.0012 | 0.0012 | 0.0000 | 0.0012 |
| Net drag count (based on wing ref area) | | **3.5** | **3.5** | **2.3** | **1.2** | **1.2** | **1.2** | **2.3** | **1.2** |
| Normalised score - installed drag | 1.19 | **2.01** | **2.01** | **1.32** | **0.69** | **0.69** | **0.69** | **1.32** | **0.69** |
| **Tank cost estimate** | | | | | | | | | |
| Wing tip tanks ($) | | $14,195 | $14,195 | $14,195 | $0 | $0 | $0 | $14,195 | $0 |
| Fuselage belly tank ($) | | $5,818 | $5,818 | $0 | $5,818 | $5,818 | $5,818 | $0 | $5,818 |
| Net cost (tank hardware only) | | $20,013 | $20,013 | $14,195 | $5,818 | $5,818 | $5,818 | $14,195 | $5,818 |
| Normalised score - tank hardware cost | 1.25 | **2.40** | **2.40** | **1.70** | **0.70** | **0.70** | **0.70** | **1.70** | **0.70** |
| **Normalised Payload-range** | | | | | | | | | |
| Payload (lbs) | | 1533 | 1761 | 1992 | **2372** | 1952 | 1582 | 1572 | 1772 |
| Payload normalised score | | 0.67 | 0.78 | 0.88 | 1.04 | 0.86 | 0.70 | 0.69 | 0.78 |
| Fuel-range normalised score | | 1.10 | 0.85 | 0.61 | 0.25 | 0.70 | 1.10 | 1.06 | 0.90 |
| **Payload-range score** | | **0.742** | 0.662 | 0.531 | 0.259 | 0.602 | **0.766** | 0.732 | **0.698** |
| | | | | | | | | | |
| **Figure of Merit** | | 0.110 | 0.085 | 0.072 | 0.153 | 0.433 | **0.679** | 0.126 | 0.552 |

### 1.3.4 Aircraft segment results

The lower two rows shown in Table 18 and Table 19 provide Payload-Range (*P-R*) score and Figure of Merit (*FoM*) results for all natural gas concept solutions. The highest scoring values for *P-R* and *FoM* are highlighted green, while other higher ranked scores are highlighted grey.

As discussed earlier, the *P-R* score is a measure of the payload-range capability excluding cost as a metric. While the *FoM* includes all metrics including cost. Therefore, the *FoM* is the appropriate means to determine those concept(s) that best meet requirements. However, *P-R* score is presented here as it also provides a direct indicator of aircraft performance.

The five highest scoring *FoMs* are listed in Table 20 based on results as highlighted in Table 18 and Table 19. All are bi-fuel configuration concepts resulting from the higher energy storage density capacity of this combined fuel type.

Table 20. Ranked Figure of Merit scores

| Ranking | Configuration | Figure of Merit | Description |
|---|---|---|---|
| 1 | 14 | 0.679 | Bi-fuel configuration comprising a LNG belly tank with AVGAS wingtip and wing tanks |
| 2 | 6 | 0.651 | Bi-fuel configuration comprising a CNG belly tank with AVGAS wingtip and wing tanks |
| 3 | 16 | 0.552 | Is a variation on Configuration 14 using the AVGAS wingtip tanks only |
| 4 | 8 | 0.459 | Is a variation on Configuration 6 using the AVGAS wingtip tanks only |
| 5 | 13 | 0.433 | Is a variation on Configuration 14 using AVGAS wing tanks only |

Note that Configurations 14, 16 and 13 are variations on the LNG belly tank design concept with utilisation of different AVGAS tanks. Configuration 9 incorporates both LNG belly tank and wingtip tanks with AVGAS wing tanks. And lastly Configurations 6 and 8 are variations on the CNG belly tank design concept with utilisation of different AVGAS tanks.

The three highest Payload-Range scores are listed in Table 21 based on results as highlighted in Table 18 and Table 19. Note that Configuration 9 has a slightly higher P-R score as compared to Configuration 14. This was a function of AVGAS fuel quantity carried in these configurations. Configuration 9 carries a higher quantity of LNG (and hence higher fuel quantity) for the same maximum fuel weight.

Table 21. Ranked Range-Payload scores

| Ranking | Configuration | Payload-Range | Description |
|---|---|---|---|
| 1 | 14 | 0.766 | Bi-fuel configuration consisting of a LNG belly tank with AVGAS wingtip and wing tanks |
| 2 | 6 | 0.765 | Bi-fuel configuration consisting of a CNG belly tank with AVGAS wingtip and wing tanks |
| 3 | 9 | 0.742 | Bi-fuel configuration consisting of LNG belly tank and LNG wingtip tanks using AVGAS wing tank only |

Based on the ranking results shown in Table 20 and Table 21, Configurations 14 and 6 best meet requirements, with other configurations also exhibiting favourable performance.

### 1.3.5 Quantisation of the fuelling station morphological matrix

Table 22 shows the quantified morphological matrices relating to the natural gas ground infrastructure solution space. In a similar format as described earlier, the upper rows of these matrices provide configuration details of each concept which are determined from the corresponding ground infrastructure morphological matrix as described earlier. For example, these rows present each natural gas energy supply, conversion, delivery, storage and dispensing option for each configuration concept along with an accompanying diagram which represents the particular setup. These diagrams are intended to provide a quick visual means to define each configuration concept.

Table 22 incorporates infrastructure costs and storage and fill time metrics to approximate the main attributes of the ground infrastructure segment. These quantities are either directly or indirectly derived from the corresponding ground infrastructure QFD matrix shown in quantisation and normalisation of metrics

As described earlier, quantisation is based on the metrics as shown in the left most column of Table 22. These metrics are grouped into blocks which present the data relevant to each metric given by fuelling infrastructure costs and the related performance metrics relating to fuel storage capacity and fill time. The final score relating to each metric, shown in **<u>bold underline</u>**, is then determined which shows the change impact or increment. This is normalised to the configuration baseline, or as a direct score in the case or storage and fill characteristics.

#### *1.3.5.1 Metrics weighting*

Each metric is weighted to account for importance in accordance with those weightings derived in the QFD matrix shown by Table 14. These weightings are shown on the left-hand side of the matrices shown in Table 22, and have been assigned to costs and storage and fill metrics respectively but are unused for this analysis.

#### *1.3.5.2 Infrastructure costs*

Natural gas fuelling station infrastructure and operating concepts are dependent on the implementation of fuel state (whether CNG or LNG). In general, CNG fuel is produced from the local natural gas mains supply and is compressed and stored in a cylinder until ready for dispensing. CNG is supplied in this way does not incur transportation costs. Whereas LNG is generally transported by heavy vehicles from a much larger liquefaction plant to the fuelling station where it is then transferred to storage cylinders until ready for dispensing. Transportation costs for LNG supplied in this way is dependent on distance travelled and hence can be a major contributor to overall supply costs.

The major cost metrics associated with natural gas refuelling infrastructure fall into two categories:

1. Station construction cost metrics which comprises the buildings, equipment and facilities
    a. For CNG this is a function of the throughput of the station and the type of fill arrangement as reported by US Department of Energy (2014).

b. For LNG this is a function of the station storage capacity as determined by a study conducted by Little (2001) and reported in TIAX (2012).

2. The delivery/production cost metrics of CNG or LNG fuels which comprises transportation of fuels to site, or the production of fuels onsite.

This thesis was based on a ground fuelling infrastructure requirement to provide sufficient fuel for 10 aircraft each at 100 US gallons (380 litres) per day with a holding capacity for 10 days as specified in the QFD matrix. This translates to the following for the CNG and LNG fuelling stations:

- CNG fuelling stations assumed a “medium station” size producing 1000 GGE/day (3800 litres/day).
- LNG fuelling stations was based on a storage capacity of 10000 gallons (37850 litres) to enable supply of 1000 GGE/day (3800 litres/day) for 10 days.

Two references were used to provide cost metrics data and details of CNG and LNG fuelling station infrastructure. The US Department of Energy (2014) report dealt with costs associated with CNG vehicle fuelling infrastructure. While the TIAX (2012) market analysis report documented costs associated with setting up LNG production facilities, fuel delivery, along with local fuelling station storage and supply facilities. Both references dealt with CNG and LNG infrastructure applied to ground transportation vehicles such as automobiles and trucks. No references could be found that dealt with supply and storage of natural gas fuels at the small scale required for General Aviation applications.

#### *1.3.5.3 Storage & fill*

Storage and fill time metrics are derived from two requirements defined in the ground fuelling infrastructure QFD matrix as follows:

1. The ground refuelling station shall be able to store the quantity of fuel required without degradation or evaporation for a period of 1 week. Unlike other CNG fuels, LNG has a finite storage time which is dependent on storage characteristics and usage rate. However, this LNG storage time limitation can be managed by controlled fuel usage and also

by good design of insulated storage cylinders. CNG can be stored indefinitely within certain limitations.

2. The ground refuelling station shall provide a refill rate of at least 200 gallons per hour. In this case both CNG and LNG fuel have the capacity to achieve this "fast fill" requirement. This could be achieved by sizing of the fuelling station cylinders and dispensing equipment accordingly.

#### *1.3.5.4 Figure of Merit*

As described earlier, the aircraft segment metrics are quantised by use of simple mathematical models allowing an approximation of the ground segment fuelling station. These metrics can then be evaluated by a *FoM* which is established on a relationship that accounts for a requirement to minimise costs and maximise storage and fill capacity respectively.

This *FoM* is formulated to provide a ranking score using these requirements and is reflected in Table 22 as follows:

$$FoM = 1 + (Storage\&fill_{score} - C_{score}) \qquad \text{Equation 22}$$

Where the following definitions apply:

*Storage & fill*$_{score}$ - Normalised score for storage & fill capability.

$C_{score}$ – Normalised score for costs associated with natural gas fuel station construction and natural gas delivery and production.

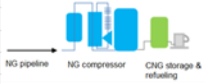
Based on this *FoM* relationship the highest-ranking solution can be determined and compared for the ground infrastructure segment solution space.

### 1.3.6 Ground infrastructure segment results

The lower row in Table 22 shows the *FoM* scores determined for all potential ground infrastructure solutions. On the basis of cost and storage and fill metrics, the NRU/GSP liquefier (Configuration 3) provides the best solution, followed closely by the baseline purpose-built liquefier (Configuration 1). Both fuelling stations rely on LNG liquefaction at a dedicated site near a natural gas source then transportation of LNG fuel via a LNG tanker. This LNG is then stored on-site at the airport in a relatively small facility which can provide for the 10-day storage hold time and a capacity of 10000 gallons (37,850 litres).

It is important to note that this case study analysis assumes that the ground infrastructure is a non-developmental segment. That is the design and development of a natural gas fuelling station is an established and mature engineering activity, and therefore does not require prototyping or extensive support and test equipment.

## Table 22. Ground segment fuelling station quantisation

| Sub -system function | Weighting | Config 1 | Config 2 | Config 3 | Config 4 | Config 5 | Config 6 | Config 7 |
|---|---|---|---|---|---|---|---|---|
| | | CNG & LNG Fuelling Infrastructure | | | | | | |
| Energy supply type | | CNG mains supply | CNG mains supply | High N2 CNG mains supply | Biogas supply | CNG mains supply | CNG mains supply | CNG mains supply |
| Energy input or state conversion | | CNG-LNG liquefaction | CNG-LNG liquefaction | CNG-LNG liquefaction | ogas cleanup and CNG-LNG liquefactio | CNG-LNG liquefaction | CNG compressor | CNG compression |
| Transportation | | LNG tanker | None | LNG tanker | LNG tanker | LNG tanker | Mobile on-site CNG tanker | |
| Energy storage type | | LNG tank | LNG tank | LNG tank | LNG tank | CNG conversion | None | CNG tank |
| Energy transfer/charge | | Station bowser from tank | Station bowser from tank | Station bowser from tank | Station bowser from tank | Station bowser from tank | Mobile on-site CNG tanker | Bowser from tank |
| | |  |  |  |  |  |  |  |
| | | Purpose built liquefier | Onsite small-scale liquefier | NRU/GSP liquefier | Bio LNG liquefier | LCNG station | Mobile on-site CNG station | Medium CNG station |
| **Costs** | | | | | | | | |
| Station costs (US$) | | $550,000 | $1,000,000 | $550,000 | $1,200,000 | | $1,200,000 | $900,000 |
| Delivery and/or prod costs (US$) | | | | | | | | |
| 1000 gallons - distance - 0 miles | | $550 | $0 | $400 | $600 | | $0 | $0 |
| 1000 gallons - distance - 300 miles | | $750 | $0 | $550 | $750 | | $0 | $0 |
| Total | | $551,300 | $1,000,000 | $550,950 | $1,201,350 | | $1,200,000 | $900,000 |
| Normalised score - Costs | 1.0 | **1.000** | **1.814** | **0.999** | **2.179** | | **2.177** | **1.633** |
| **Storage & fill** | | | | | | | | |
| Storage hold time (days) | | 10 | 10 | 10 | 10 | | 10 | 10 |
| Fill time | | Fast fill | Fast fill | Fast fill | Fast fill | | Fast fill | Fast fill |
| Normalised score - Storage & fill | 1.0 | **1.0** | **1.0** | **1.0** | **1.0** | | **1.0** | **1.0** |
| **Figure of Merit** | | 1.000 | 0.186 | 1.001 | -0.179 | 1.000 | -0.177 | 0.367 |

## 1.4 CONCEPT SELECTION VALIDATION – STEP 3

### 1.4.1 Overview

The next step in this methodology involves the application of the Pugh Matrix (PM) which is used to validate the candidate concept solution as determined from the previous morphological matrix analysis.

### 1.4.2 Pugh's decision matrices

Burge (2009) states that the Pugh decision matrix allows a number of design candidate concept solutions to be compared, leading to the best concept solution that meets a set of requirements. The advantages of this PM technique is its ability to handle a large number of decision criteria, and that it that it can applied in teams, as stated by Burge (2009).

It is also an ideal method to employ, where the independency of the design team can be used review and validate the output from the previous morphological analysis. Independent validation of the concept is important as the next step in this methodology commits further analysis effort in terms of change propagation, engineering and certification analysis.

The underpinning theory associated with the PM pairwise comparison is provided in Chapter 4, and therefore is not repeated here.

### 1.4.3 Candidate design concepts

Table 23 shows the PM associated with the six favoured candidate design concepts as determined from the previous morphological analysis step. The baseline used for these pairwise comparisons is Configuration 14 which was the highest scoring configuration concept. Note that this PM has adopted the same format for header rows by listing each candidate design concept and providing a pictorial depiction of each option for easy visualisation.

### 1.4.4 Pairwise comparison

The Configuration 14 baseline is assigned a score of satisfactory “S” against all of the requirements, which are the same as those shown by the aircraft segment QFD matrix in Table 14.

The other candidate design concepts are then compared in a pairwise fashion against Configuration 14 for each requirement using the scoring system as described by Burge (2009) and outlined as follows:

- better than the baseline a “+” is entered in the appropriate cell
- worse than the baseline a “-” is entered in the appropriate cell
- the same than the baseline a “S” is entered in the appropriate cell
- much better than the baseline a “++” is entered in the appropriate cell
- much worse than the baseline a “--” is entered in the appropriate cell

For each candidate design option, the total score can be calculated by summing the number of “+"s and “–”s. The highest ranked score is generally the winning design option. Note that weightings are not applied to the requirements at this time. Rather they will be applied when if/required in cases where competing candidates cannot be resolved.

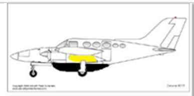
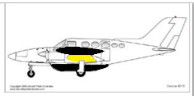
### 1.4.5 Pugh matrix results

The lower row in Table 23 shows the ranking of natural gas solution concepts using the Pugh matrix pairwise comparison method. The results of this comparison are highlighted in Table 23 with the three highest ranked solutions listed as follows:

1. Configuration 14 - Bi-fuel - LNG belly tank with AVGAS wingtip and wing tanks.
2. Configuration 6 - Bi-fuel - CNG belly tank with AVGAS wingtip and wing tanks.
3. Configuration 16 - Bi-fuel - LNG belly tank using AVGAS wingtip tanks.

The two highest ranking results given by Configurations 14 and 6 are also consistent with the two highest ranked *FoM* results determined by the earlier quantified morphological matrix analysis. On this basis Configuration 14 is selected as the preferred candidate for further analysis.

Table 23. Pugh's matrix – Natural gas modification

| Pugh Concept Selection Matrix | |  Config 14 |  Config 16 |  Config 13 |  Config 9 |  Config 6 |  Config 8 |
|---|---|---|---|---|---|---|---|
| Energy storage location 1 | Weighting | None | None | None | LNG wing tip tanks | None | None |
| Energy storage location 2 | | LNG fuselage belly tank | LNG fuselage belly tank | LNG fuselage belly tank | LNG fuselage belly tank | CNG fuselage belly tank | CNG fuselage belly tank |
| Existing energy storage 1 AVGAS tanks | | AVGAS wing tank | None | AVGAS wing tank | AVGAS wing tank | AVGAS wing tank | None |
| Existing energy storage 2 AVGAS tanks | | AVGAS wing tip tank | AVGAS wing tip tank | None | None | AVGAS wing tip tank | AVGAS wing tip tank |
| Payload-range score | | 0.766 | 0.698 | 0.602 | 0.742 | 0.765 | 0.642 |
| Figure of Merit | | 0.68 | 0.55 | 0.43 | 0.11 | 0.65 | 0.46 |
| **Selection Criteria** | | | | | | | |
| The modification shall minimise the impacts to range, payload, & cruise speed performance. | 17.4% | S | "-" | "- -" | "+" | "-" | "- -" |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 21.7% | S | S | S | "-" | S | S |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 13.0% | S | S | S | "- -" | "+" | "+" |
| The modification shall minimise emissions. | 13.0% | S | S | S | S | S | S |
| The modification shall be compatible with a range of commuter aircraft types. | 17.4% | S | S | S | "-" | S | S |
| The modification shall not impact passenger, baggage or cargo space. | 17.4% | S | S | S | S | S | S |
| | | | | | | | |
| | Total + | 0 | 0 | 0 | 1 | 1 | 1 |
| | Total - | 0 | -1 | -2 | -4 | -1 | -2 |
| | Total score | 0 | -1 | -2 | -3 | 0 | -1 |
| | Weighted total + | 0 | 0 | 0 | 0.174 | 0.130 | 0.130 |
| | Weighted total - | 0 | -0.174 | -0.348 | -0.522 | -0.174 | -0.348 |
| | **Weighted score** | **0** | **-0.174** | **-0.348** | **-0.348** | **-0.043** | **-0.217** |
| | Ranking | 1 | 3 | 5 | 5 | 2 | 4 |

## 1.5 ASSESSMENT OF PROPAGATION EFFECTS AND CHANGE OPTIONS – STEP 4

### 1.5.1 Overview

The assessment of change propagation effects and change options can be conducted at any stage of the design process. However here it is applied as an intermediate step of this conceptual design methodology. As discussed in Chapter 4 of this thesis, the conduct of this step could occur before the evaluation of the design change impacts. However, it is conducted after this juncture to provide inputs to the engineering assessment step. In reality, these propagation effect design synthesis activities would be conducted as parallel iterative processes.

This assessment of propagation effects and change options are described in the paper by Koh et al. (2012). The method builds on the QFD matrix, which is an early step in applied in this methodology and precedes the more detailed change prediction analysis. The underpinning theory associated with this method is described in Chapter 4 with details of each of the four steps required for implementation provided in Chapter 5.

### 1.5.2 Rating change options and interactions

Performance ratings are assigned to the change options in this first step (e.g. 'reduce LNG installation aerodynamic drag'). This step will indicate how well the change options will perform in addressing such requirements as 'long range' and 'low life cycle cost'. Like other steps in this conceptual design methodology, supporting information can be acquired through design team reviews, or from design records.

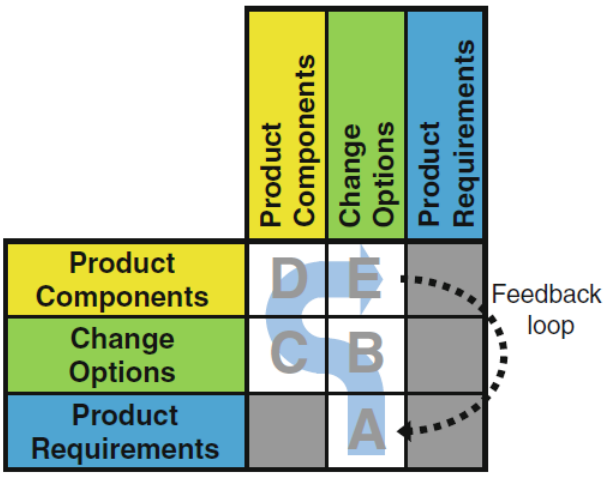



Figure 42. Modelling method steps using Multi-Domain Matrices
Koh et al. (2012)

Figure 42 shows several fields of the Multiple-Domain Matrix (MDM) which are used to assess propagation effects and change options. This first step involves Field A and Field B of the MDM which are used to rate the change options in relation to the product requirements. As described by Koh et al. (2012) the rating scale used is bipolar in nature. For example, a positive value is assigned to the appropriate element if decreasing aerodynamic drag can significantly increase aircraft range. Whereas, a negative value will be assigned to the change option, if the attribute causes a decrease in range. If a change in aerodynamic drag does not have any influence over the range or cruise speed, a '0' entry is assigned. This approach is similar to the Pugh pairwise decision matrix, as described by Burge (2009). Field A is assigned a '-5' to '5' bipolar rating scale which is consistent with the approach applied by Koh et al. (2012). This field incorporates the constraints implicit in the design and describes how the change options will be impacted if the related parameters interact.

Field B accounts for the interactions between the parameters involved in different change options, and like Field A, is bipolar in nature. In this case, if the implementation of the 'reduce LNG fuel modification LCC cost' change option conflicts with the 'reduce LNG installation drag' change option, a negative rating will be assigned in Field B. Conversely, if the change options are complementary, a positive interaction rating will be assigned. Lastly, if both of the change options are not related (i.e. no interaction), then the element will be assigned as '0', or left blank. Unlike Field A, the rating scale used in Field B range is assigned negative and positive unity values from '-1' to '1'. This represents the conditions where the change options 'totally conflict each other' and 'totally complement each other', respectively.

### 1.5.3 Change options to product components for change propagation analysis

Figure 42 shows Field C and Field D of the MDM which are used to characterise how change options are linked to relevant product components for change propagation analysis. Field C elements are binary in nature, with value of '1' assigned to the appropriate cells if a change option is related to a given product component. Conversely, if a change option is not related, a value of '0' is assigned. This binary mapping identifies the change instigating components for later change propagation analysis conducted in Field D.

### 1.5.4 Revised ratings for change options

Koh et al. (2012) fully describes the process used to support the selection of the best change option(s). Therefore, this step that revises ratings will not be presented in entirety in this thesis. Rather the approach here is to reference this work and to summarise the major steps.

This step links the combined change propagation likelihood of the impacted product components back to the relevant change options in Field E as shown in Figure 42. A value of ‘1’ is assigned to the appropriate elements if a change option is related to a given product component. Conversely, if a change option is not related to a given product component, a value of ‘0’ is assigned.

Field D is determined by the design team by stepping through all elements along each column in this asymmetric matrix. The input values to Field D are not bipolar, and therefore range from ‘0’ (no interaction) to ‘1’ (strong interaction). By using the change propagation method, the combined change propagation likelihood can be determined as described by Koh et al. (2012). This change propagation analysis is undertaken in Field D using the same technique as described by Clarkson et al. (2004). Indeed, the CPM dependency matrix described here in Field D is also analysed later in this section as part of a more detailed change propagation analysis. Therefore, the processes involved in this analysis will not be described here.

The information determined in Field A to Field E of this MDM is then used to support the prediction of change propagation effects. For example, a ‘reduce LNG installation drag’ change option describes how the ‘drag’ parameter will be impacted. This addresses the product requirement of ‘range’ and involves changes to the ‘LNG fuselage belly tank’ component. However, if a change is made to the ‘LNG fuselage belly tank’ component, then this is likely to cause a change in the ‘LNG fuel lines’ component. Furthermore, parameters related to the ‘LNG fuel lines’ component are likely to be impacted as well.

Subsequently, the performance ratings for all change options can be revised to better reflect the change propagation effects. This is carried out using the relationship as follows, described by Koh et al., (2012):

$$A^{*}_{x,j} = A_{x,j} + \sum_{k=1}^{n}[L_{k,j} \times B_{k,j} \times A_{x,k}] \quad \text{Equation 23}$$

The application of this equation and the methodology employed in its implementation are not discussed here. However full details can be found in the paper by Koh et al. (2012).

Table 24. Natural gas system change dependencies MDM

| | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Decrease LNG installation drag | Decrease LNG installation weight | Reduce LNG fuel system related LCC | Maintain engine power output | Maintain aircraft payload volume and space capacity | The modification shall minimise the impacts to range, payload, & cruise speed performance. | The modification shall comply with airworthiness standards and minimise impact to the TC basis. | The modification lifecycle costs (LCC) shall be kept to a minimum | The modification shall minimise emissions. | The modification shall be compatible with a range of commuter aircraft types. | The modification shall not impact passenger, baggage or cargo space. |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| LNG fuselage belly tank | 1 | 0.4 | 0 | 0.09 | 0 | 0 | 0.3 | 0.1 | 0.1 | 0 | 0.1 | 0.2 | 1 | 1 | | | | | | | | | |
| LNG fuel lines | 0.5 | 1 | 0.1 | 0.5 | 0.2 | 0.4 | 0.5 | 0.4 | 0 | 0.1 | 0 | 0.1 | | 1 | | | | | | | | | |
| ECU | 0 | 0 | 1 | 0.4 | 0.4 | 0 | 0.5 | 0 | 0.1 | 0.4 | 0.1 | 0 | | 1 | 1 | 1 | | | | | | | |
| LNG pressure control regulator | 0.1 | 0.1 | 0.1 | 1 | 0 | 0 | 0.1 | 0.1 | 0 | 0 | 0 | 0.1 | | 1 | | | | | | | | | |
| LNG heat exchanger/vaporisor | 0 | 0.1 | 0 | 0.1 | 1 | 0 | 0 | 0 | 0 | 0.1 | 0 | 0 | | 1 | | 1 | | | | | | | |
| LNG/AVGAS valve | 0 | 0.1 | 0 | 0 | 0 | 1 | 0.1 | 0 | 0 | 0 | 0 | 0.0 | | 1 | | 1 | | | | | | | |
| (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | 0.4 | 0.1 | 0.5 | 0.4 | 0.1 | 0.1 | 1 | 0.1 | 0 | 0 | 0 | 0.1 | | 1 | | | | | | | | | |
| LNG fuel shutoff valve | 0.1 | 0.1 | 0 | 0 | 0 | 0.1 | 0 | 1 | 0 | 0 | 0 | 0 | | 1 | | | | | | | | | |
| LNG tank pressure gauge | 0.1 | 0 | 0 | 0.1 | 0 | 0 | 0.1 | 0 | 1 | 0 | 0 | 0 | | 1 | | 1 | | | | | | | |
| LNG gas temperature sensor | 0 | 0 | 0.4 | | 0.1 | 0 | 0 | 0 | 0 | 1 | 0 | 0 | | 1 | | 1 | | | | | | | |
| LNG fuel quantity gauge | 0.3 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 1 | 0 | | 1 | | 1 | | | | | | | |
| (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | 0.1 | 0.1 | 0 | 0 | 0 | 0 | 0.1 | 0.1 | 0 | 0 | 0 | 1 | | 1 | | | | | | | | | |
| Decrease LNG installation drag | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0.8 | -0.4 | 0 | -0.2 | | | | | | |
| Decrease LNG installation weight | 1 | 0 | 0 | 1 | 0 | 0 | 1 | 1 | 0 | 0 | 0 | 1 | 0.8 | 0 | -0.4 | 0 | -0.4 | | | | | | |
| Reduce LNG fuel system related LCC | 1 | 0 | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | -0.8 | -0.6 | 0 | -0.4 | -0.4 | | | | | | |
| Maintain engine power output | 1 | 0 | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | -0.6 | 0 | 0 | | | | | | |
| Maintain aircraft payload volume and space capacity | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0.2 | 0.6 | 0.2 | 0 | 0 | | | | | | |
| The modification shall minimise the impacts to range, payload, & cruise speed performance. | | | | | | | | | | | | | 5 | 5 | 4 | 4 | 3 | | | | | | |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | | | | | | | | | | | | | 1 | 1 | 2 | 0 | 0 | | | | | | |
| The modification lifecycle costs (LCC) shall be kept to a minimum | | | | | | | | | | | | | 4 | 4 | 5 | 3 | 3 | | | | | | |
| The modification shall minimise emissions. | | | | | | | | | | | | | 3 | 2 | 0 | 1 | 1 | | | | | | |
| The modification shall be compatible with a range of commuter aircraft types. | | | | | | | | | | | | | 1 | 1 | 3 | 1 | 1 | | | | | | |
| The modification shall not impact passenger, baggage or cargo space. | | | | | | | | | | | | | 1 | 1 | 2 | 1 | 5 | | | | | | |

### 1.5.5 Selection of best change option

The various change options generated in the previous section can be evaluated based on the revised performance ratings, with the best change option selected for further development. As stated by Koh et al. (2012) there are various ways to undertake this assessment, with one approach involving the selection of the change option with the best performance rating against a given modification requirement. Table 25 through Table 28 are provided as summaries of Field A of the MDM as shown in Table 24. Accordingly, these tables show the same change options and requirements which are subsequently analysed here to determine the rate the best change option for further development.

For example, it is apparent from Table 25 that the first requirement dealing with 'Minimising range, payload and cruise speed … impact' is important to the success of the modification. This corresponds to the change option 'Decrease LNG installation drag', which in this case has a value of '7.2'. Therefore, the change option 'Decrease LNG installation drag' would be the best option as it has the highest performance rating of '7.2' for the requirement 'Minimising range, payload and cruise speed …impacts'.

Another approach is to select the change option with the best overall attributes. This is achieved by comparing each column sum. The first change option comprising 'Decrease LNG installation drag' has the highest total performance rating of '21.0', indicating that it has the best overall attributes. Therefore, from an overall performance rating perspective, the first change option comprises the best solution closely followed by the second being to 'Decrease LNG installation weight' with a total performance rating of '19.0'.

Another option is to assign weightings to each row as shown in Table 26. These weightings are assigned to better reflect the importance of each product requirement. This is analogous to the use of weighted scoring as applied in the QFD matrix shown by Table 10. For example, given that the requirement to 'Comply with airworthiness standards' is important, a higher weighting is assigned to emphasise its importance. In this instance, the importance is determined to be 21.7 % of the overall product requirements, and hence the performance ratings can be adjusted as shown. Again, the change option with the best overall attributes is, 'Decrease LNG installation drag', with a score of ‘3.3, closely followed by ‘Decrease LNG installation weight’ and ‘Reduce LNG fuel system LCC’.

Table 25. Dependencies without weighting – Natural Gas Mod

| **Field A* without weighting** | Decrease LNG installation drag | Decrease LNG installation weight | Reduce LNG fuel system related LCC | Maintain engine power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall minimise the impacts to range, payload, & cruise speed performance. | 7.2 | 6.6 | 3.6 | 3.4 | 2.1 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 1.4 | 1.3 | 2.0 | -0.3 | -0.4 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 5.7 | 5.3 | 4.7 | 2.3 | 2.0 |
| The modification shall minimise emissions. | 3.9 | 2.9 | -0.1 | 1.0 | 0.9 |
| The modification shall be compatible with a range of commuter aircraft types. | 1.4 | 1.3 | 2.9 | 0.6 | 0.5 |
| The modification shall not impact passenger, baggage or cargo space. | 1.4 | 1.6 | 1.9 | 0.7 | 4.6 |
| | 21.0 | 19.0 | 15.0 | 7.7 | 9.7 |

Table 26. Dependencies with weighting – Natural Gas Mod

| **Field A* with weighting** | Decrease LNG installation drag | Decrease LNG installation weight | Reduce LNG fuel system related LCC | Maintain engine power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall minimise the impacts to range, payload, & cruise speed performance. | 1.2 | 1.1 | 0.6 | 0.6 | 0.4 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 0.3 | 0.3 | 0.4 | -0.1 | -0 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 0.7 | 0.7 | 0.6 | 0.3 | 0.3 |
| The modification shall minimise emissions. | 0.5 | 0.4 | 0.0 | 0.1 | 0.1 |
| The modification shall be compatible with a range of commuter aircraft types. | 0.2 | 0.2 | 0.5 | 0.1 | 0.1 |
| The modification shall not impact passenger, baggage or cargo space. | 0.2 | 0.3 | 0.3 | 0.1 | 0.8 |
| | 3.3 | 3.0 | 2.5 | 1.2 | 1.5 |

Another approach is to compare the simple difference between Field A and Field A*. This provides a simple summary of the change propagation effects as shown by Table 27. In this case, a positive difference indicates that the relevant attributes are affected favourably (e.g. ‘Decrease LNG installation weight’ on ‘Minimising range, payload, cruise speed…impact’). Whereas a negative difference indicates the opposite (e.g. ‘Maintain engine power output’ on ‘Minimising range, payload, cruise speed…impact’).

Table 27. Difference between Field A and A* – Natural Gas Mod

| **Simple difference between Field A and A*** | Decrease LNG installation drag | Decrease LNG installation weight | Reduce LNG fuel system related LCC | Maintain engine power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall minimise impact to range, payload, & cruise speed.. | 2.2 | 1.6 | -0.4 | -0.6 | -0.9 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 0.4 | 0.3 | 0.0 | -0.3 | -0.4 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 1.7 | 1.3 | -0.3 | -0.7 | -1.0 |
| The modification shall minimise emissions. | 0.9 | 0.9 | -0.1 | 0.0 | -0.1 |
| The modification shall be applicable to a range of commuter aircraft types. | 0.4 | 0.3 | -0.1 | -0.4 | -0.5 |
| The modification shall not impact passenger, baggage or cargo space. | 0.4 | 0.6 | -0.1 | -0.3 | -0.4 |
| | 6.0 | 5.0 | -1.0 | -2.3 | -3.3 |

Change options sensitive to change propagation can be analysed by computing the absolute difference between Field A and A* shown by Table 28. For example, change option 'Decrease LNG installation drag' is the change option most sensitive to the requirement 'Minimising range, payload, cruise speed…impact' as it exhibits the greatest absolute difference along the first row of the matrix (i.e. an absolute difference of '2.2'). Conversely, the change option 'Decrease LNG installation drag' is the most sensitive change option overall as it has the greatest absolute total difference.

It should be noted that the above analysis does not attempt to provide a means to alleviate the design team from their responsibility for decision-making. Instead, this analysis provides support the design team by highlighting change propagation effects using a structured process. Koh et al. (2012) states that, even though change propagation is a necessary part of the modification process, unplanned change propagation may adversely affect the development of the modification. Hence, the implementation of change options should always be subject to a properly managed process even if the analysis initially predicts a favourable change propagation impact.

Table 28. Absolute difference between Field A and A* – Natural Gas Mod

| **Absolute difference between Field A and A*** | Decrease LNG installation drag | Decrease LNG installation weight | Reduce LNG fuel system related LCC | Maintain engine power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall minimise the impacts to range, payload, & cruise speed performance. | 2.2 | 1.6 | 0.4 | 0.6 | 0.9 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 0.4 | 0.3 | 0.0 | 0.3 | 0.4 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 1.7 | 1.3 | 0.3 | 0.7 | 1.0 |
| The modification shall minimise emissions. | 0.9 | 0.9 | 0.1 | 0.0 | 0.1 |
| The modification shall be compatible with a range of commuter aircraft types. | 0.4 | 0.3 | 0.1 | 0.4 | 0.5 |
| The modification shall not impact passenger, baggage or cargo space. | 0.4 | 0.6 | 0.1 | 0.3 | 0.4 |
| | 6.0 | 5.0 | 1.0 | 2.3 | 3.3 |

## 1.6 EVALUATION OF DESIGN CHANGE IMPACTS – STEP 5

### 1.6.1 Overview

A project that modifies or retrofits an aircraft with a new sub-system or component will invoke changes. Indeed, the integration of a fuel system modification such as detailed in this case study will also impact external interfaces such as ground infrastructure. Therefore, the conceptual design phase of this project will need to embrace a systems view of change, and how it might propagate through what is a complex system comprising an aircraft segment and its systems, sub-systems and components; and the ground segment comprising the fuel production, transportation, storage and dispensing systems and sub-systems.

This section of this natural gas modification case study focusses on the aircraft segment. Although the conceptual design methodology models the aircraft and ground infrastructure, the change impact of the natural modification on the natural gas ground infrastructure is minimal insofar that changes in fuel type will not invoke a redesign of the natural gas fuelling infrastructure architecture. The extent of change is restricted to interfaces such as natural gas fuelling nozzle configuration and static-charge grounding clips. In this instance it is assumed that LNG refuelling interfaces currently used in the ground transportation industry would be adapted without little or no change for the aviation application. Nevertheless, the major change in the system is the provision of the natural gas ground infrastructure, which does not currently exist at airports. This aspect is addressed in the preceding section as a morphological matrix-based analysis. Therefore, the impact of the natural gas modification is limited here to the aircraft segment only. If in other cases an analysis of change propagation is required across aircraft-ground segments then the methods and techniques shown here can be extended as another Multiple-Domain Matrix (MDM), as described below.

### 1.6.2 Change propagation analysis

This section focuses on the change propagation analysis methods used to model the dependencies between system, subsystems and components of a particular natural gas fuel modification solution. This method can be used to support the risk/impact assessment of the solution which fits into the broader change management process within a design organisation.

Clarkson et al. (2001), outlines a method that predicts change propagation in complex design. The method outlined is referred to as the Change Propagation Method (CPM), which is fully described in Chapter 4 and Chapter 5. This section illustrates the practical application of this method which has been simplified and adapted to fit within a broader matrix-based conceptual design methodology.

This change propagation model is derived from the product itself in this case being the LNG fuel system architecture as illustrated in Figure 43. This LNG fuel system can be decomposed into sub-systems and components and their related dependencies. Furthermore, the existing aircraft fuel system can be decomposed into sub-systems and components as illustrated in Figure 43. Again, this existing AVGAS fuel system will possess dependencies although the design objective under a bi-fuel arrangement will ensure that these AVGAS sub-systems and components will not be modified. This design objective will maintain the existing approved configuration of these sub-systems and components, thus negating the requirement for additional recertification activities. Nevertheless, this design objective is testing in certain corner-conditions where bi-fuel fuel valves are required and changes are required to engine fuel injection and engine control units. These relationships and dependencies are analysed in detail in this section.

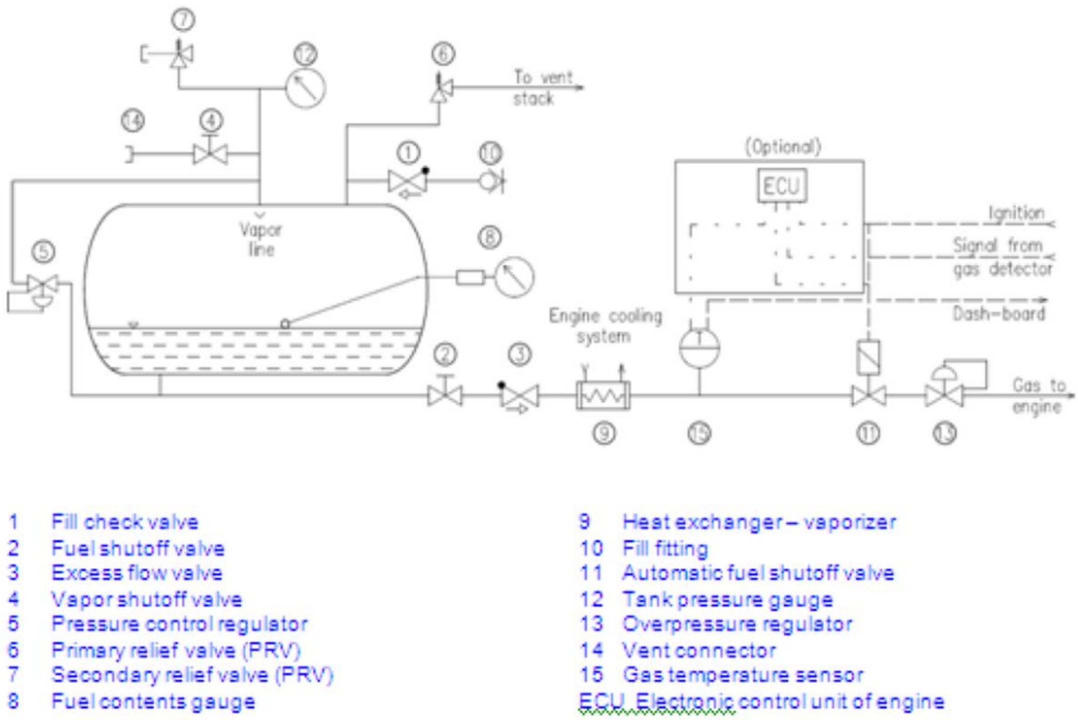



Figure 43. Typical LNG fuel system schematic

#### *1.6.2.1 Modification risk elements*

The CPM as outlined by Clarkson et al. (2001), relies on systems decomposition and dependencies as described above, with the dependencies obtained from an analysis of the aircraft architecture. These are defined in terms of the likelihood that the redesign of the sub-system or component will force the redesign of another and the subsequent impact, or extent, of that redesign. This is characterised in the likelihood definitions provided in Table 29.

Table 29. Direct likelihood definitions (*l*)

| **Direct likelihood** | | **Definitions** |
|---|---|---|
| Frequent | 0.9 | Multiple linkage types comprising mechanical, electrical and informational flows |
| Probable | 0.6 | Multiple linkage types, however omitting a linkage type(s) (e.g. mech static/mech dynamics) |
| Occasional | 0.3 | Single linkage or single type (e.g. mech static, mech dynamic, spatial, fluid flow, fluid flow dynamic) |

Therefore, all parameters in a complex system are connected with each other through the interaction of sub-systems and components to generate the desired behaviour of the entire system. The degree that this interaction occurs is determined by the linking parameters types and their attributes as defined in Table 29, and the direct impact definitions provided in Table 30. The likelihood definitions are derived from linking parameters as described by Eckert et al. (2004) and are further described in Chapter 4 and Chapter 5.

As described by Clarkson et al. (2001) the second element of this process is the assessment of the design change and the subsequent impact, or extent, of that redesign. Table 30 provides definitions of design change impact on the extent of engineering redesign required. As stated by Clarkson et al. (2001), both the likelihood and impact level shown in Table 29 and Table 30 are assigned a numerical value between 0 and 1 with this referring to the total change experienced during the redesign process. In this instance, these values are assigned 0.9, 0.6 and 0.3 commensurate with the likelihood (*l*) or impact (*i*) respectively.

Table 30. Direct impact definitions (*i*)

| **Direct Impact** | | **Definitions** |
|---|---|---|
| High | 0.9 | High level of redesign work to address change or change propagation impacts. |
| Medium | 0.6 | Medium level of redesign work to address change or change propagation impacts. |
| Low | 0.3 | Minimal level of redesign work to address change or change propagation impacts. |

These two relationships are combined to represent a scale of impact severity as shown in Table 31, where the impact severity is defined as the product of the likelihood (*l*) of the change and the impact (*i*) or cost of the subsequent change. This terminology is borrowed from Department of Defense, MIL-STD-882 (2012) systems safety management and simplified here in Table 31 to facilitate a three-level category which can be colour coded as shown.

Table 31. Change impact severity matrix

| IMPACT CATEGORY | HIGH 0.9 | MEDIUM 0.6 | LOW 0.3 |
|---|---|---|---|
| FREQUENT 0.9 | 0.8 | 0.5 | 0.3 |
| PROBABLE 0.6 | 0.5 | 0.4 | 0.2 |
| OCCASIONAL 0.3 | 0.3 | 0.2 | 0.1 |

These likelihood (*l*) and impact (*i*) values may be derived from past design modifications and from the experiences of senior designers. For this reason, the change propagation methods described here should be conducted as a design team activity as described in Chapter 5.

The final outcome of this process is to represent impact severity (*S*) of the change in one sub-system or component resulting in a change in a neighbouring sub-system or component by propagation over a common interface. This can be expressed in the relationship:

$$S_{jk} = i_{jk} \times l_{jk} \qquad \text{Equation 24}$$

Where *S* is the resultant impact severity at the *jk* element of the impact severity matrix.

$i_{jk}$ is the impact at the at the *jk* element of the direct impact matrix.

$l_{jk}$ is the likelihood at the at the *jk* element of the direct likelihood matrix.

The subsequent risks associated with the propagation of these changes to other sub-systems or components can then be determined from this data. This relationship is described in Chapter 5, and is implemented here in each direct change dependency matrix as presented in the following sections.

#### *1.6.2.2 Modification sub-system to aircraft sub-systems dependencies*

The CPM as outlined by Clarkson et al. (2001), relies on systems decomposition and dependencies involving a two-stage product model analysis of the aircraft architecture. A number of aircraft subsystems or components are identified, which together represent the whole of the aircraft segment. This aircraft subsystem is identified in the direct likelihood Design Structure Matrix (DSM) as shown in the lower matrix of Table 32. These subsystems are based on those developed by Clarkson et al. (2001), with tailoring to account for a fixed wing aircraft using Air Transport

Association of America (ATA), (2002) Specification 100 code terminology for guidance. This results in eighteen key aircraft subsystems all based on the same architecture being a simplified description of typical small aircraft. This DSM comprises a square matrix with eighteen key aircraft subsystems on each axis. Clarkson et al. (2001) states this number of aircraft subsystems is of a convenient size to model and provides adequate coverage without being overly complex to manage.

In contrast to the approach described by Clarkson et al. (2001) an upper matrix associated with this direct likelihood DSM is derived from the LNG modification subsystem and components. These modification subsystems and components are derived from the architecture of the LNG modification as shown by Figure 43. The relationship of LNG subsystems and components to aircraft subsystems is provided by this upper matrix through each shaded element. For example, the LNG fuselage belly tank, LNG fuel lines, LNG/AVGAS valve, LNG fuel shutoff valve, LNG pressure gauge and LNG quantity gauge interface directly with the fuselage and are therefore grouped to this aircraft subsystem. In a similar way other LNG subsystems and components are grouped to the various aircraft subsystems. Lastly Table 33 shows a key to abbreviations of LNG modification subsystems and components to assist in cross-referring these items to the schematic shown in Figure 43.

The lower matrix DSM provides column headings showing instigating subsystems and the row headings the impacted subsystems, whose designs change as a result of change to the instigating subsystems. Each column heading corresponds to an aircraft subsystem which relates to a group of LNG modification subsystems or components comprising the instigating change, with these changes assessed against the affected aircraft subsystems shown as rows in Table 32. For example, the instigating change resulting from the fuselage subsystem impacts the aircraft fuel system (added LNG/AVGAS valve and ECU interfaces), avionics (added LNG fuel gauges), auxiliary electrical (added fuel gauges power supply), cabling and piping (added LNG electrical cables and fuel lines), equipment & furnishings (added LNG regulators and valves), fire protection systems (added LNG related fire detection and suppression) and ice/rain protection (added LNG icing protection). For each impacted aircraft subsystem, a value of likelihood is derived from previous design modifications and from the experiences of senior designers as discussed earlier. This process results in a value of change likelihood being assigned in accordance with the definitions provided in Table 29. For example, a value of change likelihood is assigned a value of 0.9 for the fuselage LNG modifications impacting the aircraft fuel system. In this case there are multiple linkages and types comprising mechanical, electrical and informational flows (refer Figure 43) resulting in a frequent likelihood for change propagation. This process is repeated for all instigating changes corresponding to modification subsystems as shown in Table 32.

Table 32. Direct likelihood MDM - Mod subsystem to Aircraft subsystem

| | | | | | | | | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| LNG fuselage belly tank | | | | | | | | | | | | | | | | | | |
| LNG fuel lines | | | | | | | | | | | | | | | | | | |
| ECU | | | | | | | | | | | | | | | | | | |
| LNG pressure control regulator | | | | | | | | | | | | | | | | | | |
| LNG heat exchanger/vaporisor | | | | | | | | | | | | | | | | | | |
| LNG/AVGAS valve | | | | | | | | | | | | | | | | | | |
| (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | | | | | | | | | | | | | | | | | | |
| LNG fuel shutoff valve | | | | | | | | | | | | | | | | | | |
| LNG tank pressure gauge | | | | | | | | | | | | | | | | | | |
| LNG gas temperature sensor | | | | | | | | | | | | | | | | | | |
| LNG fuel quantity gauge | | | | | | | | | | | | | | | | | | |
| (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | | | | | | | | | | | | | | | | | | |
| **Direct Likelihood** | Fuselage | Engines | Wings | Nacelles | Stabilisers | Propellers | Hydraulic system | Fuel system | Engine auxiliaries | Flight Control System | Avionics | Auxiliary electrical | Cabling & Piping (C&P) | Landing gear | Equipment and Furnsihings (E&F) | Fire protection systems | Environmental Control Systems (ECS) | Ice and rain protection |
| Fuselage | X | | | | | | | 0.6 | | | 0.9 | | | 0.3 | 0.9 | 0.6 | | 0.6 |
| Engines | | X | | | | | | 0.3 | 0.3 | | | | | | | | | |
| Wings | | | X | | | | | 0.9 | | | | | 0.9 | | | | | 0.6 |
| Nacelles | | | | X | | | | | 0.6 | | | | | | | | | |
| Stabilisers | | | | | X | | | | | | | | | | | | | |
| Propellers | | | | | | X | | | | | | | | | | | | |
| Hydraulic system | | | | | | | X | | | | | | | | | | | |
| Fuel system | 0.9 | 0.9 | | | | | | X | 0.6 | | | | 0.6 | | | 0.3 | | 0.6 |
| Engine auxiliaries | | 0.6 | | | | | | 0.6 | X | | | | | | | | | |
| Flight Control System | | | | | | | | | | X | | | | | | | | |
| Avionics | 0.9 | 0.6 | | | | | | 0.9 | 0.6 | | X | 0.9 | | | 0.9 | | | |
| Auxiliary electrical | 0.6 | 0.6 | | | | | | 0.9 | 0.9 | | 0.9 | X | | | | 0.3 | 0.3 | 0.3 |
| Cabling & Piping (C&P) | 0.6 | 0.9 | 0.9 | | | | | 0.9 | 0.9 | | 0.9 | 0.9 | X | | 0.9 | 0.6 | 0.3 | 0.3 |
| Landing gear | | | | | | | | | | | | | | X | | | | |
| Equipment and Furnishings (E&F) | 0.9 | | | | | | | 0.9 | | | 0.6 | 0.6 | | | X | | 0.3 | |
| Fire protection systems | 0.3 | | | | | | | 0.3 | | | | | 0.3 | | | X | | |
| Environmental Control Systems (ECS) | | | | | | | | | | | | | | | | | X | |
| Ice and rain protection | 0.6 | | | | | | | | | | | 0.3 | | | | | | X |

It should be noted that the MDM representation of direct likelihood has been developed on the approach by Clarkson et al. (2001). However, this approach differs by extending the relationships into the modification architecture domain. This is shown by each shaded element in the upper LNG subsystems and components to aircraft subsystems matrix. This extension to the modification architecture domain is also applied to the impact matrix shown in Table 34, although it is not shown to reduce the size of the matrix in this instance.

Table 33. Modification subsystem abbreviations

| | Definition |
|---|---|
| AFSV | Auto fuel shutoff valve |
| EFV | Excess flow valve |
| PRV | Primary relief valve |
| SRV | Secondary relief valve |
| OR | Overpressure regulator |
| FF | Fuel fitting |
| CV | Check valve |
| VC | Vent connector |
| Vapor SV | Vapor Shutoff valve |

The development of the direct impact matrix follows a similar process as that outlined for the direct likelihood matrix as described above. Again, this change impact process is derived from a history of previous design changes and from the views from experienced designers. Each column heading corresponding to the aircraft subsystem is cross referenced to the LNG modification subsystems and components (not shown in Table 34) comprising the instigating change. These instigating changes are assessed against the affected aircraft subsystems as shown in Table 34. Using the same example previously provided above, the aircraft fuselage subsystems are related to the LNG modification through the LNG fuselage belly tank, LNG fuel lines, LNG/AVGAS valve, LNG fuel shutoff valve, LNG pressure gauge and LNG quantity gauge. The instigating change resulting from the fuselage LNG modifications impact the aircraft fuel system (LNG/AVGAS valve and ECU interfaces), avionics (LNG fuel gauges), auxiliary electrical (fuel gauges power supply), cabling and piping (LNG electrical cables and fuel lines), equipment & furnishings (LNG regulators and valves), fire protection systems (LNG related fire detection and suppression) and ice/rain protection (LNG icing protection). Obviously, the structure of the matrices ensures that these impacts align with the direct likelihood results as identified in Table 32.

As an example, a value of change impact is assigned a value of 0.9 for fuselage related LNG fuel system impacts on the aircraft fuel system. In this case there is potentially a high level of redesign work to address change or change propagation impacts (refer Table 30). This is due to the multiple redesign activities required to address fuselage-based LNG subsystem and component installations interfacing the aircraft fuel system (e.g. LNG/AVGAS valve and AVGAS fuel line re-routing).

Table 34. Direct impact matrix – Modification subsystems to Aircraft subsystems

| **Direct Impact** | Fuselage | Engines | Wings | Nacelles | Stabilisers | Propellers | Hydraulic system | Fuel system | Engine auxiliaries | Flight Control System | Avionics | Auxiliary electrical | Cabling & Piping (C&P) | Landing gear | Equipment and Furnsihings (E&F) | Fire protection systems | Environmental Control Systems (ECS) | Ice and rain protection |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Fuselage | X | | | | | | | 0.6 | | | 0.3 | | | 0.9 | 0.3 | 0.9 | | 0.9 |
| Engines | | X | | | | | | 0.6 | 0.9 | | | | | | | | | |
| Wings | | | X | | | | | 0.6 | | | | | 0.6 | | | | | 0.9 |
| Nacelles | | | | X | | | | | 0.6 | | | | | | | | | |
| Stabilisers | | | | | X | | | | | | | | | | | | | |
| Propellers | | | | | | X | | | | | | | | | | | | |
| Hydraulic system | | | | | | | X | | | | | | | | | | | |
| Fuel system | 0.9 | 0.6 | | | | | | X | 0.9 | | | | 0.6 | | | 0.9 | | 0.9 |
| Engine auxiliaries | | 0.9 | | | | | | 0.9 | X | | | | | | | | | |
| Flight Control System | | | | | | | | | | X | | | | | | | | |
| Avionics | 0.6 | 0.3 | | | | | | 0.6 | 0.3 | | X | 0.3 | | | 0.6 | | | |
| Auxiliary electrical | 0.6 | 0.6 | | | | | | 0.6 | 0.6 | | 0.3 | X | | | | 0.3 | 0.3 | 0.3 |
| Cabling & Piping (C&P) | 0.6 | 0.6 | 0.3 | | | | | 0.6 | 0.6 | | 0.3 | 0.3 | X | | 0.6 | 0.3 | 0.9 | 0.3 |
| Landing gear | | | | | | | | | | | | | | X | | | | |
| Equipment and Furnishings (E&F) | 0.6 | | | | | | | 0.3 | | | 0.6 | 0.6 | | | X | | 0.6 | |
| Fire protection systems | 0.3 | | | | | | | 0.3 | | | | | 0.6 | | | X | | |
| Environmental Control Systems (ECS) | | | | | | | | | | | | | | | | | X | |
| Ice and rain protection | 0.6 | | | | | | | | | | | 0.6 | | | | | | X |

As stated above, the modification change severity change matrix is generated using the relationship shown by $S_{jk} = i_{jk} \times l_{jk}$ Equation 24. This change severity matrix shown by Table 35, predicts the change in one subsystem as a result of a change to another related subsystem. Note that this matrix is created by *direct* subsystem dependencies only. This analysis differs from the CPM outlined by Clarkson et al. (2001) where the analysis combined both *direct* and *indirect* dependencies in order to evaluate change severity risk. This combined approach allows for a more detailed analysis of change propagation paths. However, this level of analysis is not undertaken here within the conceptual design process. The focus in this analysis is the determination of change severity identified in each matrix and then using a traditional risk parsing format for insertion into a standard project risk matrix.

Table 35 shows the change severity/risk as a result of the modification subsystems impacting the aircraft subsystems. These values of change severity are colour coded in accordance with Table 31 to allow easy identification. For example, the results of this change severity matrix show one occurrence of significant risk of change propagation. This relates to the example cited earlier where the installation of the LNG/AVGAS valve and AVGAS fuel line re-routing may require considerable redesign or rigorous application of design controls to mitigate change propagation risk. Furthermore, there is an additional impact of this change on certification where the impacts of these change resulting from the LNG modification is to be assessed against the airworthiness design standard. This is further discussed in Section 1.8 which deals with certification Domain Mapping Matrices (DMM).

An extension of the change propagation related DSM work described by Clarkson et al. (2001) includes the provision of an additional column and row providing totals of change severity. The additional column shown on the right-hand side of the change severity matrix shows the change severity total of each aircraft subsystem impacted by LNG modification subsystems as described above. These totals highlight the degree that aircraft subsystems are affected by the LNG modification. For example, aircraft cabling and piping is impacted by the LNG modification to the greatest extent although the rows indicated relatively low values of severity. This result highlights to the design team that design controls be applied to mitigate or manage change propagation relating to this aircraft subsystem.

In a similar way, the additional row shown on the lower part of the change severity matrix shows the change severity total of each LNG modification related aircraft subsystem impacted by each aircraft subsystems. These totals highlight LNG modification subsystems that are highly affected by the aircraft subsystems. For example, the LNG modification impacts numerous aircraft subsystems although the columns indicate relatively low values of severity. Again, this highlights to the design team that design controls be applied to manage or mitigate change propagation relating to the LNG modification impacting the aircraft subsystems as noted.

Table 35. Mod change severity/risk matrix – Mod subsystems to Aircraft subsystems

| **Mod Risk Matrix - Mod Subsystem to Aircraft Subsystem** | Fuselage | Engines | Wings | Nacelles | Stabilisers | Propellers | Hydraulic system | Fuel system | Engine auxiliaries | Flight Control System | Avionics | Auxiliary electrical | Cabling & Piping (C&P) | Landing gear | Equipment and Furnsihings (E&F) | Fire protection systems | Environmental Control Systems (ECS) | Ice and rain protection | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Fuselage | 0 | | | | | | | 0.4 | | | 0.3 | | | 0.3 | 0.3 | 0.5 | | 0.5 | 2.3 |
| Engines | | 0 | | | | | | 0.2 | 0.3 | | | | | | | | | | 0.5 |
| Wings | | | 0 | | | | | 0.5 | | | | | 0.5 | | | | | 0.5 | 1.6 |
| Nacelles | | | | 0 | | | | | 0.4 | | | | | | | | | | 0.4 |
| Stabilisers | | | | | 0 | | | | | | | | | | | | | | 0.0 |
| Propellers | | | | | | 0 | | | | | | | | | | | | | 0.0 |
| Hydraulic system | | | | | | | 0 | | | | | | | | | | | | 0.0 |
| Fuel system | 0.8 | 0.5 | | | | | | 0 | 0.5 | | | | 0.4 | | | 0.3 | | 0.5 | 3.1 |
| Engine auxiliaries | | 0.5 | | | | | | 0.5 | 0 | | | | | | | | | | 1.1 |
| Flight Control System | | | | | | | | | | 0 | | | | | | | | | 0.0 |
| Avionics | 0.5 | 0.2 | | | | | | 0.5 | 0.2 | | 0 | 0.3 | | | 0.5 | | | | 2.3 |
| Auxiliary electrical | 0.4 | 0.4 | | | | | | 0.5 | 0.5 | | 0.3 | 0 | | | | 0.1 | 0.1 | 0.1 | 2.3 |
| Cabling & Piping (C&P) | 0.4 | 0.5 | 0.3 | | | | | 0.5 | 0.5 | | 0.3 | 0.3 | 0 | | 0.5 | 0.2 | 0.3 | 0.1 | 3.9 |
| Landing gear | | | | | | | | | | | | | | 0 | | | | | 0.0 |
| Equipment and Furnishings (E&F) | 0.5 | | | | | | | 0.3 | | | 0.4 | 0.4 | | | 0 | | 0.2 | | 1.7 |
| Fire protection systems | 0.1 | | | | | | | 0.1 | | | | | 0.2 | | | 0 | | | 0.4 |
| Environmental Control Systems (ECS) | | | | | | | | | | | | | | | | | 0 | | 0.0 |
| Ice and rain protection | 0.4 | | | | | | | | | | | 0.2 | | | | | | 0 | 0.5 |
| | 3.1 | 2.2 | 0.3 | 0.0 | 0.0 | 0.0 | 0.0 | 3.6 | 2.4 | 0.0 | 1.2 | 1.1 | 1.1 | 0.3 | 1.4 | 1.1 | 0.5 | 1.8 | |

### *1.6.2.3 Modification components to aircraft subsystems dependencies*

This DSM-based approach can be extended to other dependencies as described below using a slightly different mapping. In this case rather than group LNG subsystems and components relative to aircraft subsystems, it is advantageous to assess LNG components separately against the aircraft subsystems. In this way the direct change dependencies can be assessed for each LNG component initiating a change against an aircraft subsystem. Direct likelihood and direct impact matrices are developed, this time resulting in a non-square Domain Mapping Matrix (DMM). These DMMs are non-square as the columns corresponding to LNG modification components does not equal the number of rows corresponding to the aircraft subsystems. Given that the same steps are involved in determining direct likelihood and direct impact DMMs, then these matrices will not be presented here. Rather, the outcome of this process, being the LNG modification change severity matrix, is presented here as Table 36. Table 36 shows the change severity/risk as a result of the modification components impacting the aircraft subsystems. As stated earlier these values of change severity are colour coded in accordance with Table 31 to allow easy identification. For example, the results of this change severity matrix show three occurrences of significant risk of change propagation. These occurrences highlight the change propagation risks associated with integration of the LNG fuel tank to the fuselage; the development and integration of the LNG heat exchanger/vaporiser with the engine exhaust system; and the physical sizing and fitment of the LNG tank to allow adequate undercarriage and ground clearances.

Again, the additional rows and columns to the right-hand side and lower part of the change severity matrix highlight those areas of severity for each impacted aircraft system and LNG components respectively. As expected, all LNG components impact the aircraft fuel subsystem with the highest scoring values being the LNG/AVGAS valve, as well as the LNG fuel tank. While the LNG fuel tank impacted the fuselage and landing gear aircraft subsystems as noted above.

Table 36. Mod change severity risk matrix – Mod components to Aircraft subsystems

| **Mod Change Risk Matrix - Mod Components to Aircraft Subsystems** | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Fuselage | 0.8 | 0.5 | | | | 0.2 | | 0.2 | 0.1 | | 0.1 | | 1.9 |
| Engines | | | 0.2 | | 0.5 | | | | | 0.1 | | | 0.8 |
| Wings | | 0.5 | | | | | | | | | | | 0.5 |
| Nacelles | | | | | | | | | | | | | 0.0 |
| Stabilisers | | | | | | | | | | | | | 0.0 |
| Propellers | | | | | | | | | | | | | 0.0 |
| Hydraulic system | | | | | | | | | | | | | 0.0 |
| Fuel system | 0.5 | 0.4 | 0.1 | 0.2 | 0.2 | 0.5 | 0.2 | 0.2 | 0.1 | 0.1 | 0.1 | 0.2 | 2.7 |
| Engine auxiliaries | | | 0.3 | | 0.8 | | | | | 0.1 | | | 1.2 |
| Flight Control System | | | | | | | | | | | | | 0.0 |
| Avionics | | | | | | | | | 0.2 | 0.1 | 0.2 | | 0.5 |
| Auxiliary electrical | | | 0.2 | | | 0.1 | 0.1 | 0.1 | 0.1 | 0.2 | 0.1 | | 0.8 |
| Cabling & Piping (C&P) | | 0.4 | | | | | | | | | | | 0.4 |
| Landing gear | 0.8 | | | | | | | | | | | | 0.8 |
| Equipment and Furnishings (E&F) | | | | | | | | | 0.2 | | 0.2 | | 0.4 |
| Fire protection systems | 0.2 | 0.2 | | | | | | | | | | | 0.4 |
| Environmental Control Systems (ECS) | | | | | 0.2 | | | | | | | | 0.2 |
| Ice and rain protection | 0.5 | 0.4 | | | | | | | | | | | 0.9 |
| | 2.9 | 2.3 | 0.7 | 0.2 | 1.7 | 0.8 | 0.3 | 0.5 | 0.6 | 0.5 | 0.6 | 0.2 | |

### *1.6.2.4 Modification components to components dependencies*

This same DSM approach is applied to assess change dependencies associated with LNG component change initiation and the impact of these changes with other LNG components. These changes may be necessary to account for changes in component ratings or settings which may be specific to the airborne operating environment. Given that the same steps are involved, then the direct likelihood and impact DSMs will not be presented here. Rather, the outcome of this process, being the LNG modification component change severity matrix, is presented as Table 37.

Table 37 shows the change severity matrix resulting from changes to LNG modification components impacting other LNG components. As stated earlier these values of change severity are colour coded in accordance with Table 31 to allow easy identification. The results of this change severity matrix show a distribution of low and medium change severity/risks. There are four medium severity risks (denoted a value of 0.5) and these correspond to component changes impacting fuel lines, and the Engine Control Unit (ECU) impact on various LNG pressure regulator and valve settings.

Again, the additional rows and columns to the right-hand side and lower part of the change severity matrix highlight those areas of severity for each impacted LNG component respectively. The right-hand column shows that main contributor to change severity/risk which is instigated by various LNG components impacting the LNG fuel lines. This is expected, as changes in LNG component geometry, sizing or location will impact the fuel line geometry, sizing and routing as a result of the common interfaces. The lower row shows the main contributor to change severity/risk instigated by various LNG valves and regulators impacting the fuel line and ECU.

Table 37. Mod change severity/risk matrix – Mod components to Mod components

| **Mod Change Risk Matrix - Mod Components to Mod Components** | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| LNG fuselage belly tank | 0 | 0.4 | | 0.1 | | | 0.3 | 0.1 | 0.1 | | 0.1 | 0.2 | 1.2 |
| LNG fuel lines | 0.5 | 0 | 0.1 | 0.5 | 0.2 | 0.4 | 0.5 | 0.4 | | 0.1 | | 0.1 | 2.8 |
| ECU | | | 0 | 0.4 | 0.4 | | 0.5 | | 0.1 | 0.4 | 0.1 | | 1.8 |
| LNG pressure control regulator | 0.1 | 0.1 | 0.1 | 0 | | | 0.1 | 0.1 | | | | 0.1 | 0.5 |
| LNG heat exchanger/vaporisor | | 0.1 | | 0.1 | 0 | | | | | 0.1 | | | 0.3 |
| LNG/AVGAS valve | | 0.1 | | | | 0 | 0.1 | | | | | | 0.2 |
| (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | 0.4 | 0.1 | 0.5 | 0.4 | 0.1 | 0.1 | 0 | 0.1 | | | | 0.1 | 1.7 |
| LNG fuel shutoff valve | 0.1 | 0.1 | | | | 0.1 | | 0 | | | | | 0.3 |
| LNG tank pressure gauge | 0.1 | | | 0.1 | | | 0.1 | | 0 | | | | 0.3 |
| LNG gas temperature sensor | | | 0.4 | | 0.1 | | | | | 0 | | | 0.5 |
| LNG fuel quantity gauge | 0.3 | | | | | | | | | | 0 | | 0.3 |
| (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | 0.1 | 0.1 | | | | | 0.1 | 0.1 | | | | 0 | 0.4 |
| | 1.5 | 0.9 | 1.1 | 1.5 | 0.7 | 0.5 | 1.7 | 0.7 | 0.2 | 0.5 | 0.2 | 0.5 | |

### 1.6.3 Change propagation risks and risk matrix

These DSM-based change propagation methods are useful in highlighting the severity of design change propagation risks. These change propagation risks can be captured within a traditional project risk matrix for consideration of the design team as shown by Table 38. Table 38 shows an example of such a risk matrix produced from the analysis of change propagation severity risks as described in this step. This table highlights the propagation matrix origin of the change severity risks, the risk title, which is based on the instigating change to the impacted item, and then a description of the risk using standard risk parsing. Given that this is a sample, the risk mitigation process is abbreviated here, noting that a traditional risk matrix would incorporate specific measures for risk mitigation and correction in its expanded version.

Table 38. LNG modification change severity propagation risks - sample

| Change Propagation Risk Matrix | | | | | |
|---|---|---|---|---|---|
| **Risk No.** | **Propagation Matrix** | **Risk Title** | **Description** | **Rated Risk** | **Actions & Comments** |
| 1 | Mod subsystem to Aircraft subsystem | Fuselage - Fuel system | There is a chance that changes to the fuselage as a result of the LNG tank and related LNG fuel system components will propagate changes to the existing AVGAS fuel system including fuselage interfaces which will impact design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 2 | Mod subsystem to Aircraft subsystem | Fuselage - Avionics | There is a chance that changes to the fuselage as a result of the LNG tank and related LNG fuel system components will propagate changes to the avionics system through installation of cockpit LNG fuel gauges, which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 3 | Mod subsystem to Aircraft subsystem | Fuselage - Equipment and Furnishings | There is a chance that changes to the fuselage as a result of the LNG tank and related LNG fuel system components will propagate changes or redesign of existing AVGAS fuel valves through installation of additional LNG fuel valves, which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 4 | Mod subsystem to Aircraft subsystem | Engine - Fuel System | There is a chance that changes to the engine as a result of the added LNG heat exchanger will propagate changes to the existing AVGAS fuel system, which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 5 | Mod subsystem to Aircraft subsystem | Engine Auxiliaries - Fuel System | There is a chance that changes to the engine auxiliaries as a result of LNG ECU and sensors will propagate changes to the existing AVGAS fuel system, which impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 6 | Mod subsystem to Aircraft subsystem | Ice & Rain Protection - Fuel System | There is a chance that changes to the ice and rain protection system as a result of the LNG fuel tank and fuel lines will propagate changes to the existing AVGAS fuel system. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 7 | Mod components to Mod components | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR - LNG fuel lines | There is a chance that changes to the various LNG valves will propagate changes to the LNG fuel lines through changes in physical geometry or performance specification, which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 8 | Mod components to Mod components | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR - ECU | There is a chance that changes to the various LNG valves will propagate changes to the ECU through changes physical interfaces or performance specifications, which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 9 | Mod components to Mod components | LNG pressure control regulator - LNG fuel lines | There is a chance that changes to the LNG pressure control regulator will propagate changes to the LNG fuel lines through changes in physical interfaces or performance specifications, which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |

| Change Propagation Risk Matrix | | | | | |
|---|---|---|---|---|---|
| **Risk No.** | **Propagation Matrix** | **Risk Title** | **Description** | **Rated Risk** | **Actions & Comments** |
| 10 | Mod components to Mod components | LNG fuselage belly tank - LNG fuel lines | There is a chance that changes to the LNG belly tank will propagate changes to the LNG fuel lines through changes in physical interfaces or performance specifications, which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 11 | Mod components to Aircraft subsystems | LNG fuselage belly tank - Fuselage | There is a chance that changes to the LNG fuselage belly tank geometry/size will propagate changes to the fuselage via the structural and fuel line interfaces, impacting design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 12 | Mod components to Aircraft subsystems | LNG fuselage belly tank - Fuel System | There is a chance that changes to the LNG fuselage belly tank geometry/size will propagate changes to the fuel system physical geometry or routing, impacting design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 13 | Mod components to Aircraft subsystems | LNG fuselage belly tank - Landing gear | There is a chance that changes to the LNG fuselage belly tank size/geometry will propagate changes to landing gear through ground clearance or undercarriage door clearance limitations, impacting design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 14 | Mod components to Aircraft subsystems | LNG fuselage belly tank - Ice & Rain Protection | There is a chance that changes to the LNG fuselage belly tank size/geometry will propagate changes to the Ice and Rain Protection system through requirement to prevent icing of fuel tank and fuel line components, impacting design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 15 | Mod components to Aircraft subsystems | LNG/AVGAS valve - Fuel System | There is a chance that changes to the LNG/AVGAS valve will propagate change to the fuel system through changes in physical interface geometry or performance specifications, impacting design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |

### 1.6.4 Case risk scatter plots

A means to visualise the risks determined from these change severity/risk matrices is described by Clarkson et al. (2001) where the *L* and *I* values are mapped and plotted on a risk scatter plot. A sample of the form of this case risk scatter plot is provided in Figure 44, where the lines indicate isopleths of equal risk. The data shown in this plot is determined by the risk severity results of the Modification severity/risk matrix shown in Table 35.

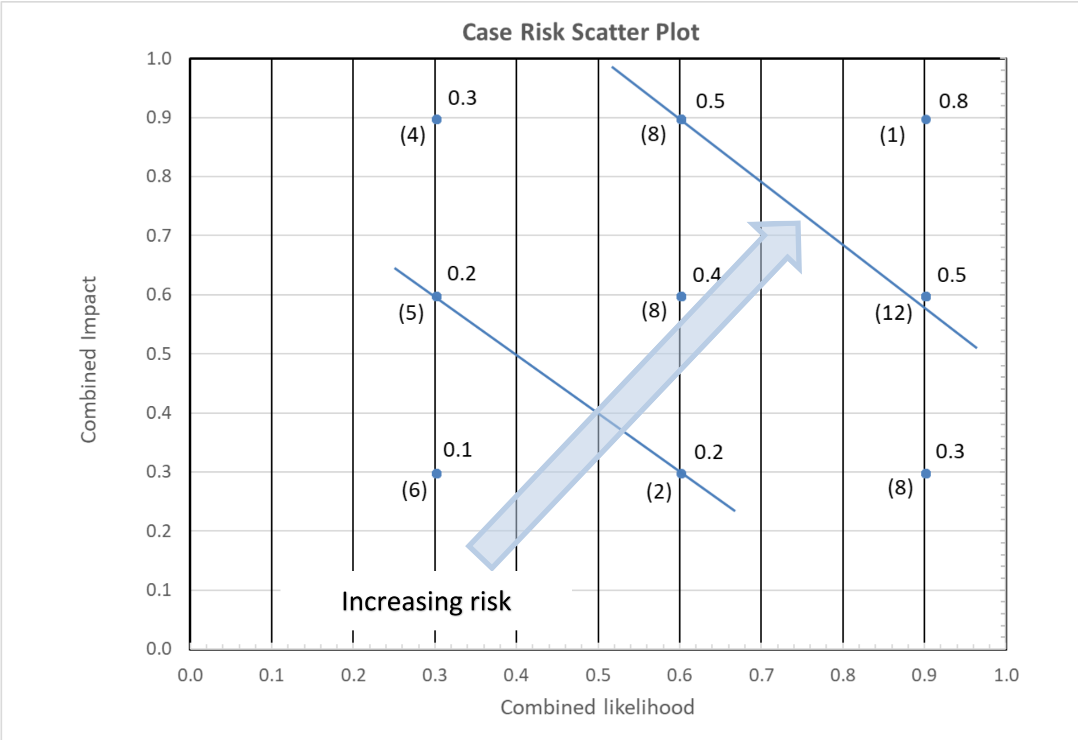



Figure 44. Sample case risk plot for Mod subsystem to Aircraft subsystem

### 1.6.5 Change propagation trees

As stated earlier, the implementation of the Change Propagation Method as described by Clarkson et al. (2001) has considered direct change impacts only in order to simplify the approach for conceptual design. The intent is to extend this analysis to combine the risk of propagation from its direct and indirect components in the next phase of the design lifecycle. A change propagation tree as described by Keller et al. (2005) is one way and to visualise the complexity of direct and indirect changes by showing the various propagation paths. The change propagation tree is constructed by starting at the root component in the system, with all other subsystems directly connected to this component represented as children. This is repeated for all children subsystems until the probability is of each branch falls under the designers defined threshold. As stated by Clarkson et al. (2001), the combined likelihood is the probability that the end effect will result, regardless of the path. This paper by Clarkson et al. (2001) provides a method to evaluate this combined risk mathematically relying on a propagation tree approach. As stated above, this conceptual design methodology does not evaluate these combined risks. However, an illustration of the partial change propagation tree based on an analysis of results of the Modification severity/risk matrix

(Table 35) is provided in Figure 45. This partial change propagation tree has been prepared using the method provided in Clarkson et al. (2001), and highlights the complexity of change propagation for what is a relativity non-complex modification in this case.

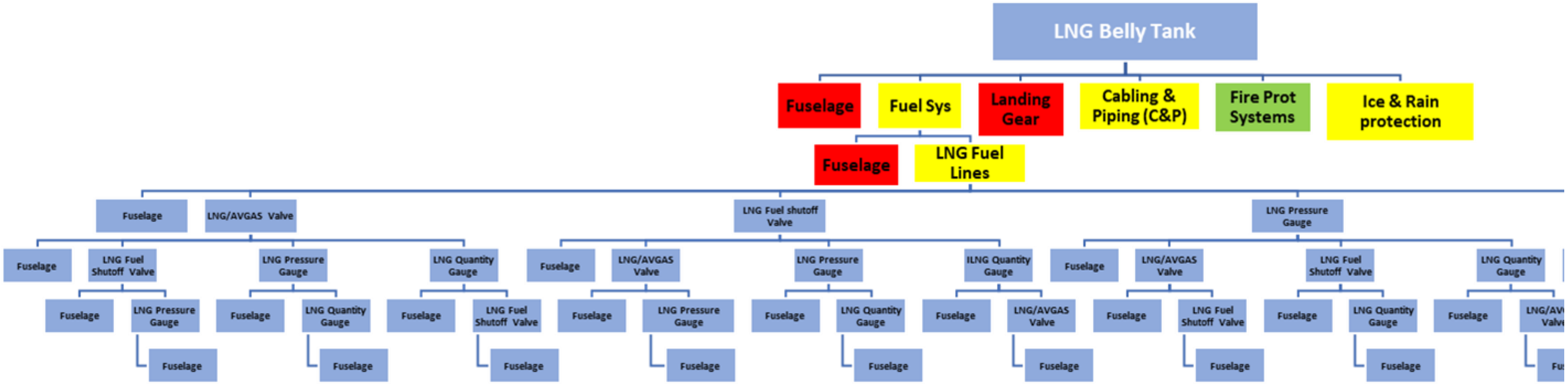



Figure 45. Partial change propagation tree for LNG belly tank impacts

## 1.7 ENGINEERING DOMAIN MAPPING MATRIX – STEP 6

### 1.7.1 Overview

In this step the methods used to evaluate the impact of changes to subsystems are extended into the engineering domain in order to evaluate the impact of costs resulting from the LNG fuel system modification. This approach uses the results of the change propagation analysis described previously in Section 1.6, which is then used to estimate the impact on engineering development costs. The method is further extended to determine the impact of the LNG fuel system modification on the development of the system specification, and also the design parameters on subsequent engineering development costs. This latter extension is also provided to evaluate the impact of ground infrastructure fuelling system components on the development of the specification and related design parameters in the same way.

This distinction between engineering development and certification as dealt with in the next section, is made here to ensure that the methodology accounts for the discrete activity that involves conceptual design and related cost estimates. These engineering activities, like other steps in the conceptual phase are iterative in nature and the use of these engineering Domain Mapping Matrices (DMM) would be undertaken within processes as described in Sections 1.5 and 1.6. Therefore, one such output of this DMM-based assessment is the draft Systems Specification document detailing the LNG fuel modification and the ground fuelling station infrastructure. This draft system specification document is based on the structure as detailed in the respective DMMs. The other output of this assessment was an initial Cost Breakdown Structure (CBS) which estimated engineering costs associated with the LNG fuel modification and the ground infrastructure fuelling system as well as the Requirements Allocation Sheet which documented the connection between allocated function, performance and the physical system.

### 1.7.2 Design change impacts

Table 39 and Table 40 shows the modification design engineering DMM describing the relationship of the LNG modification components to requirements and design change parameters. These two tables have been split into two parts to facilitate presentation in this thesis.

Part 1 of this DMM shown by Table 39 corresponds to the LNG aircraft modification which is based on design change propagation data determined from Section 1.6. Specifically, this DMM data are shown transposed from Table 36 and Table 37 modification change severity/risk results. As described earlier, the colour coding provides a measure of engineering development risk impacting engineering cost. These DMM data are transposed to provide LNG modification components as common columns to which cost data estimates can be aligned either from an aircraft system or as a modification component. The right-hand side of the DMM shown in Table 39 shows engineering costs resulting from LNG fuel system modification impacts on aircraft subsystems. These costs are broken down into (1) engineering management (2) engineering design, and (3) engineering development and support, using a similar cost structure as described in Fabrycki & Blanchard (1991). The determination of these engineering costs is not within scope of this thesis. However, it is sufficient in this conceptual design phase to identify those costs impacted by the LNG fuel system modification by those blue highlighted elements in Table 39. The basis of cost estimation is described in Chapter 4.

Table 39. LNG Mod design engineering DMM – Part 1

| Mod Components to Aircraft Subsystems | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Engineering Costs: Engineering management | Engineering Costs: Engineering design | Engineering Costs: Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Fuselage | 0.8 | 0.5 | | | | 0.2 | | 0.2 | 0.1 | | 0.1 | | | | |
| Engines | | | 0.2 | | 0.5 | | | | | 0.1 | | | | | |
| Wings | | 0.5 | | | | | | | | | | | | | |
| Nacelles | | | | | | | | | | | | | | | |
| Stabilisers | | | | | | | | | | | | | | | |
| Propellers | | | | | | | | | | | | | | | |
| Hydraulic system | | | | | | | | | | | | | | | |
| Fuel system | 0.5 | 0.4 | 0.1 | 0.2 | 0.2 | 0.5 | 0.2 | 0.2 | 0.1 | 0.1 | 0.1 | 0.2 | | | |
| Engine auxiliaries | | | 0.3 | | 0.8 | | | | | 0.1 | | | | | |
| Flight Control System | | | | | | | | | | | | | | | |
| Avionics | | | | | | | | | 0.2 | 0.1 | 0.2 | | | | |
| Auxiliary electrical | | | 0.2 | | | 0.1 | 0.1 | 0.1 | 0.1 | 0.2 | 0.1 | | | | |
| Cabling & Piping (C&P) | | 0.4 | | | | | | | | | | | | | |
| Landing gear | 0.8 | | | | | | | | | | | | | | |
| Equipment and Furnishings (E&F) | | | | | | | | | 0.2 | | 0.2 | | | | |
| Fire protection systems | 0.2 | 0.2 | | | | | | | | | | | | | |
| Environmental Control Systems (ECS) | | | | | 0.2 | | | | | | | | | | |
| Ice and rain protection | 0.5 | 0.4 | | | | | | | | | | | | | |

| Mod Components to Mod Components Change Propagation | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Engineering management | Engineering design | Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| LNG fuselage belly tank | 0 | 0.4 | | 0.1 | | | 0.3 | 0.1 | 0.1 | | 0.1 | 0.2 | | | |
| LNG fuel lines | 0.5 | 0 | 0.1 | 0.5 | 0.2 | 0.4 | 0.5 | 0.4 | | 0.1 | | 0.1 | | | |
| ECU | | | 0 | 0.4 | 0.4 | | 0.5 | | 0.1 | 0.4 | 0.1 | | | | |
| LNG pressure control regulator | 0.1 | 0.1 | 0.1 | 0 | | | 0.1 | 0.1 | | | | 0.1 | | | |
| LNG heat exchanger/vaporisor | | 0.1 | | 0.1 | 0 | | | | | 0.1 | | | | | |
| LNG/AVGAS valve | | 0.1 | | | | 0 | 0.1 | | | | | | | | |
| (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | 0.4 | 0.1 | 0.5 | 0.4 | 0.1 | 0.1 | 0 | 0.1 | | | | 0.1 | | | |
| LNG fuel shutoff valve | 0.1 | 0.1 | | | | 0.1 | | 0 | | | | | | | |
| LNG tank pressure gauge | 0.1 | | | 0.1 | | | 0.1 | | 0 | | | | | | |
| LNG gas temperature sensor | | | 0.4 | | 0.1 | | | | | 0 | | | | | |
| LNG fuel quantity gauge | 0.3 | | | | | | | | | | 0 | | | | |
| (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | 0.1 | 0.1 | | | | | 0.1 | 0.1 | | | | 0 | | | |

Part 2 of the engineering DMM shown in Table 40 describes the relationship of the LNG modification components to requirements and design change parameters. This analysis evaluates the impact of the LNG modification components on requirements and design changes parameters to facilitate a functional view and physical view of the LNG modification. This aspect of the DMM provides a 'reverse' view of the functional analysis step undertaken in Systems Engineering and assigns engineering resources to these activities at a functional level. As stated above costs are identified by the blue highlighted elements in Table 40. Again, the determination of these costs is not within scope of this thesis.

Table 40. LNG Mod design engineering DMM – Part 2

| **Mod Components to System Specification Reqts** | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Engineering management | Engineering design | Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| The modification shall minimise the impacts to range, payload, & cruise speed performance. | | | | | | | | | | | | | | | |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | | | | | | | | | | | | | | | |
| The modification lifecycle costs (LCC) shall be kept to a minimum | | | | | | | | | | | | | | | |
| The modification shall minimise emissions. | | | | | | | | | | | | | | | |
| The modification shall be compatible with a range of commuter aircraft types. | | | | | | | | | | | | | | | |
| The modification shall not impact passenger, baggage or cargo space. | | | | | | | | | | | | | | | |

| **Mod Components to Design Change Parameters** | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Engineering management | Engineering design | Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Decrease LNG installation drag | | | | | | | | | | | | | | | |
| Decrease LNG installation weight | | | | | | | | | | | | | | | |
| Reduce LNG fuel system related LCC | | | | | | | | | | | | | | | |
| Maintain engine power output | | | | | | | | | | | | | | | |
| Maintain aircraft payload volume and space capacity | | | | | | | | | | | | | | | |

Table 41 shows the ground infrastructure fuelling station development DMM describing the relationship of the LNG fuel station components to requirements and design parameters. The main purpose of this DMM is to present engineering costs in a common framework as described in the following section, noting that this ground fuelling station is assumed to be a non-developmental product. That is this ground segment does not require engineering development and support (i.e. prototypes and developmental testing) activities with the associated costs. This analysis evaluates the impact of the LNG fuelling station components on requirements and design parameters to facilitate a functional view and physical view of the LNG modification in a similar way to that described in Table 40 for the LNG fuel modification. As described above, this aspect of the DMM also provides a 'reverse' view of the functional analysis step undertaken in Systems Engineering. The Requirements Allocation Sheet is not necessary in this instance as the ground infrastructure has adopted a standard LNG fuelling station configuration.

Table 41. Ground infrastructure – LNG fuelling station development DMM

| LNG fueling station to System Specification Reqts | LNG storage tank | LNG bulk conditioner & heat exchanger | LNG lines and hoses | Truck offloading pump | Vapor return to truck | LNG pressure relief valve | LNG vent valve | Dispensing and conditioning pump | LNG refueling dispenser | LNG dispensing hose | Dispensing meter and controls | Containment wall | Engineeing costs: Project management | Engineeing costs: Engineering design |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| The ground refuelling station shall be able to store fuel without degradation or evaporation for a period of 1 week. | ■ | ■ | ■ | ■ | ■ | ■ | ■ | | | | | ■ | ■ | ■ |
| The ground refuelling station shall provide a refill rate of at least 200 gallons per hour (GGE). | | | | | | | | ■ | ■ | ■ | ■ | | ■ | ■ |
| The ground refuelling station shall provide sufficient fuel for 10 aircraft at 100 US gallons per day (GGE) with a holding capacity for 10 days. | ■ | | | | | | | | | | | | ■ | ■ |
| The ground refuelling station shall comply with CNG and LNG filling station safety standards (NFPA 52, the Vehicular Gaseous Fuel Systems Code, CFR Title 49, Part 193, NFPA 59A, Standard for the Production, Storage, and Handling of Liquefied Natural Gas (LNG)) | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ |
| The ground refuelling station shall be designed to minimise capital and operating costs. | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ | ■ |

| LNG fueling station to Design Parameters | LNG storage tank | LNG bulk conditioner & heat exchanger | LNG lines and hoses | Truck offloading pump | Vapor return to truck | LNG pressure relief valve | LNG vent valve | Dispensing and conditioning pump | LNG refueling dispenser | LNG dispensing hose | Dispensing meter and controls | Containment wall | Project management | Engineering design |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Decrease fill time | ■ | | | | | | | ■ | ■ | ■ | ■ | | ■ | ■ |
| Maximise fuel quantity storage time | ■ | ■ | | | | ■ | ■ | | | | | | ■ | ■ |
| Accessible Siting | ■ | | ■ | ■ | ■ | | | | | | | ■ | ■ | ■ |
| Minimise running costs | ■ | ■ | | | | ■ | ■ | ■ | | | | | ■ | ■ |

### 1.7.3 Cost estimation aspects

As stated above the means of cost estimation as shown in the right-hand columns of Table 39, Table 40 and Table 41 are based on Cost Breakdown Structure (CBS) definitions as detailed by Fabrycki & Blanchard (1991) and are fully described in Chapter 4.

## 1.8 CERTIFICATION DOMAIN MAPPING MATRIX – STEP 7

### 1.8.1 Overview

This section describes the methods used to evaluate the impact of the LNG fuel system modification on airworthiness certification requirements (certification domain) along with the provision of associated certification cost estimates. This approach uses a similar method to that employed for change propagation analyses described previously in Section 1.6. However, the method is extended to estimate the impact on certification related costs. In addition, the method provides as an output, a draft compliance summary (which comprises the certification DMM) as an important input to the draft Certification Program Plan (CPP). This CPP is a key planning document required by airworthiness regulators and applicants supporting a new type design or a significant modification (i.e. a STC).

### 1.8.2 Airworthiness standards and related codes

Airworthiness design standards specify regulatory requirements which must be achieved to provide an airworthy and safe type design, with advisory circulars and other guidance documentation providing information in relation to the acceptable means to demonstrate compliance against these design standards. These airworthiness standards address certification requirements for an aeronautical product and cover the aircraft, engine and propeller. Type certification of the aircraft, engine and propeller implies that aircraft, engine or propeller is manufactured according to the approved design can be issued an Airworthiness Certificate.

#### *1.8.2.1 Components*

This LNG fuel system modification comprises numerous components (fuel tank, valves and pressure regulators) that do not comply with any particular standard apart from those applicable to ground vehicle transportation. Aviation has a system of Technical Standard Orders (TSOs) which are minimum performance standards for specified materials, parts, and appliances used on aircraft. The apparent absence of aviation TSOs applicable to LNG fuel system components introduces additional certification challenges which need to be addressed as part of the broader certification effort. Although the certification DMM can be used to evaluate TSO compliance it will not be dealt with here in this thesis for reasons of brevity.

#### *1.8.2.2 Fuels*

The certification of aviation fuels is a complex area with Ziulkowski (2011) providing a good summary on certification aspects of aviation gasoline (AVGAS). Therefore, a change in fuel will have an impact to aircraft performance, fuel consumption, operating instructions, and instrument markings when compared to the original certification basis of the aircraft and engine. Furthermore, a change in AVGAS specification (e.g. removal of TEL) will have an adverse effect also on the certification of parts as a Parts Manufacturer Authorisation (PMA) or a Technical Standard Order (TSO) as described above. Therefore, it is apparent that a change of fuel will invalidate previous certification, and with it comes a considerable cost, technical risk, and possible adverse impacts to airworthiness and safety.

On this basis a DMM certification analysis of alternate aviation fuels is not presented here due to its inherent complexity as described in Chapter 5.

### 1.8.3 CAR Part 3 and FAR Part 23 - Aircraft certification impacts

#### *1.8.3.1 Certification basis*

The Cessna 421B aircraft was originally type certificated in the US and FAA Type Certificate A7CE, (2007) was issued to the aircraft and described as follows:

> *"Part 3 of the Civil Air Regulations dated May 15, 1956, except Subpart B, as amended by 3-1 through 3-5 and 3-8; Subpart B, paragraphs 23.25 through 23.253 of the Federal Aviation Regulations dated February 1, 1965, as amended by 23-1 through 23-7.*"

Civil Aeronautics Board (1956) Part 3 of the Civil Airworthiness Regulations (CAR) is a historical regulation, while the Federal Aviation Administration (1965) Federal Aviation Regulations (FAR) Part 23 is in current usage. Therefore, the certification basis of the Cessna 421B aircraft is a composite of these two regulations, with CAR Part 3 encompassing FAR Part 23 Subpart B, paragraphs 23.25 through 23.253. Given that both of these standards comprise a significant set of airworthiness requirements, then excerpts are shown in Table 42 and Table 43 to illustrate the methods and approaches.

#### *1.8.3.2 DMM structure*

Table 42 and Table 43 provide excerpts of the CAR Part 3 and FAR Part 23 aircraft certification DMM which are prepared using change likelihood and impact methods as described in Section 1.6. This DMM therefore presents the change severity/risk matrix for the LNG fuel system modification impacts on the certification basis as shown. For each applicable LNG fuel modification component, a likelihood and impact assessment is undertaken through reference to the particular requirement as defined in CAR Part 3 and FAR Part 23. It should be noted that Table 42 and Table 43 list the applicable CAR Part 3 and FAR Part 23 sub-paragraph requirements headings only. It is therefore necessary to refer to the actual content of each sub-paragraph as found in each respective design standard (i.e. CAR Part 3 and FAR Part 23) in order to make this assessment. This could also be achieved by hyperlinking these sub-paragraph headings to actual content. The process follows the same change propagation analysis as described earlier in this section and is therefore best undertaken by the design team.

#### *1.8.3.3 DMM coverage*

It should be emphasised that the certification DMM is extensive and is based on the airworthiness certification requirements of the applicable design standard, whether for the aircraft, the engine or any other applicable standard. For example, there are more than 1000 requirements that make up the CAR Part 3 and FAR Part 23 certification basis, and these cannot be reproduced in entirety in this thesis due to length. As described in Chapter 3, these regulations address all aspects of airworthiness, including aircraft flight performance, flight characteristics, flight loads, design and construction, pilot compartment (human-machine interface aspects), powerplant installation, fuel systems, electrical systems, and operating limitations and information (flight manual). For example, there are requirements that deal with human-machine interfaces such as operation of fuel valves (motion and effect of cockpit controls), control knob shape, fuel system arrangement and layout, and flight manual operating limitations information associated with management of fuel systems. These are all addressed by the applicable CAR Part 3 and FAR Part 23 airworthiness requirements. A sample of the fuel valve related requirements applicable under CAR Part 3 is described later in Table 43.

#### *1.8.3.4 Samples of certification DMM*

Table 42 shows a sample of the evaluation of the impact of the LNG fuel system modification on the airworthiness certification domain using the change propagation type methods as discussed earlier. CAR Part 3 – Subpart A contains general requirements which are applicable to all LNG modification components. For this reason, the costings are applicable to certification management only as no compliance finding are required. The FAR Part 23 – Subpart B requirements (as applicable under the certification basis described above) as shown on the lower part of the DMM are strongly impacted by the LNG fuselage belly tank. These higher severity/risk values indicate the impact of additional drag caused by the LNG fuselage belly tank affecting takeoff, climb, glide performance. The other lower severity/risk values indicate a secondary impact of valve and pressure regulator, ECU and other LNG component functions which may affect engine power output, thus affecting performance. These secondary impacts may be mitigated or eliminated by good design and by thorough systems testing. However, it is important to note that corresponding certification costs in this instance are assigned by the highest severity/risk impact corresponding to this requirement. In this case, performance requirements impacted by the LNG fuselage belly tank will affect certification management, compliance findings, support equipment development and test operations costs.

The right-hand side of the DMM shown by Table 42 presents the certification costs resulting from LNG fuel system modification as described above. These costs are fully described in Chapter 4 and Chapter 5. The determination of these certification costs is not within scope of this thesis, although the approach here is to use the results of the certification severity/risk matrix to inform the particular certification cost categories including allowances for severity/risk.

In addition, reference should also be made to change propagation analysis results in the previous section Table 35, Table 36 and Table 37 in order to characterise change severity/risks along with functional and performance impacts borne out from each particular sub-paragraph requirement.

Table 42. CAR 3/FAR Part 23 aircraft certification DMM – Excerpt 1

| | | | | | | | | | | | | | Certification costs | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Mod Components to CAR 3 and FAR Part 23 Airworthiness Reqts - Airplanes** | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Certification management | Airworthiness compliance & support | Equip development & inst support | Test operations |
| **CAR 3 - Subpart A - General** | | | | | | | | | | | | | | | | |
| §3.0 Applicability of this part. | | | | | | | | | | | | | | | | |
| §3.1 Definitions | | | | | | | | | | | | | | | | |
| §3.10 Eligibility for type certificate | | | | | | | | | | | | | | | | |
| §3.11 Designation of applicable regulations | | | | | | | | | | | | | | | | |
| §3.12 Recording of applicable regulations | | | | | | | | | | | | | | | | |
| §3.13 Type certificate | | | | | | | | | | | | | | | | |
| §3.14 Data required | | | | | | | | | | | | | | | | |
| §3.15 Inspections & tests | | | | | | | | | | | | | | | | |
| §3.16 Flight tests | | | | | | | | | | | | | | | | |
| §3.17 Airworthiness experimental and production certs | | | | | | | | | | | | | | | | |
| §3.18 Approval of materials parts, processes and appliances | | | | | | | | | | | | | | | | |
| §3.19 Changes in type design | | | | | | | | | | | | | | | | |
| §3.20 Airplane categories | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| **FAR Part 23 - Subpart B - Flight** | | | | | | | | | | | | | | | | |
| **General** | | | | | | | | | | | | | | | | |
| §23.21 Proof of compliance. | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | | | | |
| §23.23 Load distribution limits. | 0.2 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | | | | |
| §23.25 Weight limits. | 0.2 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | | | | |
| §23.29 Empty weight & corresponding CoG. | 0.2 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | 0.1 | | | | |
| §23.31 Removable ballast. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §23.33 Propeller speed and pitch limits | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| **Performance** | | | | | | | | | | | | | | | | |
| §23.45 General. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.49 Stalling speed. | 0.3 | | | | | | | | | | | | | | | |
| §23.51 Takeoff speeds. | 0.2 | | | | | | | | | | | | | | | |
| §23.53 Takeoff performance. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.55 Accelerate-stop distance. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.57 Takeoff path. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.59 Takeoff distance and takeoff run. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.61 Takeoff flight path. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.63 Climb: General. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.65 Climb: All engines operating. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.66 Takeoff climb: One-engine inoperative. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.67 Climb: One engine inoperative. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.69 Enroute climb/descent. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.71 Glide: Single-engine airplanes. | 0.2 | | | | | | | | | | | | | | | |
| §23.73 Reference landing approach speed. | 0.2 | | | | | | | | | | | | | | | |
| §23.75 Landing distance. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| §23.77 Balked landing. | 0.4 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | 0.2 | | | | 0.2 | | | | |
| **Flight Characteristics** | | | | | | | | | | | | | | | | |
| §23.141 General. | 0.2 | | | | | | | | | | | | | | | |
| **Controllability and Maneuverability** | | | | | | | | | | | | | | | | |
| §23.143 General. | 0.2 | | | | | | | | | | | | | | | |
| §23.145 Longitudinal control. | 0.2 | | | | | | | | | | | | | | | |
| §23.147 Directional and lateral control. | 0.2 | | | | | | | | | | | | | | | |
| §23.149 Minimum control speed. | 0.2 | | | | | | | | | | | | | | | |
| §23.151 Acrobatic maneuvers. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |

Table 43 shows a sample of the LNG fuel system modification impact on airworthiness certification requirements relating to the fuel tanks and fuel system components. CAR Part 3 – Subpart E contains requirements which are applicable to all LNG modification fuel system components, of which a sample is provided in Table 43. This evaluation illustrates the impact of certification on LNG fuel tanks, fuel system components and instrumentation, where the severity impact values highlight potential changes to the design or specification to comply with requirements. These changes may involve hardware redesign activities, new installation specifications, new design limitation, or additional test requirements. These certification-related change activities affect costs, where in this instance, they are rated by the highest severity/risk impact corresponding to the respective requirement shown in Table 43. In this case, performance requirements impacted by the LNG fuselage belly tank, fuel lines, AVGAS/LNG valve(s), and fuel system instruments will affect the certification costs associated with management, compliance findings, support equipment development and test operations activities.

To illustrate how this process is undertaken, the requirement relating to § 3.551 fuel valves is analysed below. As stated earlier, each requirement is denoted by a sub-paragraph heading, with the particulars of the sub-paragraph content being hyperlinked accordingly. The hyperlinking of subparagraph content is not presented in this thesis. However, an example of this content for CAR Part 3 sub-paragraph § 3.551 - fuel valves is provided as follows:

> *"§ 3.551 Fuel valves. (a) Means shall be provided to permit the flight personnel to shut off rapidly the flow of fuel to any engine individually in flight. Valves provided for this purpose shall be located on the side of the fire wall most remote from the engine.*
>
> *(b) Shut-off valves shall be so constructed as to make it possible for the flight personnel to reopen the valves rapidly after they have been closed.*
>
> *(c) Valves shall be provided with either positive stops or "feel" in the on and off positions and shall be supported in such a manner that load resulting from their operation or from accelerated flight conditions are not transmitted to the lines connected to the valve. Valves shall be*

*installed such that the effect of gravity and vibration will tend to turn their handles to the open rather than the closed position."*

Following through the example above, one can see that the fuelling valve requirement imposes multiple sub-requirements with one such sub-requirement being a "positive stop or "feel" in On/Off positions, and fuel lines that can withstand loads from operation or from accelerated flight conditions...". Furthermore, the fuelling valve needs to withstand vibration and gravity effects as stated. Indeed, a standard fuel/LNG valve used in automotive applications is unlikely to comply to these sub-requirements. This would require an engineering redesign or modification of the valve(s) and the fuel line interfaces, with this activity invoking a change propagation analysis process. These changes would then be analysed using the change propagation matrices, as determined by this certification DMM analysis. This will impact engineering which is handled within the Engineering Domain Mapping Matrix (DMM) as described earlier, where the associated change severity risks will be recorded, with initial costs associated with engineering redesign accounted for accordingly as per the methodology. Indeed, change severity risks associated with the AVGAS/LNG valve(s) have been previously identified in the sample risk matrix presented in Table 38.

The same can be applied to all other remaining airworthiness requirements such as flight manual operating limitations information (i.e. AVGAS/LNG fuel management arrangements), cockpit controls (i.e. location of fuel valve(s)), motion and effect of cockpit controls (i.e. operation of the fuel valve(s)), and control knob shape (i.e. what size and shape of fuel valve(s)).

It can therefore be seen that incorporation of the certification DMM analysis step is an essential component of this design methodology. For without it, important design considerations may be overlooked.

Lastly, the application of the certification DMM acts as a framework from which to judge the airworthiness of a proposed design concept. The excerpt of requirements relating to the fuel system as shown in Table 43 also highlights potential omissions. In this case the architecture of the LNG fuel system is based on a typical heavy vehicle fuel system. The typical heavy vehicle LNG fuel system used in this case (refer Figure 43) does not incorporate a fuel strainer or a fuel drain. However, the CAR Part 3 excerpt shown in Table 43 specifies requirements for §3.552 fuel strainer and §3.553

fuel system drains. This potential non-compliance needs to be addressed with the resulting analysis recorded in the Certification Program Plan (CPP) for NAA consideration, as discussed in Chapter 4. This potential non-compliance may also be incorporated as a requirement in the Design Specification Document (DSD), which would be updated or refined in the next design phase.

Table 43. CAR 3/FAR Part 23 aircraft certification DMM – Excerpt 2

| Mod Components to CAR 3 and FAR Part 23 Airworthiness Reqts - Airplanes | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Certification costs: Certification management | Airworthiness compliance & support | Equip development & inst support | Test operations |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Fuel Tanks** | | | | | | | | | | | | | | | | |
| §3.440 General. | 0.4 | | | | | | | | | | | | | | | |
| §3.441 Fuel tank tests. | 0.4 | | | | | | | | | | | | | | | |
| §3.442 Fuel tank installation. | 0.4 | | | | | | | | | | | | | | | |
| §3.443 Fuel tank expansion space. | 0.4 | | | | | | | | | | | | | | | |
| §3.444 Fuel tank sump. | 0.4 | | | | | | | | | | | | | | | |
| §3.445 Fuel tank filler connection. | 0.4 | | | | | | | | | | | 0.4 | | | | |
| §3.446 Fuel tank vents and carburetor vapor vents. | 0.5 | | | | | | 0.5 | | | | | 0.5 | | | | |
| §3.447-A Fuel tank vents. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.448 Fuel tank outlets. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| **Fuel Pumps** | | | | | | | | | | | | | | | | |
| §3.449 Fuel pump and pump installation. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| **Lines, Fittings and Accessories** | | | | | | | | | | | | | | | | |
| §3.550 Fuel system lines and fittings. | | 0.4 | | | | | | | | | | | | | | |
| §3.551 Fuel valves and controls. | | | | | | 0.5 | 0.5 | 0.5 | | | | | | | | |
| §3.552 Fuel strainer. | | | | | | | | | | | | | | | | |
| **Lines, Fittings and Accessories** | | | | | | | | | | | | | | | | |
| §3.553 Fuel system drains. | | | | | | | | | | | | | | | | |
| §3.554 Fuel system instruments. | | | | | | | | | 0.4 | 0.4 | 0.4 | | | | | |

### 1.8.3.5 *Cost estimation aspects*

As described earlier the means of cost estimation is shown in the right-hand columns of Table 42 and Table 43.

### 1.8.3.6 *Operating concept considerations*

It is important to emphasise that the LNG fuel modification as considered here is a bi-fuel configuration. This means that the aircraft can be operated on AVGAS or LNG fuel via an AVGAS/LNG selector valve. As discussed previously, one operating concept may involve use of AVGAS fuels only for takeoff and landing, with LNG fuel used only for climb, cruise and descent. In this way takeoff and landing performance remains the same as previously certificated apart from the additional drag effects caused by the LNG fuel tank installation. Nevertheless, it is possible that takeoff and landing may be undertaken using LNG fuel, either as an operational need, or unintentionally. In this case other airworthiness requirements may find that this condition may also require certification performance data and flight manual operating limitations information.

### 1.8.4 CAR Part 13 – Engine certification impacts

This approach uses the same method to that described in the previous section dealing with aircraft airworthiness certification. Again, this method provides as an output, a draft compliance summary (which comprises the certification DMM) as an important input to the draft Certification Program Plan (CPP). In this instance, engine certification may be dealt with in the same CPP supporting aircraft certification.

#### *1.8.4.1 Certification basis*

The Cessna 421B aircraft is powered by two Continental GTSIO-520-H reciprocating engines which were type certificated in the US and FAA Type Certificate E7CE (2011) was issued and described as follows:

> *CAR 13 effective June 15, 1956, as amended by 13-1 through 13-4. Application for type certificate dated November 30, 1962. Type Certificate No. E7CE issued July 24, 1964, for Model GTSIO-520-C; -D added February 27, 1967; -E added April 1, 1968; -H added April 28, 1970; -F added May 12, 1971; -K added July 31, 1974; -L added June 27, 1975; -M added January 7, 1976; -N added May 15, 1980. FAR 33.8, amendment 33-3, effective March 4, 1967 applicable to the GTSIO-520-L and -M, and -N.*

Civil Aeronautics Board (1956) Part 13 of the Civil Airworthiness Regulations (CAR) is a historical regulation, with an excerpt shown in Table 37 to illustrate the methods described here.

#### *1.8.4.2 DMM structure*

Table 37 provides an excerpt of the CAR Part 13 aircraft engine certification DMM which uses change likelihood and impact methods as described earlier. As per the previous section dealing with aircraft airworthiness certification, this DMM presents the change severity/risk matrix for the LNG fuel system modification impacts on the engine certification basis. For each applicable LNG fuel modification component, a likelihood and impact assessment is undertaken through reference to the particular requirement as defined in CAR Part 13.

The analysis approach is the same as that previously described noting that this engine certification standard deals with reciprocating and turbine engines. Given that the Cessna 421B engines are of the reciprocating type, requirements relating to turbine engines are not applicable. The non-applicable requirements are denoted in the DMM with an ‘X’. The same terminology is also adopted for non-applicable requirements in the aircraft airworthiness DMM shown in Table 35. In these instances, it is not necessary to determine certification costs. Table 44 is based on Cost Breakdown Structure (CBS) definitions as detailed by Fabrycki & Blanchard (1991). The totals of the highlighted columns are incorporated into a separate CBS document which can then be combined with the same from the Engineering DMM as described earlier. This ‘initial’ CBS can be then refined and updated in subsequent design phases to be later combined with the traditional project Work Breakdown Structure (WBS).

It is important to note that Table 44 highlights high certification risks associated with the ECU impacts on fuel and induction system, and ignition systems where natural gas fuels will require changes to these systems to ensure commensurate power and torque output with that of AVGAS. These natural gas-related changes may include high compression ratio intake valves and increased lift intake and exhaust valves as described by Abu Bakar et al. (2012). Although achievable, certification of such natural gas-related aero engine modifications and the design controls required to manage propagation of related changes will be problematic in terms of risk and subsequent impact to cost.

Table 44. CAR Part 13 engine certification DMM – Excerpt

| **Mod Components to CAR Part 13 Aircraft Engine Airworthiness** | LNG fuselage belly tank | LNG fuel lines | ECU | LNG pressure control regulator | LNG heat exchanger/vaporisor | LNG/AVGAS valve | (1) AFSV (2) EFV (3) PRV (4) SRV (5) OR | LNG fuel shutoff valve | LNG tank pressure gauge | LNG gas temperature sensor | LNG fuel quantity gauge | (1) LNG FF (2) LNG CV (3) VC (4) Vapor SV | Certification management | Airworthiness compliance & support | Equip development & inst support | Test operations |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Subpart A - General** | | | | | | | | | | | | | | | | |
| § 13.0 Applicability of this part. | | | | | | | | | | | | | | | | |
| § 13.1 Defiinitions. | | | | | | | | | | | | | | | | |
| § 13.10 Eligibility for type certificates. | | | | | | | | | | | | | | | | |
| § 13.11 Designation of applicable regulations. | | | 0.1 | | 0.1 | | | | | 0.1 | | | | | | |
| § 13.12 Recording of applicable regulations. | | | | | | | | | | | | | | | | |
| § 13.13 Type certificate. | | | | | | | | | | | | | | | | |
| § 13.14 Data required. | | | | | | | | | | | | | | | | |
| § 13.15 Inspections and tests. | | | | | | | | | | | | | | | | |
| § 13.16 Required tests. | | | | | | | | | | | | | | | | |
| § 13.17 Production certificates | | | | | | | | | | | | | | | | |
| § 13.18 Approval of materials parts processes & appliances. | | | 0.1 | | 0.1 | | | | | 0.1 | | | | | | |
| § 13.19 Changes in type design. | | | 0.1 | | 0.1 | | | | | 0.1 | | | | | | |
| § 13.20 Identification plate. | | | | | | | | | | | | | | | | |
| § 13.21 Instruction manual. | | | 0.1 | | 0.1 | | | | | 0.1 | | | | | | |
| **Subpart B - Reciprocating engines - Design & construction** | | | | | | | | | | | | | | | | |
| § 13.100 Scope. | | | 0.1 | | 0.1 | | | | | 0.1 | | | | | | |
| § 13.101 Materials. | | | 0.4 | | 0.4 | | | | | 0.4 | | | | | | |
| § 33.102 Fire prevention. | | | | | 0.4 | | | | | | | | | | | |
| § 13.103 Vibration. | | | 0.4 | | 0.4 | | | | | | | | | | | |
| § 13.104 Durability. | | | 0.4 | | 0.4 | | | | | | | | | | | |
| § 13.110 Fuel and induction system. | | 0.4 | 0.8 | 0.5 | 0.8 | 0.5 | 0.5 | 0.4 | 0.3 | | 0.3 | 0.4 | | | | |
| § 13.111 Ignition systems. | | | 0.8 | | | | | | | 0.1 | | | | | | |
| § 13.112 Lubrication system. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| § 13.113 Engine cooling. | | | | | 0.4 | | | | | | | | | | | |
| § 13.114 Engine mounting attachments. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| § 13.115 Accessory attachments. | | | | | 0.4 | | | | | | | | | | | |
| **Reciprocating aircraft engines - Block tests** | | | | | | | | | | | | | | | | |
| § 13.150 General. | | | 0.1 | | 0.1 | | | | | 0.1 | | | | | | |
| § 13.151 Vibration test. | X | X | X | X | X | X | X | X | X | X | X | X | | | | |
| § 13.152 Calibration tests. | | | 0.5 | | 0.5 | | | | | 0.3 | | | | | | |
| § 13.153 Detonation test. | | | 0.5 | | 0.5 | | | | | 0.3 | | | | | | |
| § 13.154 Endurance test. | | | 0.5 | | 0.5 | | | | | 0.3 | | | | | | |
| § 13.155 Operation test. | | | 0.5 | | 0.5 | | | | | 0.3 | | | | | | |
| § 13.156 Teardown inspection | | | 0.3 | | 0.5 | | | | | 0.3 | | | | | | |
| § 13.157 Engine adjustments and parts replacements. | | | 0.3 | | 0.5 | | | | | 0.3 | | | | | | |

### 1.8.5 LNG fuelling station safety standards

The same certification DMM approach can be applied to LNG fuel stations where an industry standard can be used as the certification basis. For example, an industry standard such as NFPA 59A, Standard for the Production, Storage, and Handling of LNG, could be used as the basis of a certification DMM. As per the previous section dealing with aircraft airworthiness certification, this DMM may present the change severity/risk matrix for the LNG fuelling station resulting from specific changes to cater for aviation operations.

## 1.9 CONCEPT ANALYSIS – STEP 8

### 1.9.1 Overview

As described earlier the main metrics associated with this commuter aircraft alternate fuel modification are those associated with aircraft performance, costs and sustainability as described in the QFD matrix. In addition, the same approach can be adopted for ground fuelling infrastructure where performance and cost are important attributes to providing an overall systems solution. Given that this conceptual design methodology is an early step in the design process, these metrics can be further modelled using the morphological matrix quantisation as a starting point to provide initial estimates of system attributes for later consideration. The output of this activity is a Design Specification, where a summary of this conceptual design process is documented for further refinement.

### 1.9.2 Aircraft Payload-Range performance

Figure 46 and Figure 47 are Payload-Range diagrams generated from range performance estimates as determined in Appendix 2. These Payload-Range diagrams provide a convenient measure of two aircraft performance metrics being range and the payload potential. The payload which can be carried while flying a given range is of high importance to commercial aircraft operators and this is an indicator of the profit potential for the aircraft.

Figure 46 provides the Payload-Range diagram for the modified Cessna 421B aircraft operating at 20000 ft (6000 m) altitude. Two sets of data are shown corresponding to two configurations. The first configuration corresponds to a flight carried out using AVGAS fuel with no belly tank. Whereas the second configuration corresponds to a flight carried out using the LNG/AVGAS bi-fuel combination. This latter LNG/AVGAS bi-fuel configuration includes a belly tank installation. This latter flight is flown using a maximum combined fuel load with LNG for the cruise phase of flight, then AVGAS is used for the remainder of the flight. Details of the respective fuel loads are summarised in Table 45. The Payload-Range diagram has been prepared using a 121 lbs (55 kg) AVGAS fixed reserve fuel. The short segment on the left-hand side of Figure 46 accounts for the potential range flown on this fixed reserve fuel load with maximum payload. This segment then decreases payload linearly as fuel load increases to the maximum range as shown on Figure 46 where this is achieved with a residual payload as shown. This Payload-Range diagram is typical of a small aircraft with residual payload capacity at maximum range as described by Gudmundsson (2014).

Table 45. Fuel loading summary for AVGAS/LNG modified Cessna 421B

| | **Fuel load** |
|---|---|
| AVGAS_wing tip tanks (lbs) | 600 |
| AVGAS_wing tanks (lbs) | 190 |
| LNG_belly tank (lbs) | 230 |
| Total fuel (lbs) | 1020 |

#### *1.9.2.1 Payload-Range results*

As discussed in Appendix 2, these Payload-Range results are based on preliminary estimates of $C_L/C_D$ ratio and LNG *SFC*, and are subject to uncertainties implicit in the analysis of aerodynamic data and SFC data. Given these uncertainties, the range results shown in Figure 46 shows that the AVGAS/LNG bi-fuel option outperforms the AVGAS configuration by a small margin of about 9 NM (17 km) at 20000 ft (6000 m). This marginal increase in range can be attributed to the lower SFC of LNG compared to AVGAS despite the increased drag caused by the LNG fuel belly tank installation.

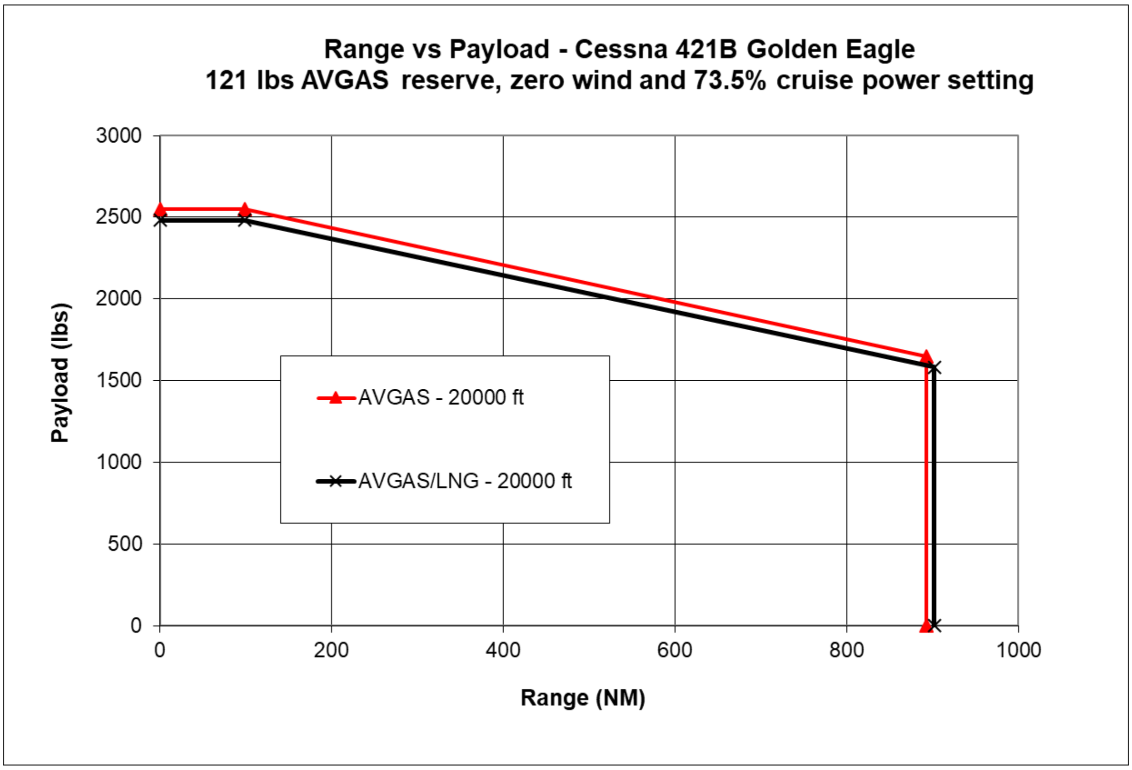



Figure 46. Payload-Range – Comparing AVGAS and AVGAS/LNG bi-fuel configurations

### *1.9.2.2 Sensitivity study*

A sensitivity study undertaken on $C_L/C_D$ ratio (specifically drag) and LNG *SFC* attempted to rationalise these uncertainties at a sample altitude of 20000 ft (6000 m) as shown by Figure 47. This analysis of uncertainty decreases $C_L/C_D$ ratio by 5% to account for variation in LNG fuel tank installation drag and increases LNG *SFC* by 10% to account for uncertainties in engine performance and SFC data. These results shown in Figure 47 indicates that the combined uncertainties in these quantities results in a 64 NM (118 km) reduction in LNG/AVGAS range performance compared to the AVGAS only configuration. Therefore, there is potential that any range improvements shown in Figure 46 could be reduced if predicted $C_L/C_D$ ratio and LNG *SFC* performance figures are not realised.

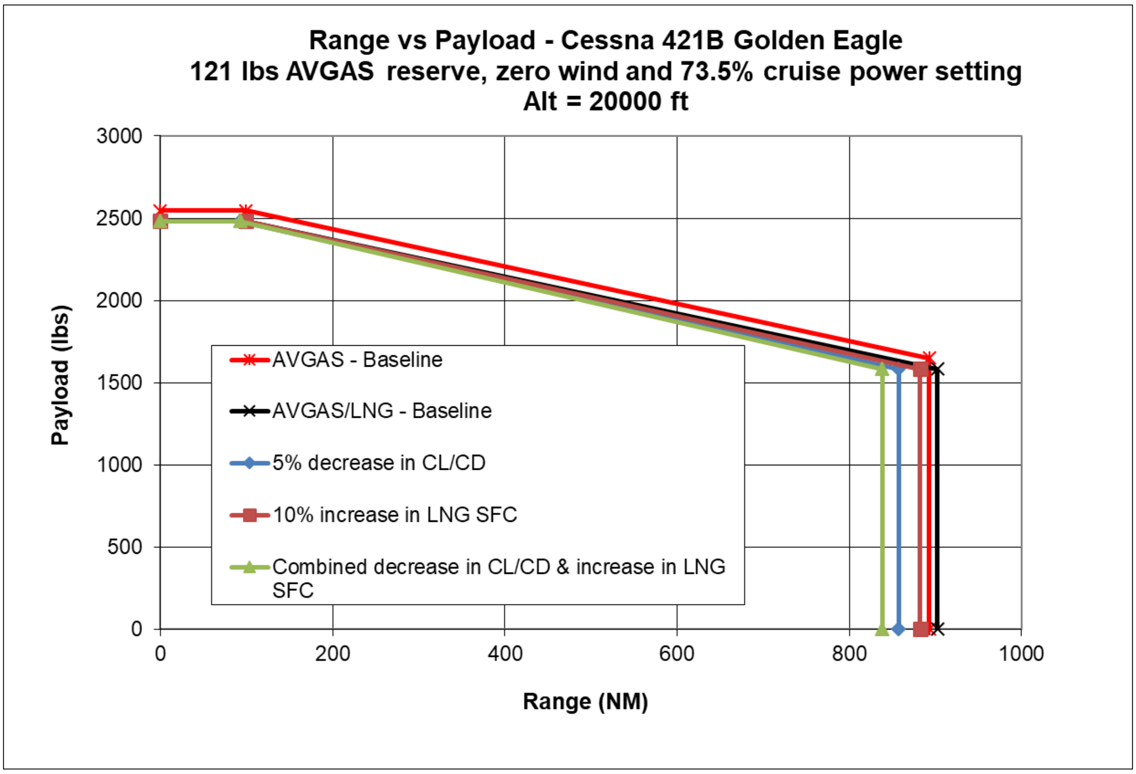



Figure 47. Payload-Range sensitivity to $C_L/C_D$ ratio and *SFC*

#### *1.9.2.3 Comparable research results*

The results as presented above are consistent with Payload-Range values predicted by Burston et al. (2013) for a similar study on larger jet powered transport aircraft modified for Bio-LNG fuels. This study assumed a A320-A350 sized aircraft modified to store Bio-LNG fuels in an insulated wing-box and purpose developed under-wing pods. Burston et al. (2013) showed that an aircraft modified for Bio-LNG fuels can achieve a small increase in range performance if cruise lift-to-drag (L/D) value of the modified aircraft does not fall below approximately 7% of the Jet-A fuel equivalent. This range performance prediction assumed 10% specific fuel consumption reduction compared with Jet-A fuel.

The Payload-Range trends presented in the study by Burston et al. (2013) are consistent with the results from this case study, and therefore this provides a point of validation supporting this analysis.

### 1.9.3 Aircraft cruise speed

The change in aircraft cruise speed performance as a result of the LNG fuel tank installation has been estimated in Appendix 2. Here the analytical predictions show a reduction in cruise airspeed of approximately 5-knots at sea level, at a 73.5% cruise power setting. Although this result is acceptable given the limitations of the analytical methods applied here, the actual reduction in cruise airspeed performance may be higher due to uncertainties in LNG fuel tank drag prediction as discussed above.

### 1.9.4 Emissions estimates

#### *1.9.4.1 Overview*

This section describes the results of a literature review of data pertaining to comparative testing of gasoline and natural gas exhaust emissions used in automotive Spark Ignition (SI) internal combustion engines. This review presents a comparative analysis of exhaust emissions for both fuel types which can be used as a measure when making design trade-offs within a conceptual evaluation framework.

#### *1.9.4.2 Methodology*

The literature review focused on data comparing emissions from natural gas and gasoline fuels. In all cases, the data obtained were derived from tests conducted on automotive SI internal combustion engines, as the literature review could not locate any data for comparative testing of AVGAS and natural gas fuels in aviation reciprocating engines.

#### *1.9.4.3 Results and discussion*

Six papers describing the results of tests comparing natural gas emissions with gasoline are shown in Table 46. These results were obtained from comparative charts and were analysed to provide the percentage reductions in emissions.

The tests reported in these papers were conducted using various SI engines at different throttle conditions over a range of engine RPMs. These SI engine test results have been collated and have compared the emissions for gasoline and natural gas fuels. In order to derive common emissions values, this data was analysed and reported in a methodology spreadsheet developed by Williams (2018a). Table 46 shows a summary of this spreadsheet analysis which presents the average percentage difference between natural gas emissions and the gasoline baseline.

Note that Table 46 presents emissions data as two metrics equating to Parts Per Million (PPM) and also g/kw-hr. The accompanying notes describe the relevant conditions and throttle setting associated with the data, with PPM data highlighted in Table 46 and with the corresponding average percentage reductions determined accordingly. Also provided is an average percentage reduction equating to all data surveyed for illustrative purposes only.

Table 46. Natural gas emissions reductions as derived from various references

| | Hydrocarbons (HC) reduction | Carbon Monoxide (CO) reduction | Carbon Dioxide ($CO_2$) reduction | Nitrous Oxides ($NO_x$) increase |
|---|---|---|---|---|
| Jahirul et al. (2010), data from Fig. 5, 6, 8 & 9[1] | 24.4% | 37.6% | 32.9% | 39.9% |
| Aljamali et al. (2014), data from Fig. 6[2] | | 45.3% | 12.8% | |
| Aslam et al. (2005), data from Fig. 11 and Fig. 12[3] | 68.6% | 87.1% | 32.2% | 43.6% |
| Ramjee & Reddy (2011), data from Fig. 7, 8 & 9[4] | 43.6% | 69.7% | 6.3% | |
| Mustafi et al. (2006), data from Fig. 5 & 6[5] | | | 23.3% | **18.8%** |
| Tabar et al. (2017), data from Fig. 4[6] | 58.6% | 81.1% | | 54.1% |
| Kalam et al. (2014), data from Fig. 11, 12 & 13[7] | 70.6% | 72.9% | | |
| **Average (all data)** | -53.2% | -65.6% | -21.5% | -29.7% |
| **Average (ppm-based metrics)** | **-55.6%** | **-69.7%** | **-29.5%** | **+45.9%** |

**Notes**

1. Data based on average of two throttle settings of 50% and 80% throttle. Complete dataset. All data in ppm metrics.
2. Data based on average of two throttle settings of 50% and 100% throttle. Data in % volume metrics.
3. Data based on single throttle setting of 100% throttle. Complete dataset. All data in ppm metrics.
4. Data based on single throttle setting of 100% throttle. All data in g/kw-hr metrics.
5. Data based on single throttle setting of 100% throttle. All data in ppm metrics. Note that $NO_x$ result (**18.8%)** indicates a decrease (reverse trend)
6. Data based on single throttle setting of 100% throttle. All data in ppm metrics.
7. Data based on single throttle setting not stated. All data in ppm metrics.

In general, these data show average percentage reductions in emissions compared to gasoline, as summarised below:

- Hydrocarbons (HC) reduction – 56%
- Carbon monoxide (CO) reduction – 70%
- Carbon dioxide ($CO_2$) reduction – 30%
- Nitrous oxides ($NO_x$) **increase** – 46%

As well as these reductions in emissions as shown above, there are no lead emissions from natural gas fuels as compared to AVGAS fuels.

### 1.9.5 Aircraft operational costs

Apart from the construction costs associated with ground fuelling infrastructure, aircraft operational costs are a significant contributor to Life Cycle Costs (LCC) associated with this LNG modification. As described in Chapter 3, the design space boundary considered in this thesis does not include modification installation and manufacturing costs, as these costs will be similar for both CNG and LNG configurations. Hence the estimation of these costs is not within the scope of this thesis. Nevertheless, the systems LCC given by operation of the aircraft and LNG fuelling station are described here in overview, although detailed breakdown of cost calculations will not be presented.

The aircraft annual operating costs can be estimated using a method described by Gudmundsson (2014), which is based on experiences associated with the actual ownership of a GA aircraft. As described by Gudmundsson (2014), typical inputs to this annual cost model includes annual fuel costs, crew costs, hangarage, annual inspections, engine overhaul, insurance and loan repayments which can then be presented as an annual flight hour cost. The inputs used to predict these flight hour costs for both AVGAS and LNG fuels for a given year are based on relationships defined by Gudmundsson (2014), with an example of these inputs shown by Table 47. LNG fuel prices are derived from appropriate year data shown in Table 48, where the footnote states that average price is \$3.45 per DGE, which has been converted to GGE as per the footnote to Table 47. This LNG fuel price is based on a very small survey size indicating that there is some variability to this price. In addition, this quoted price for LNG is not inclusive of costs for "higher specification" LNG for aviation. It is expected that aviation specification LNG price would be higher than that used for ground transportation applications due to certification cost and fuel quality control overheads.

AVGAS fuel price would also be derived from appropriate data and would be used as an input to Table 47, and to derive annual AVGAS fuel costs for comparison with LNG.

Table 47. Example aircraft operating cost input parameters

| | |
|---|---|
| F1 - Maintenance performed by LAME | 0 |
| F2 - Engine access | 0 |
| F3 - Retractable landing gear | 0.02 |
| F4 - VFR radios installed | 0.02 |
| F5 - IFR radios installed | 0.04 |
| F6 - | 0.01 |
| F7 - | 0 |
| F8 - | 0 |
| Maintenance to flight hours ratio | 0.39 |
| RAP - Hourly rate for LAME | 65 |
| QFLGT - No of flight hours flown annually | 800 |
| | |
| AVGAS price in $US/gallon | $4.26 |
| CNG price in $US/gallon (GGE) | $2.12 |
| LNG price in $US/gallon (DGE) | $3.45 |
| LNG** price in $US/gallon (GGE) | $3.04 |
| BHP cruise | 275.6 |
| SFC cruise - AVGAS | 0.45 |
| SFC cruise - LNG | 0.38 |
| FF cruise - AVGAS - Fuel flow in gallons per hour | 20.75 |
| FF cruise - LNG - Fuel flow in gallons (GGE) per hour | 17.5 |
| | |
| CAC - Insured value of aircraft (unmodified) | $150,000 |
| CAC - Insured value of aircraft (LNG mod fitted) | $292,280 |
| Npp - No of engines | 2 |
| | |
| Ncrew - No of crew (pilot only) | 1 |
| Rcrew - Hourly rate of crew | $60 |
| | |
| Cstor - Hangarage cost per month | $250 |
| | |
| P - Principal amount borrowed - baseline | $100,000 |
| P - Principal amount borrowed - LNG mod | $242,280 |
| I - Interest (%) | 0.05 |
| n - No of pay periods (15 years = 12x15) | 180 |

Gudmundsson (2014)

**NOTES**

** - This LNG price is based on that used for ground transportation. The price for "higher specification" LNG for aviation applications is expected to be higher.

GGE – Gallon of Gasoline Equivalent is a storage measure based on gasoline energy content.

DGE – Diesel Gallon Equivalent is a storage measure based on diesel energy content. Diesel has a higher energy content than gasoline, with 1 DGE = 1.136 GGE.

Table 48. Overall fuel prices for CNG & LNG - 2012

**Table 2. October 2012 Overall Average Fuel Prices on Energy-Equivalent Basis**

| | *Nationwide Average Price in Gasoline Gallon Equivalents* | *Nationwide Average Price in Diesel Gallon Equivalents* | *Nationwide Average Price in Dollars per Million Btu* |
|---|---|---|---|
| Gasoline | $3.82 | $4.26 | $33.10 |
| Diesel | $3.70 | $4.13 | $32.11 |
| CNG | $2.12 | $2.36 | $18.34 |
| Ethanol (E85) | $4.91 | $5.47 | $42.51 |
| Propane | $3.54 | $3.94 | $30.63 |
| Biodiesel (B20) | $3.82 | $4.26 | $33.08 |
| Biodiesel (B99-B100) | $4.32 | $4.82 | $37.47 |

[7] A total of 13 LNG price points were collected with an average fuel price of $2.00 per gallon, or $3.45 per DGE. Because of the small number of price points, this data is not reflected in the table.

US Department of Energy (2012)

### 1.9.6 LNG fuelling station cost

Figure 48 shows the cost of a typical LNG fuelling station as reported in TIAX (2012). The 10,000 gallons LNG storage capacity is required to support commuter category aircraft fleet operations. This has been previously determined in the morphological analysis described earlier in this Section. This storage capacity equates to $US 550,000 cost in 2010-year dollars as shown in Figure 48. This equates to approximately $US 580,000 out-turned to year 2012. This LNG fuelling station cost includes site selection, land costs, engineering design, project management, station construction, station commissioning and associated quality inspections. It should be noted however that these stations costs do not include aviation related safety certification and accreditation costs to authorise siting of this station within the confines of an airport. Aviation safety certification and accreditation requirements may result in significant additional costs which cannot be predicted at this time.

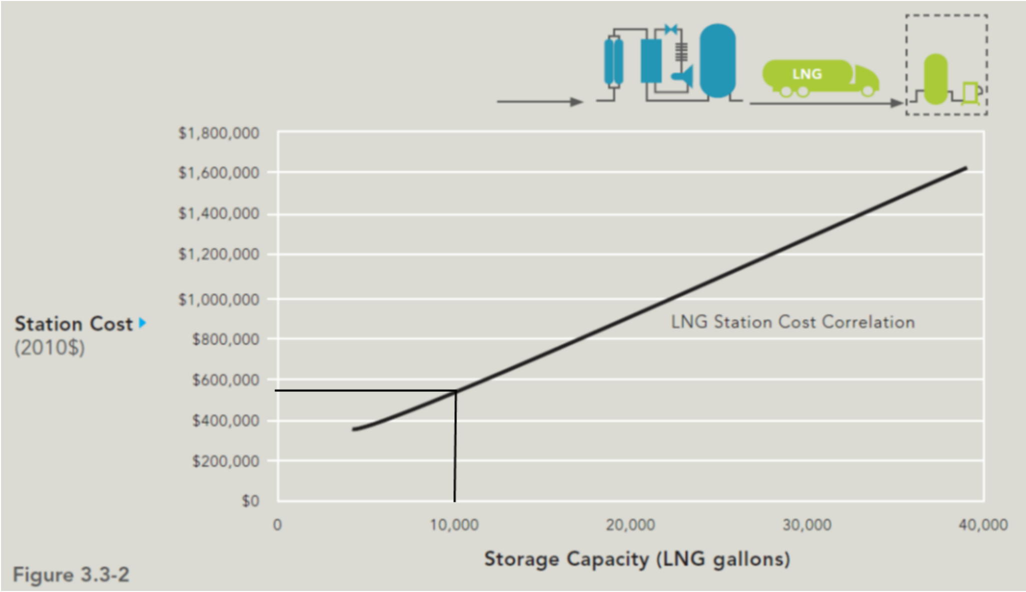



Figure 48. LNG station cost as a function of storage capacity
TIAX, (2012)

# Appendix 2. NG fuels case study – supporting analysis

## 2.1 OVERVIEW

Quantisation of the morphological matrix is a key component of the matrix-based conceptual design methodology. This section provides an overview of the analysis supporting quantisation of those metrics associated with the aircraft and ground fuelling segments. As described in Appendix 1, these metrics are derived from the respective QFD matrices for the aircraft and ground fuelling segments. Specifically, this quantisation of metrics is characterised by simplified relationships addressing CNG and LNG cylinder weight/cost, CNG/LNG fuel tank(s) installed drag, fuelling station infrastructure cost, storage and fill characteristics.

In addition, this section also describes the analysis supporting the range estimation for a particular natural gas concept selected from the quantisation process described above. Although this analysis is not directly part of the matrix quantisation, it is included here as it contains relatively detailed calculations that compare tank installation drag predictions, and the associated impact on aircraft range.

## 2.2 AIRCRAFT SEGMENT QUANTISATION METRICS

### 2.2.1 Fuel tank weight and cost estimates

An analysis was undertaken on CNG and LNG fuel tank weight and cost characteristics by surveying commercially available data and information provided in product specifications, installation and maintenance manuals, and industry journals. This data and information then was used to develop tank weight and cost metrics as described in the following sections.

#### *2.2.1.1 CNG cylinder analysis*

This analysis of cylinder weight and cost metrics was based on data gathered on commercial CNG fuel tanks used in ground transportation applications (i.e. for automobiles and trucks). No commercial data exists for CNG cylinders used in aviation applications, although it is noted that similar fuel tanks have been fitted to small general aviation aircraft as prototype demonstrators. These prototype CNG tank installations have not been certificated nor have entered production. Chapter 3 of this thesis provides examples of CNG fuel tank technology applied to aviation.

This analysis therefore supports the natural gas case study shown at Appendix 1, where simple relationships were developed to characterise cylinder weight and cost as quantisation metrics. The analysis surveyed CNG cylinder design specifications, product manuals, journal articles and maintenance information for a range of cylinder types and technologies. The weight and cost data were tabulated and processed to provide average weight and cost for each CNG cylinder type, which was then presented as metrics similar to that reported by Trudgeon (2005). These metrics were expressed in terms of weight/litre volume as well as cost/litre volume.

Figure 49, Figure 50 and Figure 51 show the weight and cost metrics for CNG cylinder types and also the weight characteristics for various cylinder types. The underpinning analysis is provided in a methodology spreadsheet developed by Williams (2018a), which cross referenced the source data with the data presented in Figure 49, Figure 50 and Figure 51. This data was derived from various sources as described above, and included:

- Sinor (1991), which compared CNG and LNG technologies for transportation applications.
- Quantum Technologies (2014), which was a product brochure providing CNG cylinder general specifications.
- Quantum Technologies (2015), which was a product brochure that detailed high pressure lightweight CNG Type IV cylinders.
- 3M CNG Tanks (2013), which was a technical data sheet for CNG tanks manufactured by the 3M company.
- GoCleanNG (2012), which was a product brochure for Magnum CEL fuel storage system.
- Chris Red (2014), which was a Journal article that discussed natural gas pressure vessels for alternate fuels.
- Owens Corning (2009), which was a technical presentation that described innovation in composite CNG cylinders.
- SkyCNG (2015), which was an online technical resource describing

SkyCNG CNG cylinders.

- Go Natural CNG (2015), which was an online technical resource describing Go Natural CNG cylinder tanks.
- G-Stor Pro (2015), which was an online technical resource describing G-Stor Pro tanks for natural gas vehicles.

This data was collated in the design methodology spreadsheet by Williams (2018a) and processed to provide weight metrics corresponding to the cylinder Type as referred in Table 49.

Figure 49 presents these weight metrics showing a decreasing weight trend for increasing cylinder Type. It is noted however at time of writing that the 'newer' Type V cylinders was not an official category with this data determined from two sources being GoCleanNG (2012) and the Composites World journal article by Chris Red (2014). This particular Type V tank configuration was rated to a service pressure of 4500 psi, whereas the Type III and Type IV were rated at lower operating pressures of 3000 psi and 3600 psi respectively. Therefore, it follows that the newer Type V cylinder configurations will have additional performance advantages over the Type IV cylinders due to potential increase in capacity and lighter weights. However, at time of writing these Type V cylinder were prototypes only, and hence were not commercially available.

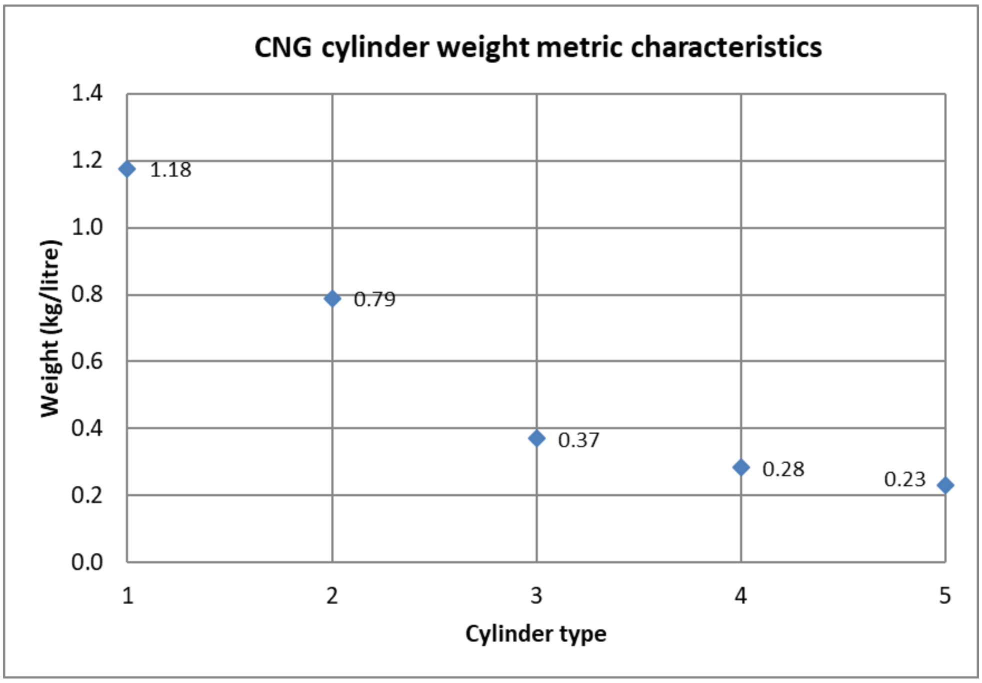



Figure 49. Derived CNG cylinder weight metrics as function of cylinder type

Cost metrics were determined in a similar way to weight metrics. Figure 50 shows CNG cylinder cost metrics derived from the data sources as described above. This cost metric curve shows an increasing cost trend for increasing cylinder Type. However, it should be noted that costs for the newer Type V tank technology could not be obtained given that this technology was at the prototype stage at time of writing. Therefore, it was necessary to extrapolate cost metrics of the Type V cylinder in order to complete this analysis. A curve was fitted to existing Type I through Type IV cylinder data and a cost metric for a Type V cylinder was approximated as shown by Figure 50. This approach was bounded by considerable uncertainty as cost data for CNG cylinders was more difficult to obtain compared to weight data discussed earlier. Given that the Type V cylinders were prototypes, then it is expected that this predicted cost could be considered as a lower bound until these cylinders are produced in commercial quantities.

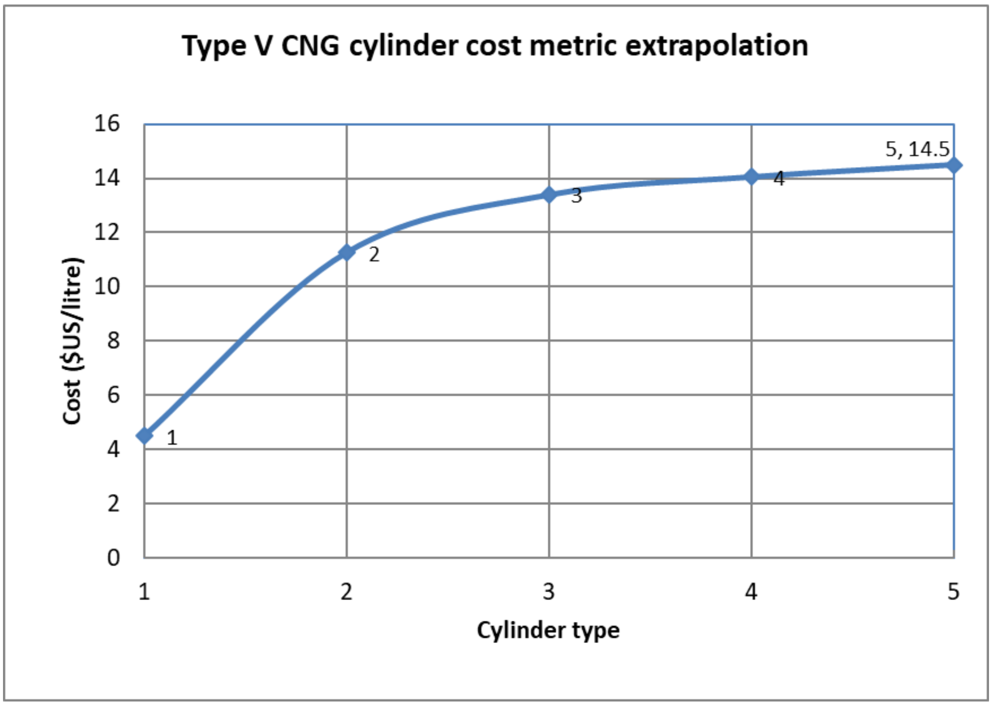



Figure 50. Chart showing CNG cylinder cost metrics extrapolated for a Type V tank

Figure 51 shows the trend in empty CNG cylinder weight for various cylinder capacities. As can be seen, there is a general linear trend for increasing cylinder weight as a function of the cylinder capacity. However, there is some scatter in data extrapolated to zero-cylinder capacity due to 'variation' in quoted weights derived from the various manufacturers and suppliers.

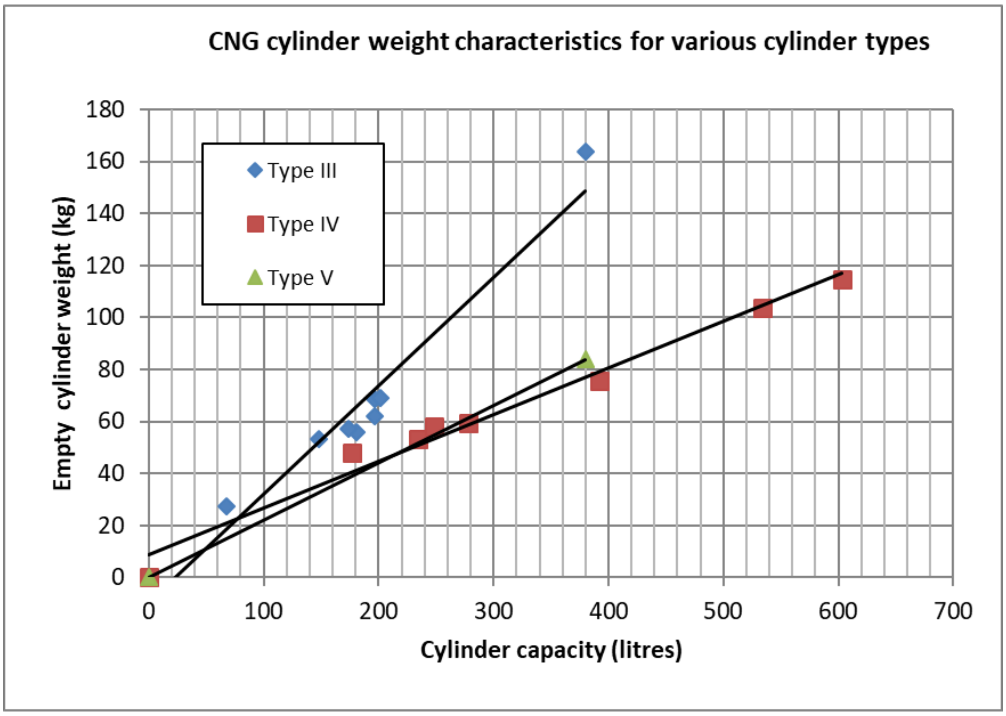



Figure 51. CNG cylinder tank empty weight characteristics as a function of capacity

Table 49 provides a summary of the derived weight and cost metrics for CNG types as described in the preceding sections. These metrics were used in quantisation of each element of the morphological matrix to determine a CNG cylinder weight and cost for a particular tank type and capacity. This approach provides a simple yet effective way to quantify CNG tank performance for these metrics and facilitates quick comparison across different installation locations.

Table 49. Summary of weight and cost metrics for various CNG cylinder types

| | | Empirical weight metric (kg/litre) | Empirical cost metric ($US/litre) |
|---|---|---|---|
| **CNG Type I** | All metal construction, generally steel | 1.18 | 4.50 |
| **CNG Type II** | Mostly steel or aluminium with fibre reinforced polymer overwrap in the hoop direction, featuring glass, carbon, or basalt fibre; the metal vessel and wound composite materials share about the same structural loading. | 0.79 | 11.26 |
| **CNG Type III** | Metal liner (typically aluminium) with full carbon fibre composite overwrap; the composite materials carry the structural loads | 0.37 | 13.39 |
| **CNG Type IV** | Metal free construction. A carbon fibre or hybrid carbon/glass fibre composite is filament wound over a thermoplastic polymer liner; the composite materials carry the load | 0.28 | 14.05 |
| **CNG Type V** | An all composite construction. The vessel is liner-less and features a carbon fibre or hybrid carbon/glass fibre composite wound over a collapsible or sacrificial mandrel; the composite materials carry the load. | 0.22 | 14.50 |

#### *2.2.1.2 LNG cylinder analysis*

This analysis of cylinder weight and cost metrics was mainly based on data gathered on commercial LNG fuel tanks used in heavy vehicle ground transportation applications (i.e. heavy haul trucks). No commercial data exists for LNG cylinders used in aviation applications, although it is noted that similar fuel tanks have fitted to large commercial transport aircraft as a prototype demonstrator (Kaminski-Morrow 2008).

Carson et al. (1980) was a final report describing the outcome of an extensive design and analysis study which investigated the potential of LNG as an alternate fuel for subsonic transport aircraft. This study focused on applications on LNG as an alternate fuel for Jet A powered transport aircraft. It set out to (1) determine a suitable LNG fuelled aircraft configuration, (2) provide a concept and structural analysis of LNG cryogenic fuels tanks, (3) establish fuel system configuration and functional requirements, (4) screen the most likely insulation materials for fuel tanks, and (5) determine airport ground facility requirements. This paper provided an extensive account of the various options for structural design layout and design details of LNG fuel tanks for jet transport aircraft. The LNG pylon fuel tank arrangement which has a similar geometrical profile to that employed by a wing tip tank considered in this study. In particular it proposed several design options for internal and external LNG fuel tank configurations based on a lightweight composite structure. These fuel tanks consisted of a sandwich structural layout comprising a 2219 aluminium inner tank with a foam thermal insulation layer and an outer composite skin fairing. This LNG fuel tank configuration had a significant advantage in that it was extremely lightweight in comparison to the stainless-steel LNG cryogenic tanks used on heavy haul trucks.

Carson et al. (1980) presented detailed weight estimates for these LNG fuel tanks, and this study has used these weight figures as a basis for estimates for lightweight LNG fuel tanks for small aircraft. Option 3 as presented by Carson et al. (1980) comprised an over-wing pylon mounted fuel tank installation which used the sandwich construction technique. This structural layout predicted an insulation weight of this configuration to be 254 kg, with a total fuel tank volume of 43.3 $m^3$ (approximately 43300 litres).

It was noted that Carson et al. (1980) did not attempt to characterise costs apart from initial costs and direct operating costs (DOC) associated with the new LNG-powered aircraft design.

This analysis therefore supports the natural gas case study shown at Appendix 1, where simple relationships were developed to characterise cylinder weight and cost as quantisation metrics. The analysis surveyed LNG cylinder design specifications, product manuals, journal articles and maintenance information for a range of existing heavy haul vehicle transportation cylinder types and technologies. The weight and cost data were tabulated and processed to provide average weight and cost for each LNG cylinder type, which was then presented as metrics similar to that adopted for the CNG cylinders. These metrics were expressed in terms of weight/litre volume as well as cost/litre volume.

Figure 52 and Table 50 show the weight and cost metrics for LNG cylinder types. The underpinning analysis is provided in a methodology spreadsheet developed by Williams (2018a), which cross referenced the source data with the data presented in

Figure 52 and Table 50. This data was derived from various sources as described above, and included:

- Carson et al. (1980), a NASA report which analysed various options for structural design layout and design details of LNG fuel tanks for jet transport aircraft.
- CryoDiffusion, (2015), which was an online product brochure that detailed cryogenic cylinders for LNG for Taxi, Buses and Trucks.
- Go With Natural Gas, (2014), which was an online resource for LNG tanks and fuel Systems - LNG Storage prepared by Natural Resources Canada.
- Taylor Wharton, (2004, 2008), which were LNG vehicle fuel tank – brochures outlining the specifications and physical characteristics of the range of LNG fuel tanks.
- Sinor (1991), which compared CNG and LNG technologies for heavy haul transportation applications.
- Zhongyou Tongyong Luxi Natural Gas Equipment, (2015), which was a product catalogue that detailed cryogenic LNG cylinders and high vacuum insulation.

These data were collated in the design methodology spreadsheet by Williams (2018a) and processed to provide weight metrics as described in Figure 52. It should be noted however that unlike the CNG tank data, reliable LNG tank weight and cost information was difficult to obtain because of the considerable variation in tank configurations and layouts. Furthermore, most data obtained for LNG cryogenic tanks were applicable to heavy haulage trucks comprising stainless-steel construction. These stainless-steel LNG tank configurations were considerably heavier as seen in Figure 52, which was not ideal for aviation applications. For this reason, these data were used as baseline from which to compare the light weight solutions as studied by Carson et al. (1980).

Figure 52 shows the trend in empty LNG cylinder weight for various cylinder capacities. As can be seen there is a general linear trend for increasing cylinder weight as a function of the cylinder capacity. However, there is some scatter in data extrapolated to zero-cylinder capacity due to ‘variation’ in quoted weights derived from the various manufacturers and suppliers. Given that the Carson et al. (1980) tanks were specifically developed for aviation applications then this data was used in LNG tank weight metrics. It is noted however that the weight estimates presented in Carson et al. (1980) were based on a design study, rather than actual measurements made from a prototype or concept demonstrator. This data when extrapolated to the scale involved here for small aircraft predicted tank empty weights of the order of 13 kg for a 250-litre capacity tank by considering the quoted insulation weight, predicting the aluminium liner weight and applying de-scaling factor of 2 for conservatism. This analysis was undertaken in the design methodology spreadsheet by Williams (2018a).

Although this predicted weight is low, it should be noted that this equates to a state-of-the-art composite structure as described by Carson et al. (1980). Furthermore, this weight equates to the tank structure only with no allowances made for LNG fairings, or attachment structure. An allowance has been made for these ancillaries, and this resulted in an additional 14.8 kg for a typical 250 litre capacity LNG tank installation as detailed in design methodology spreadsheet by Williams (2018a) and was processed to provide the LNG weight metric used in the quantised morphological matrix.

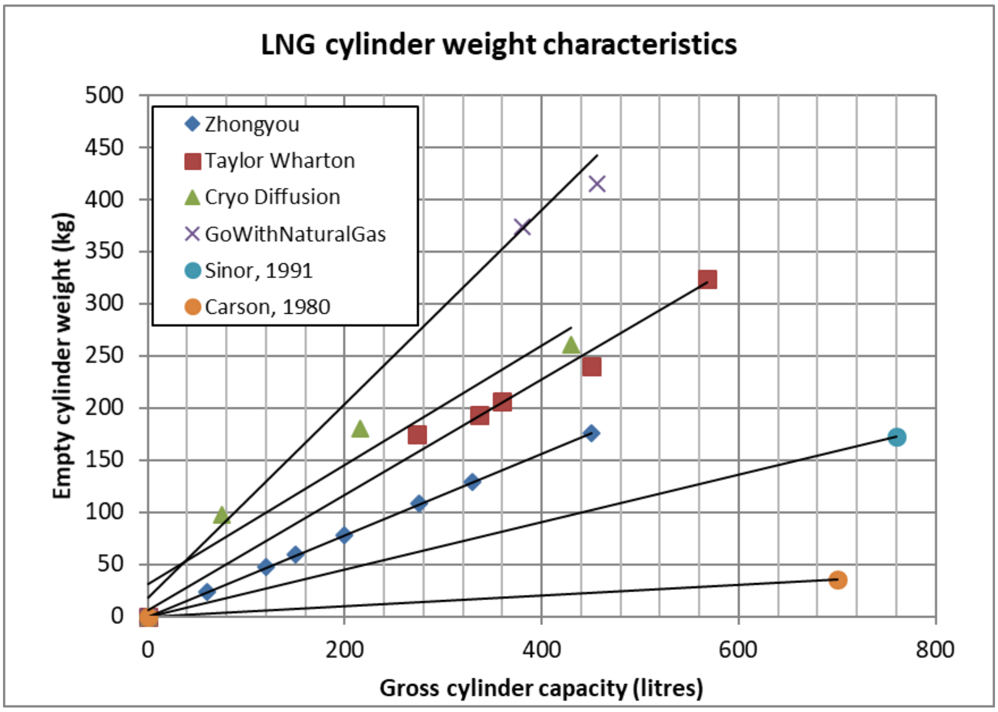



Figure 52. LNG empty weight characteristics as a function of gross cylinder capacity.

Table 50 provides a summary of the derived cost metrics for LNG types as determined by data presented by references in this table. These metrics were used in quantisation of each element of the morphological matrix to determine a LNG cylinder cost for a particular tank type and capacity. The data used for LNG cost metrics equates to that provided by Iuzzolino (2004) for the Dynetek tanks, as this approximated the capacity and configuration of tank utilised in this case study This approach provides a simple yet effective way to quantify LNG tank performance for these metrics and facilitates quick comparison across different installation locations.

Table 50. LNG cylinder costs

| LNG cylinder costs | | | | |
|---|---|---|---|---|
| **Reference** | **Manufacturer** | **Capacity (litres)** | **Cost $US** | **Cost $US/litre** |
| Iuzzolino (2004) | Chart Industries | 177 | 8000 | **45.2** |
| Iuzzolino (2004) | Dynetek | 318 | 7400 | **23.3** |
| Argonne Lab (2013) | Westport* | 720 | 90000 | **125.0** |

* This cost is for a full fuel tank installation/conversion of a truck

### 2.2.2 Fuel tank drag estimates

The Hoerner method described as follows is based on fluid dynamic drag information and data presented in Hoerner (1965). This method can be applied quickly, and it relies on the derivation of data from graphs presented within Hoerner (1965) which were derived from wind tunnel test or flight test data. It is therefore an ideal method to quantise aerodynamic drag effects as described in Appendix 1. However, it must be stated that the Hoerner method is an approximation, with further modelling undertaken to refine these drag estimates as shown later.

There are two sets of drag calculations, relating to (1) the wing tip tank installation, and (2) the fuselage belly tank installation. These are presented below.

#### *2.2.2.1 Wing tip tank drag contribution*

This method predicts the wing tip tank drag contribution as a result of the installation of modified wing tip fuel tanks. The natural gas modification replaces the existing streamlined profile Cessna 421B wing tip tanks with a streamlined cylindrical cross section wing tip tank configuration as shown by Figure 53. Therefore, this method comprises two main steps to evaluate the change in airframe drag as a result of removal of the existing streamlined wing tip tanks, and replacement with less streamlined (higher drag) cylindrical natural gas fuel tanks.

*Drag contribution of natural gas wing tip tank installation* - This analysis step predicts drag coefficient by first estimating the skin friction, then the base drag contribution using the approaches as described by Hoerner (1965). The method assumes that the body shape of the natural modified wing tip tank is similar to the body-of-revolution streamline shapes that have been wind tunnel tested previously. Although the data presented in Hoerner (1965) is generally similar, it is not the exact shape and geometry of the wing tip fuel tank considered here. Therefore, it should be appreciated that this method can be considered as approximate only. However, this is satisfactory for conceptual design purposes as described in this thesis.

It is important to note that this analysis relates to the CNG fuel tanks and therefore the values associated with LNG fuel tanks will vary due to small differences in tank dimensions. The quantisation of these drag contributions therefore takes account of these differences and are hence represented in the relevant morphological matrices discussed in Appendix 1.

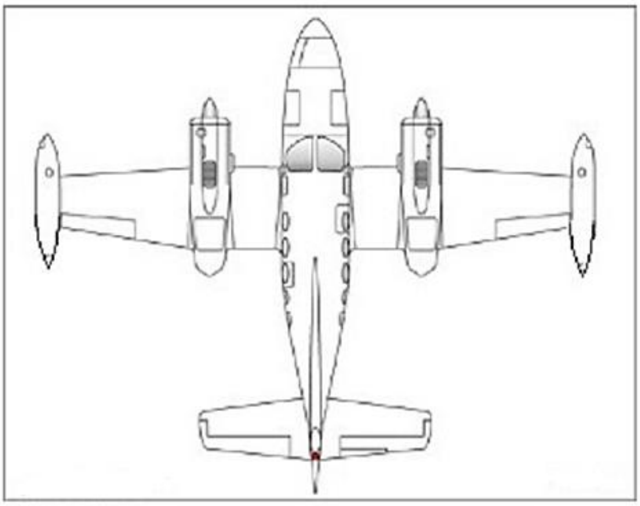


Figure 53. Cessna 421B with natural gas wing tip tank modification

As stated above this method derived the skin friction drag component based on wetted area, and then derives the base drag coefficient. In order to use this method several parameters relating to geometry and flight Reynolds number are calculated.

For the natural gas wing tip fuel tank, the following parameters apply:

| | | | |
|---|---|---|---|
| Length | $l$ | = | 10 ft or 3.05 m |
| Diameter | $d$ | = | 1.98 ft or 0.604 m |
| Therefore | $l/d$ | = | 5.04 |

Based on the Cessna 421B owner's manual data assume that the design airspeed is 200 KCAS or 337.6 fps

| | | | |
|---|---|---|---|
| Air density | $\rho$ | = | 0.001496 lb/ft$^3$ at 15000 ft |
| Air viscosity | $\mu$ | = | $0.0343\times10^{-5}$ lb/ft.sec |

Reynolds No. based on length, $R_l = \dfrac{\rho v l}{\mu}$ Equation 25

$$R_l = \frac{0.001495 \times 337.6 \times 10}{0.0343\times10^{-5}}$$

$$R_l = 1.47\times10^{7}$$

Refer Hoerner (1965), reproduced here as Figure 54 (Figure 22 page 6-16) for a $l/d$= 5.04 and $R_l$= $1.47\times10^7$. The figure shows that the value of $C_{Dwet}$ cannot be estimated accurately. However, an estimate is provided by:

Wetted area drag coefficient $C_{Dwet}$ = 0.0028

Based on Hoerner (1965) page 6-18, the equation to determine wetted area is presented as follows:

Wetted area $S_{wet} = 0.75 \times l \times \pi \times d$ Equation 26

$$S_{wet} = 0.75 \times 10 \times \pi \times 1.98$$

$$S_{wet} = 46.3\text{ft}^2$$

Tank reference area $S_{ref} = 3.1\ \text{ft}^2$ as before

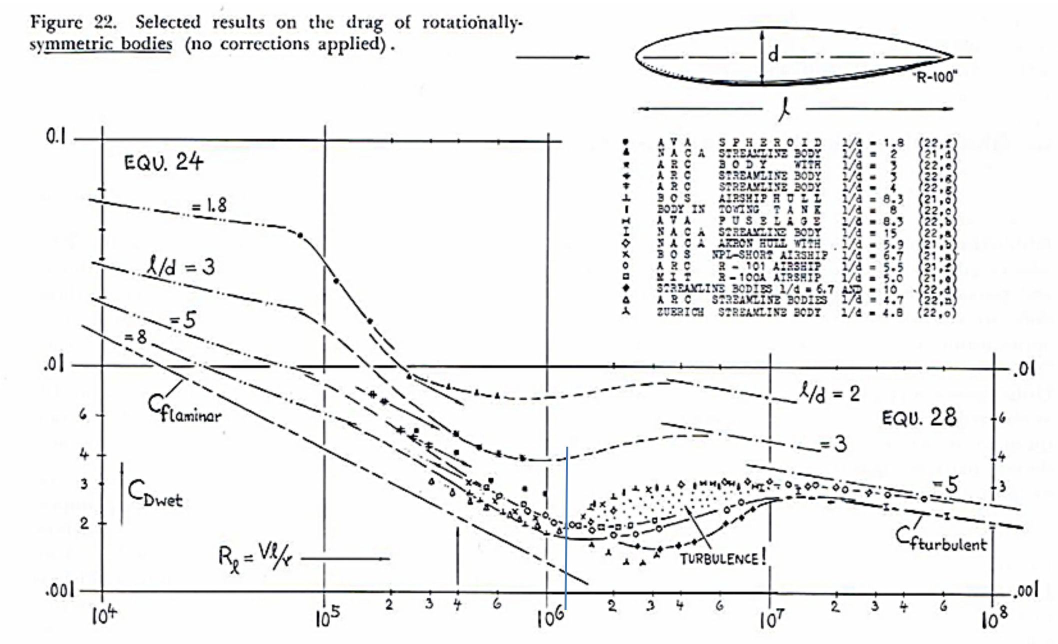



Figure 54. Extract from Figure 22 of Hoerner (1965)

The natural gas fuel tank drag coefficient $C_{Df} = C_{Dwet} \times \left(\frac{S_{ref\_wet}}{ref\_tank}\right)$ Equation 27

$$C_{Df} = 0.0028 \times \left(\frac{46.3}{3.1}\right)$$

$$C_{Df} = 0.042$$

The contribution due to base drag can now be calculated as follows:

Base area $S_b = \dfrac{\pi \times d^2}{4}$ Equation 28

$$S_b = \frac{\pi \times 0.66^2}{4}$$

$$S_b = 0.344\ \text{ft}^2$$

Therefore, in order to evaluate the empirical function as shown in Hoerner (1965), and reproduced as Figure 55, the following approach can be used.

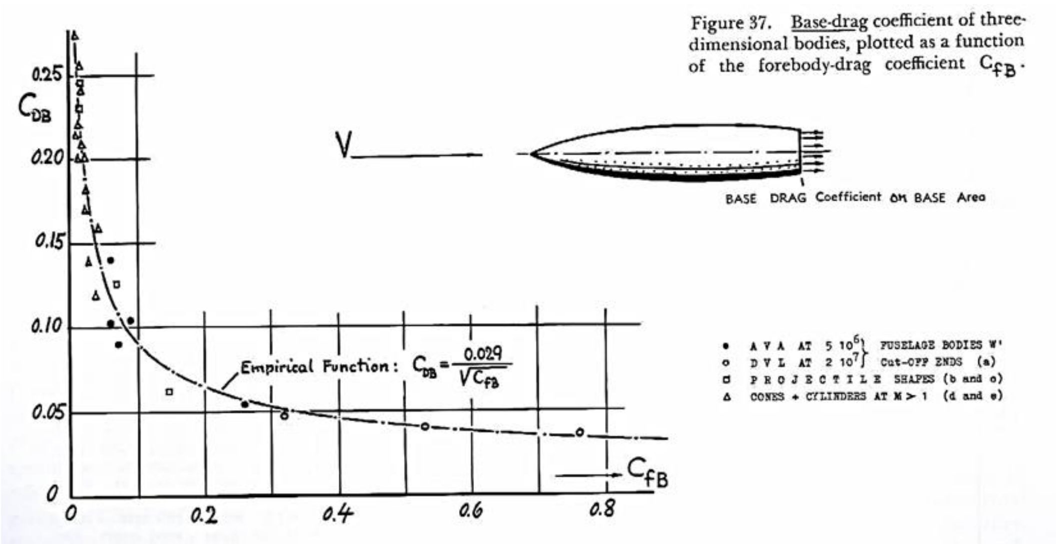



Figure 55. Extract from Figure 37 of Hoerner (1965)

Base drag coefficient $\quad C_{Db} = \dfrac{0.029}{\sqrt{C_{fb}}}$ $\quad$ Equation 29

Based on Hoerner (1965) equation 33, page 3-19 to 3-20:

Base drag friction coefficient $\quad C_{fb} = \dfrac{C_f S_{wet}}{S_b}$ $\quad$ Equation 30

$$C_{fb} = \frac{0.0028 \times 46.3}{0.344}$$

$$C_{fb} = 0.344$$

And therefore $\quad C_{Db} = \dfrac{0.029}{\sqrt{0.344}}$

$$C_{Db} = 0.047$$

Therefore, the total fuel tank drag coefficient can be determined by:

$C_{Dtank} = C_{Df} + C_{Db}$ $\quad$ Equation 31

$$C_{Dtank} = 0.042 + 0.047$$

$$C_{Dtank} = 0.089$$

Based on the Cessna 421B owner's manual data, the Cessna 421B wing reference area is determined as $S_{ref} =$ 215 ft$^2$

The fuel tank drag coefficient $\quad C_{Dtank} = C_D \times \left(\dfrac{S_{ref\_tank}}{S_{ref}}\right)$ $\quad$ Equation 32

$$C_{Dtank} = 0.089 \times \left(\frac{3.1}{215}\right)$$

$$C_{Dtank} = 0.0013$$

This is the drag coefficient for one tank. Therefore, for two fuel tank installation this drag coefficient becomes:

Two CNG wing tip tank installations $C_{Dtank} = 2\times0.0013$

$C_{Dtank} = \underline{0.0026}$

*Drag contribution of streamlined AVGAS wing tip tank installation* - This analysis step predicts drag coefficient of the installed streamlined wing tip installation by using the method presented by Torenbeek (1982). Torenbeek (2005) states that a typical value for wing tip tanks of $\Delta(C_DS)$ = 0.055 times the tank frontal area should be used.

The wing tip tank frontal area can be estimated from the Cessna Model 421B three view drawing, as follows:

$S_t$ = 3.06 $ft^2$ for one tank

Therefore $\Delta C_D = \frac{0.055}{3.06}$ Equation 33

$\Delta C_D = 0.0179$

Based on the Cessna 421B owner's manual data the Cessna 421B wing reference area is determined as $S_{ref}$ = 215 $ft^2$

The drag coefficient of one tank $C_{Dtank} = \Delta C_D \times \left(\frac{S_{ref\_tank}}{S_{ref}}\right)$ Equation 34

$C_{Dtank} = 0.0179 \times \left(\frac{3.06}{215}\right)$

$C_{Dtank} = 0.000256$

This is the drag coefficient for one tank. Therefore, for two fuel tank installation this drag coefficient becomes:

Two streamlined wing tip tank installations $C_{Dtank} = 2\times0.000256$

$C_{Dtank} = \underline{0.0005}$

Therefore, the net drag contribution of the natural gas wing tip tanks can be calculated by:

$\Delta C_D$ = *Wing tip fuel tank drag coefficient added – Wing tip drag coefficient removed*

$\Delta C_D = C_{Dtank\_A} - C_{Dtank\_R}$ Equation 35

$\Delta C_D = 0.0026 - 0.0005$

$$\Delta C_D = \underline{0.0021}$$

This relationship is applied within the morphological matrix quantisation in combination with other drag contributions as a result of other fuel tank installations (i.e. fuselage belly tank – discussed below).

#### *2.2.2.2 Fuselage belly tank drag contribution*

This method predicts the drag coefficient of the natural gas fuselage belly tank installation that is geometrically similar to the installation shown by Figure 56. The natural gas belly tank is installed beneath the fuselage in an arrangement similar to cargo pod which are commonly installed on other GA aircraft. The impact of drag is analysed using a method as described by Hoerner (1965). This method can be applied quickly, and it relies on the derivation of data from various graphs found within Hoerner (1965). Figure 56 shows the natural gas belly tank modification installed beneath the fuselage. Note that the natural gas fuel tank is cylindrical in shape and is therefore covered by a lightweight fairing to reduce aerodynamic drag.

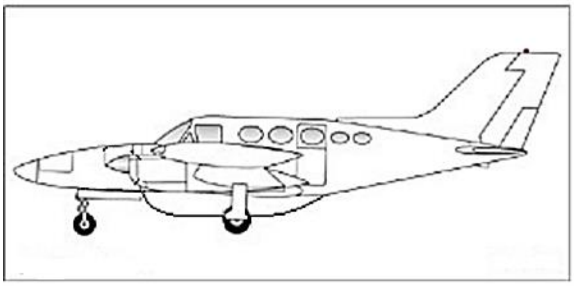


Figure 56. Cessna 421B with natural gas fuselage belly tank modification

Again, it is important to note that this particular analysis relates to a CNG fuel tank installation and therefore the values associated with LNG fuel tank installation will vary due to small differences in tank dimensions.

This analysis predicts drag coefficient by estimating super-velocity and negative pressure impacts on the tank, using the approaches as described by Hoerner (1965). As described earlier, the aircraft parameters associated with this analysis are as follows:

The design airspeed $v$ is 200 KCAS or 337.6 fps

Air density $\rho$ = 0.001496 lb/ft$^3$ at 15000ft

Hoerner (1965) states that the given the belly tank location shown by Figure 57, a $C_p$ = -0.1 and a pressure differential $\Delta P_x/q$ = 0.2, a drag coefficient is obtained in the order of $C_{Do}$ = 0.1. Hoerner (1965) states that the actual drag obtained from flight tests is larger than this predicted drag coefficient.

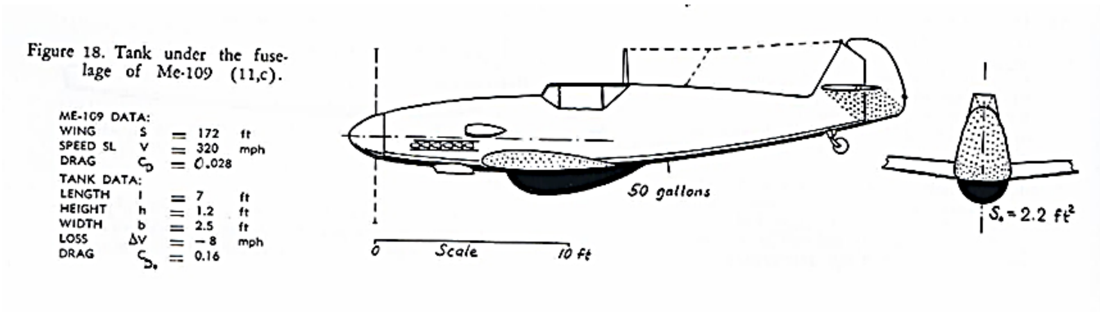



Figure 57. Extract from Figure 18 of Hoerner (1965)

It is therefore concluded that super-velocity and negative pressure transferred to the tank from the wing roots to the location of the tank beneath the fuselage are to be considered.

Assuming that the aircraft is operating at a $C_L$ = 0.15, the Hoerner (1965) method indicates that based on flight tests, the pressure on the on the lower side of the wing section is in the order of $C_p$ = -0.4, due to thickness ratio. Furthermore Hoerner (1965) states that assuming that half of this value to be transferred to the location of the tank, the pressure ratios can now be estimated thus:

$$C_p = (-0.1 - 0.2) = -0.3, \text{ and } \Delta P_x/q = 0.2 + 0.2 = 0.4$$

Assuming that this fuel tank geometry $h/x$ = 9%, Hoerner (1965) states that Figure 58 provides a $C_{Do}$ = 0.08, based on 1.3×1.4×0.08 = 0.15 is obtained.

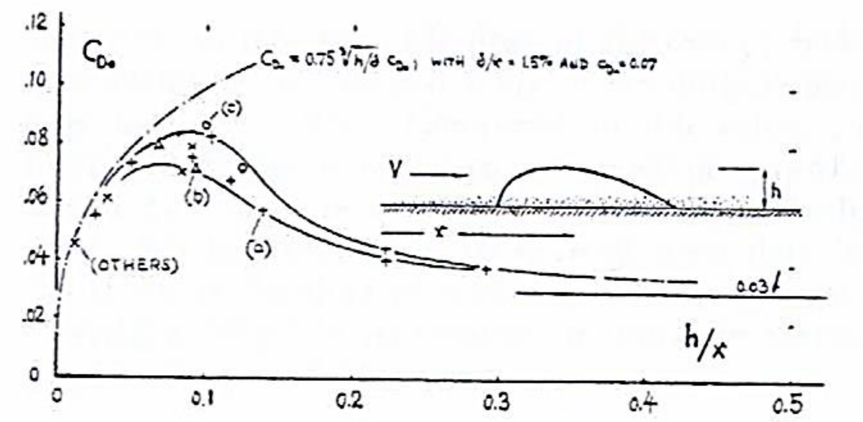



Figure 9. Drag of streamline half bodies placed on a plane wall as a function of their height ratio (6).

Figure 58. Extract from Figure 9 of Hoerner (1965)

As stated in Figure 59 from Hoerner (1965) the drag of a similar belly tank configuration on a fighter type aircraft is $C_{Do}$ = 0.16.

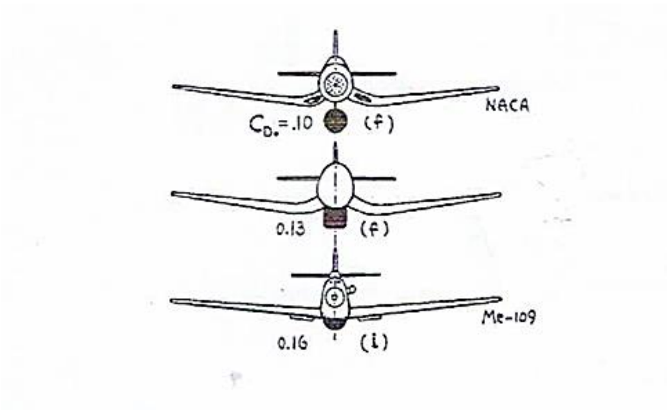



Figure 43. Examples of external tanks under the fuselage of low-wing airplanes (30).

Figure 59. Extract from Figure 43 of Hoerner, (1965)

As before the Cessna 421B wing reference area $S_{ref}$ = 215 ft$^2$

Tank reference area is estimated by $S_t$ = 1.77 ft$^2$ based on a height and width of approximately 1.3 ft.

Based on aircraft reference area the $C_D$ of the fuel tank can be calculated:

The belly tank drag coefficient $C_{Dtank} = C_D \times \left(\frac{S_{ref\_tank}}{S_{ref\_Cessna}}\right)$ Equation 36

$$C_{Dtank} = 0.16 \times \left(\frac{1.77}{215}\right)$$

$$C_{Dtank} = \underline{0.0012}$$

This value of belly tank drag coefficient is applied within the morphological matrix quantisation in combination with any other drag contributions as a result of other fuel tank installations and is accounted for accordingly.

## 2.3 NG FUEL STATION QUANTISATION

As discussed earlier, metrics associated with natural gas fuelling infrastructure have been derived from requirements defined in the ground fuelling infrastructure QFD matrix. These metrics relate to the infrastructure costs; and storage and fill characteristics.

## 2.4 PERFORMANCE CHARACTERISTICS ESTIMATION

### 2.4.1 Overview

In section 2.2.2 a method was presented that estimated the drag contribution of each fuel tank installation option using Hoerner (1965). This method was used to quantify drag estimates to enable selection of a conceptual design concept in combination with other metrics.

The range performance of any aircraft is determined by the aircraft Lift/Drag (*L/D*) ratio and variables such as the Specific Fuel Consumption (*SFC*) characteristics, fuel loading fraction and propeller efficiency. Of these parameters, the drag contribution caused by the installation of an additional external fuel tank will have a dominant effect on range performance with other variables being equal. This analysis assumes that the change in drag caused by the installation of the belly tank adversely impacts drag, with a much smaller impact on lift.

On this basis, this section sets out to verify estimates of drag which were used to determine range performance. The approach has been to compare predictions of fuel tank installation drag contribution with those predicted using the Da Vinci Technologies (2004) *Airplane PDQ* software, and also data extracted from the Cessna Aircraft Company (1974), C421B aircraft owner's manual. In addition, traditional analysis methods using the Breguet's range equation were used to calculate aircraft range based on predictions of *L/D* ratios using the modified drag data. This was undertaken for both the unmodified and modified aircraft configurations.

In order to verify the accuracy of the drag estimates, two steps were undertaken involving comparisons of drag for (1) the unmodified aircraft configuration (i.e. no fuselage belly tank) and (2) the LNG tank modified configuration which includes a belly tank.

A number of software-based tools were used to undertake this analysis, which are discussed in the following sections.

### 2.4.2 Tools

#### *2.4.2.1 Airplane PDQ*

The Da Vinci Technologies (2004), *Airplane PDQ* software package is a conceptual design tool for general aviation aircraft. *Airplane PDQ* generates several detailed analysis reports including: Performance charts, Trim prediction, Drag Breakdown, Weight estimates, Weight and Balance analysis, CG limits, Design summary, and a Design check. It is uncertain as to the actual methods employed to predict aerodynamics and performance reports, although a discussion with the software developer indicated that it applied traditional component build-up techniques. These techniques are common and consistent with other tools of the same generic type such as those given by the Naval Surface Weapons Center (NSWC) aeroprediction code NSWC, (1994, 1995). Given the uncertainties associated with the underpinning theory, *Airplane PDQ* has been found to provide conservative predictions of drag and range performance given its intended purpose as a conceptual design tool. The authors previous work using this tool has compared the results of the Hoerner (1965) methods, *Airplane PDQ* and the *NSWC Aeroprediction Code AP95* for a wing pylon mounted "pod" installation on a propeller-driven military aircraft. The freestream drag estimates for this pod installation have been compared as reported by Williams (2004). This analysis showed that the total installed drag count figures to be within ±15% for freestream conditions. This same study by Williams (2004) shows that range predictions for the "clean" (no pod installation) and the carriage condition (pods mounted to wing pylons) are conservatively predicted with differences being approximately 1%. It was noted by Williams (2004) that this difference in range was difficult to rationalise in real terms, as other factors such as changing winds, flight profiles and engine performance would mask the effect of the actual additional drag increment. To this extent *Airplane PDQ* was not used to predict range in this analysis.

However, *Airplane PDQ* has one main advantage in that data from the *X-Plane* flight simulation model can be imported directly into this software package and used to provide a breakdown of drag components. This also facilitates development of an accurate aircraft geometry model, and also has a secondary benefit in that the *X-Plane* flight model can be used to validate general flight performance and flying qualities for gross modelling errors or oversights.

#### *2.4.2.2 X-Plane modelling*

The Laminar Research Inc (2003) *X-Plane* flight simulator is a general-purpose software package which in this instance was used primarily used to model aircraft geometry. As a flight simulator it could also be used to model flight performance and flying qualities. This package was divided into four modules with this study using three modules being the *X-Plane* flight simulator, Plane Maker and Airfoil maker modules only. The Laminar Research Inc (2003), X-Plane simulation software has been previously certificated for use in single-engine and twin-engine flight simulators by the US Federal Aviation Administration (2002).

An X-plane model of the Cessna C421B aircraft was prepared using data provided in the Cessna Aircraft Company (1974), C421B aircraft owner’s manual and also data provided in Janes All the Worlds Aircraft by Taylor et al. (1983-84). The actual numerical input details of the Cessna 421B model is not presented in this study, although an example of the flight model is illustrated in Figure 60.

Figure 60. X-Plane flight model of the Cessna 421B aircraft showing LNG belly tank

Figure 60 shows the LNG fuel tank installation beneath the aircraft fuselage. This LNG fuel tank is covered by a lightweight aerodynamic fairing which is approximately 11.7 ft (3.57 m) in length and protrudes below the fuselage 1.5 ft (0.457 m). The general profile of this aerodynamic fairing is shown by Figure 61, which is an *Airplane PDQ* screenshot of the same X-plane model.

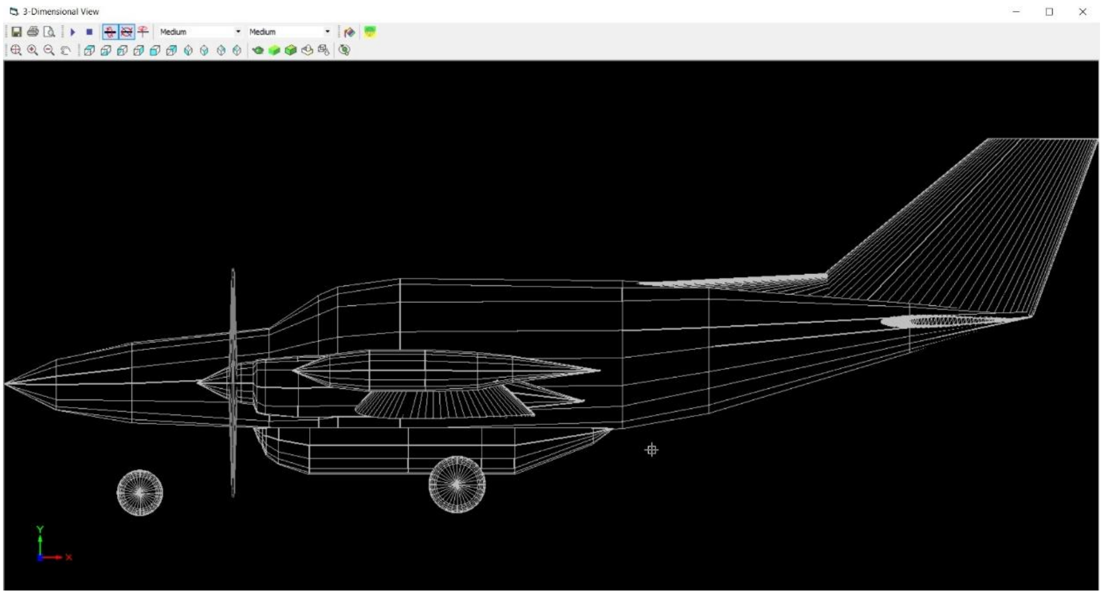


Figure 61. *Airplane PDQ* representation of the LNG tank fairing

### 2.4.3 Unmodified configuration aerodynamic data

#### *2.4.3.1 Aircraft owner's manual data*

Cruise performance data is provided in the Cessna Aircraft Company (1974), C421B aircraft owner's manual, with this data used to derive the zero-lift drag coefficient $C_{Do}$. Cruise performance data was presented at various airspeeds and used to calculate average $C_L/C_D$ (lift/drag) ratio based on mid-cruise weight and constant altitude. In this case the Cessna Aircraft Company (1974), C421B aircraft owner's manual data was used to solve Breguet's range equation assuming values for propeller efficiency, AVGAS Specific Fuel Consumption and fuel loads as provided in the aircraft owner's manual. The method applied is detailed in Section 2.6.1. The results of this analysis provided $C_L/C_D$ data at sea level and various power settings as shown in Table 51.

Table 51. $C_L/C_D$ ratio estimated from C421B Owner's manual performance charts

| | Alt (ft) | Range | Prop eff | SFC | Wbegin | Wend | delWf/W1 | CL/CD |
|---|---|---|---|---|---|---|---|---|
| %73.5 HP | 0 | 743 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 8.7 |
| %69.8 HP | 0 | 772 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 9.0 |
| %64.8 HP | 0 | 807 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 9.5 |
| %59.6 HP | 0 | 842 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 9.9 |
| %54.7 HP | 0 | 871 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 10.2 |
| %49.3 HP | 0 | 901 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 10.6 |
| %43.8 HP | 0 | 910 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 10.7 |
| %42.3 HP | 0 | 914 | 0.8 | 0.45 | 7450 | 6430 | 0.14 | 10.7 |

Lift ($C_L$) and drag ($C_D$) coefficients were then determined based on:

- Aircraft reference area $S_{ref}$ = 215 ft$^2$,
- Airspeed values KTAS (knots true airspeed) = CAS (calibrated airspeed) at sea level, corresponding to engine power settings provided in the Aircraft operators manual.
- Sigma atmosphere density ratio $\sigma = 1$ for sea level flight condition.
- Air density $\rho$ = 0.00238 slugs/ft$^3$ at sea level.

The $C_L$ values were determined from the lift equation using average weight values (mid-range), with drag coefficient values derived from the $C_L/C_D$ values shown in Table 51.

Table 52. Lift ($C_L$) and drag ($C_D$) estimated from C421B Owner's manual performance charts

| Sref | b | A | KTAS | sigma | density | CAS | fps | CL | CD |
|---|---|---|---|---|---|---|---|---|---|
| 215 | 41.8 | 8.14 | 182 | 1.000 | 0.00238 | 182 | 307.2 | 0.287 | 0.033 |
| 215 | 41.8 | 8.14 | 178 | 1.000 | 0.00238 | 178 | 300.5 | 0.300 | 0.033 |
| 215 | 41.8 | 8.14 | 173 | 1.000 | 0.00238 | 173 | 292.0 | 0.318 | 0.034 |
| 215 | 41.8 | 8.14 | 167 | 1.000 | 0.00238 | 167 | 281.9 | 0.341 | 0.035 |
| 215 | 41.8 | 8.14 | 161 | 1.000 | 0.00238 | 161 | 271.8 | 0.367 | 0.036 |
| 215 | 41.8 | 8.14 | 153 | 1.000 | 0.00238 | 153 | 258.3 | 0.407 | 0.039 |
| 215 | 41.8 | 8.14 | 143 | 1.000 | 0.00238 | 143 | 241.4 | 0.466 | 0.044 |
| 215 | 41.8 | 8.14 | 141 | 1.000 | 0.00238 | 141 | 238.0 | 0.479 | 0.045 |

Figure 62 shows the plot of $C_L^2$ vs $C_D$ where the y-intercept of the $C_L^2$ vs $C_D$ line provides the average zero-lift drag coefficient. This is shown by an average value $\boldsymbol{C_{Do}}$ **= 0.025** for the Cessna C421B aircraft. As a comparison, Lan & Roskam (2008) state that the zero-lift drag coefficient for a similar configuration Cessna 310 twin-engine aircraft has a $C_{Do}$ = 0.0263.

Although it is acknowledged that increased accuracy could be obtained by conducting flight tests to obtain specific $C_L$ and drag $C_D$ data, obtaining this data was considered to be out of scope for this thesis. Given that these results compared favourably with another aircraft of similar configuration, it was considered satisfactory to illustrate this aspect of the conceptual design methodology.

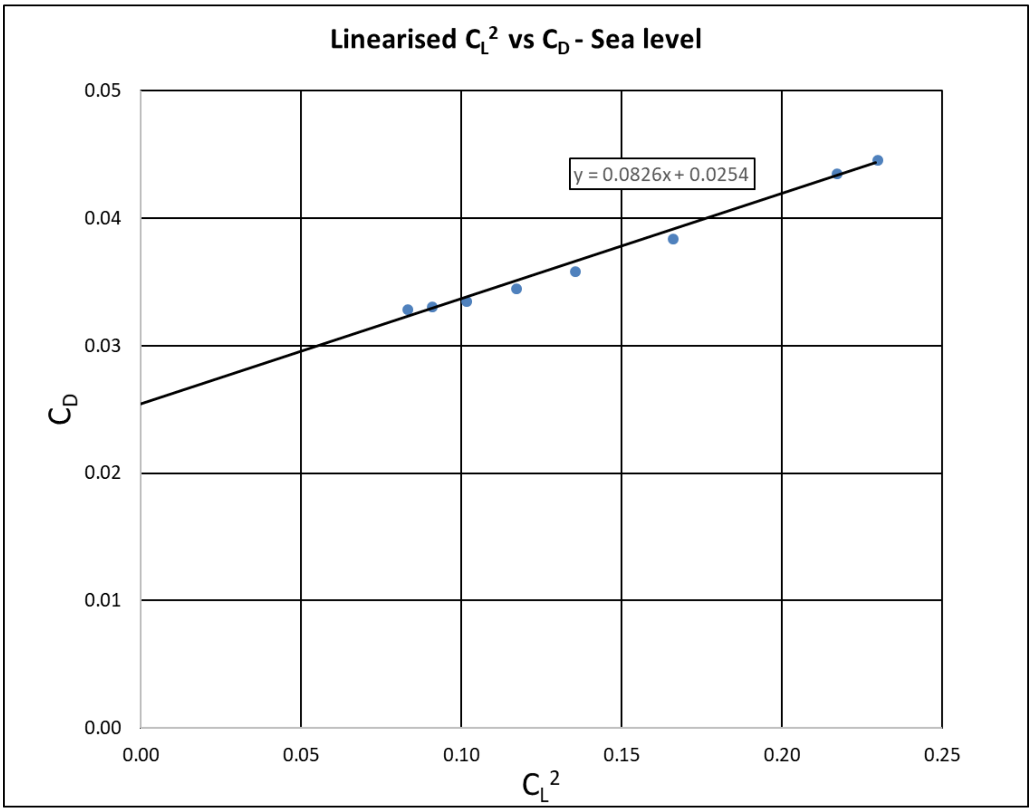



Figure 62. Linearised $C_L^2$ vs $C_D$ for Cessna 421B aircraft at sea level – AFM data

#### *2.4.3.2 Airplane PDQ data*

As stated above the Cessna C421B aircraft was modelled in the *Airplane PDQ* software using geometry generated from an X-plane model of the same. *Airplane PDQ* can provide a number of reports, with one such report being a Drag breakdown of various components of the aircraft in terms of coefficient values and percentage of the total. Table 53 shows an excerpt of the *Airplane PDQ* drag breakdown report for a unmodified Cessna 421B configuration. As highlighted, the total zero-lift drag coefficient for the Cessna 421B was conservatively predicted to be $C_{Do}$ = 0.0028.

This value was higher than the estimated drag coefficient predicted in Section 2.4.3.1 using Operators manual data. This suggests that the *Airplane PDQ* provides conservative predictions of drag, which is consistent with other comparisons reported operations by Williams (2004). Nevertheless, the two values of drag determined from the Cessna C421B aircraft owner's manual and from Airplane PDQ, compare within acceptable tolerances for the drag parameter which is traditionally difficult to predict accurately. For this reason, wind tunnel tests are undertaken in later design phases of an aircraft modification project are required to refine drag predictions. However, drag prediction using analytical models are often the only means available during conceptual design.

Table 53. *Airplane PDQ* drag breakdown report excerpt – Cessna 421B aircraft

**Drag Breakdown Report – Cessna 421B – Clean**

Design Name: C421B
File Name: Cessna 421B clean_2.dwg

| **Component** | | **Cd** | **Percent Total** |
|---|---|---|---|
| Main Wing | | | |
| | Total Drag Coeff | 0.01229 | 43.53 % |
| Horizontal Stabilizer | | | |
| | Total Drag Coeff | 0.002382 | 8.438 % |
| Vertical Stabilizer | | | |
| | Total Drag Coeff | 6.616e-04 | 2.344 % |
| Fuselage | | | |
| | Total Drag Coeff | 0.005916 | 20.96 % |
| TOTAL Airplane Drag | | | |
| | Aircraft Drag Coeff | 0.02823 | 100. % |

## 2.4.4 LNG tank configuration aerodynamic data

### *2.4.4.1 Hoerner method*

The drag contribution of the LNG belly tank installation is approximated using the method as described by Hoerner (1965). This analysis is shown in Section 2.2.2.2, with the estimate of $\boldsymbol{C_D}$ **= 0.0012** based on aircraft reference area. This equates to a drag increase of approximately 5% as referenced to the unmodified aircraft drag coefficient reported in Section 2.4.3.1.

#### *2.4.4.2 Airplane PDQ*

As stated above, the Cessna C421B aircraft was modelled in the *Airplane PDQ* software using geometry generated from an *X-plane* model of the same. In the case the geometry of the LNG fuselage belly tank fairing was modelled using the geometry as shown in Figure 61. From these results, the total zero-lift drag coefficient for the Cessna 421B fitted with a LNG tank was shown to be $C_{Do}$ = 0.0031. Also shown in these results was the drag breakdown of the LNG fuel tank which equated to $\boldsymbol{C_D}$ **= 0.0019**. Again, the drag contribution of the LNG tank as predicted by Airplane PDQ is higher than that predicted by the Hoerner (1965) method.

### 2.4.5 Discussion

As stated above *Airplane PDQ* appears to conservatively predict aircraft performance. For this reason, the general approach has been to apply the conservative *Airplane PDQ* drag prediction as determined by this section to the Breguet's range equation. Breguet's range equation allows the flight to be broken into segments that use LNG fuel or AVGAS fuel, noting that two fuels are used in this bi-fuel arrangement. For example, this analysis assumes that the aircraft is flown on LNG for cruise segments of flight then switched over to AVGAS fuel when required to complete the flight, or when operationally required.

Therefore, in order to facilitate the prediction of range using the approach described above, the drag contribution of the LNG fuel tank installation will be that conservatively predicted by *Airplane PDQ*. The drag contribution of the LNG fuel tank installation is given by $C_{DLNG}$ = **0.0019.**

To determine the new *L/D* ratio (or $C_L/C_D$ ratio), this drag contribution is added to the aircraft baseline drag coefficient, as determined from C421B aircraft owner's manual data for each flight condition. The $C_L/C_D$ ratio data can then be used in combination with natural gas SFC data to calculate range using the Breguet's range equation, as described in the following sections.

The drag coefficient increment resulting from the LNG fuel tank can also be used to predict the resultant cruise speed performance as detailed later in this Section.

## 2.5 NATURAL GAS SPECIFIC FUEL CONSUMPTION ESTIMATION

### 2.5.1 Overview

This section describes the results of a literature review and analysis related to the determination of natural gas fuel Specific Fuel Consumption (SFC) data.

### 2.5.2 Methodology

This method relied on a literature search focussing on SFC data comparing natural gas and gasoline fuels. In all cases the data obtained was derived from the results of tests conducted on Spark Ignition (SI) internal combustion engines. No comparative data could be obtained for AVGAS and natural gas fuels tested on aviation SI engines such as those considered in this case study. Therefore, there will be small differences in results due to the differences in the octane rating of automotive gasoline fuels (91 to 98 Research Octane Rating - RON) and that of AVGAS (100 lean rating). Uncertainties are accounted for in a sensitivity study of SFC as described in Section 2.6 where SFC is varied to determine the effect on aircraft range performance.

### 2.5.3 Results and discussion

Table 54 describes a review of SI engine tests presented in six papers which were conducted to compare SFC results for gasoline and natural gas fuels. The tests as reported in these papers were conducted using various SI engines at different throttle conditions over a range of engine RPMs. This review has collated the results of these SI engine tests and compared the SFC results for gasoline and natural gas fuels. In order to derive a single value of SFC these results were averaged over the engine high operating RPM range. Table 54 shows a summary of this analysis and the average percentage difference between natural gas SFC and the gasoline baseline. The change in SFC due to natural gas fuel operation was then applied as change to the SFC as determined for the Cessna 421B during cruise as described below.

The results of this review presented in Table 54 noted that there was an average 15.4% reduction in SFC for natural gas fuels compared to gasoline. Although this trend was apparent in these six papers, there were two papers that presented SFC data which indicated that SFC of natural gas was higher than that of gasoline. It is noted that this may be due to specific conditions associated with the tests conducted and the setup of the SI engine. In the case of these two papers, the details were insufficient to ascertain the particular sets of conditions that produced natural gas SFC higher than gasoline.

Lastly it is important to note that natural gas-related modifications can be incorporated to SI engines to ensure that they maintain the same or similar power and torque performance characteristics as the gasoline baseline. These natural gas-related changes may include high compression ratio intake valves and increased lift intake and exhaust valves as described by Abu Bakar et al. (2012), all which have the ability to affect SFC performance in natural gas IC engines. Furthermore, it should be noted that similar power and torque performance characteristics could be achieved in aircraft engine applications by changing turbocharger boost levels in accordance with an Engine Control Unit schedule. However, certification of such natural gas-related aero engine changes and the propagation of these changes is problematic as described in later sections of this thesis.

Figure 63 and Figure 64 are samples of such data determined from Aslam (2005) and Mustafi et al. (2006) for engines tested at a Wide-Open Throttle (WOT) condition. It is apparent from these results that the SFC of natural gas is lower than that of gasoline throughout the engine RPM range, with this difference being about 15.4% as shown in Table 54.

#### *2.5.3.1 AVGAS Specific Fuel Consumption*

The Cessna Aircraft Company (1974), C421B aircraft owner's manual states that the total fuel consumption at cruise as follows:

Total fuel flow = 249 lbs/hr at 73.5% BHP as determined from Figure 6-10 at 20000 ft altitude – lean mixture.

The maximum BHP for each engine = 375 BHP as shown by the FAA Type Certificate Data Sheet (TCDS), 3A13 (2006).

Therefore, the developed power for 2 engines = 2× 375× 0.735 = 551.25 BHP

$$\text{Hence the } SFC = \frac{Fuel\ flow\ [\frac{lbs}{hr}]}{HP} = \frac{249}{551.25} = 0.45 \text{ lbs/hp/hr}$$

This SFC compares favourably with typical General Aviation SI piston engines where SFC values vary between 0.4 – 0.5 lbs/hp/hr.

Table 54. Summary of natural gas vs Gasoline Specific Fuel Consumption (SFC) analysis

| | % difference to gasoline at high throttle setting |
|---|---|
| Jahirul et al., (2010), data from Fig. 3 | 14.8% |
| Aslam et al., (2005), data from Fig. 3 and Fig. 6 | 16.3% |
| Ramjee & Reddy, (2011), data from Fig. 6 | 14.3% |
| Mustafi et al., (2006), data from Fig. 2 | 18.7% |
| Tabar et al., (2017), data from Fig. 3 | 20.3% |
| Faizala M et al.,(2009), data from Fig. 4 | 8.0% |
| **Average % difference** | **15.4%** |
| | |
| AVGAS Specfic Fuel Consumption (lb/hp-hr) | 0.45 |
| **NG Specific Fuel Consumption (lb/hp-hr)** | **0.38** |

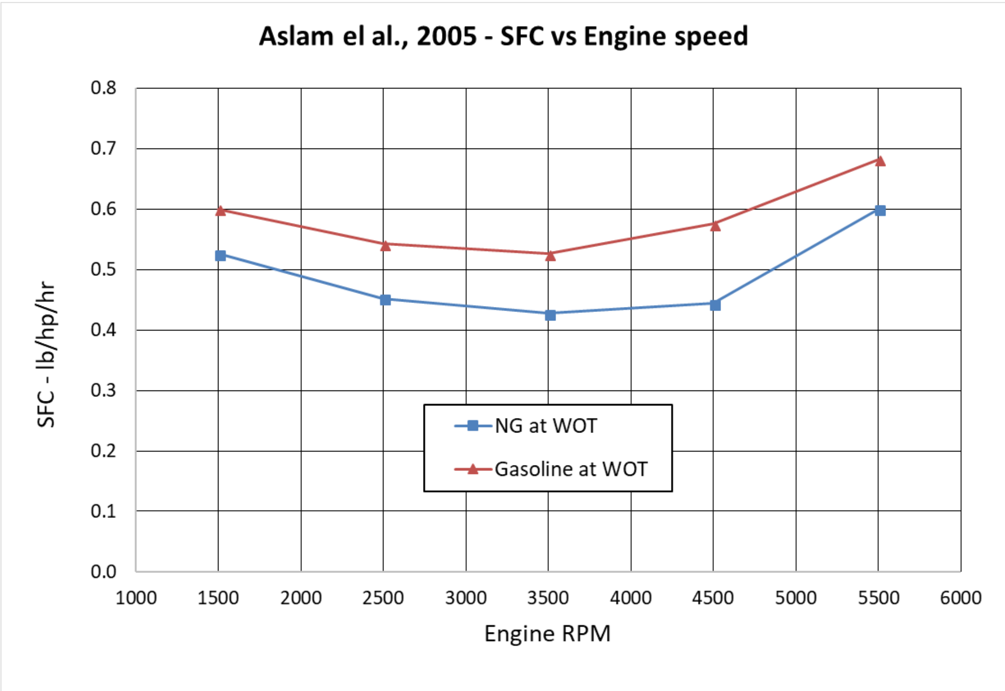



Figure 63. Aslam (2005) - SFC vs engine RPM for natural gas and gasoline fuels

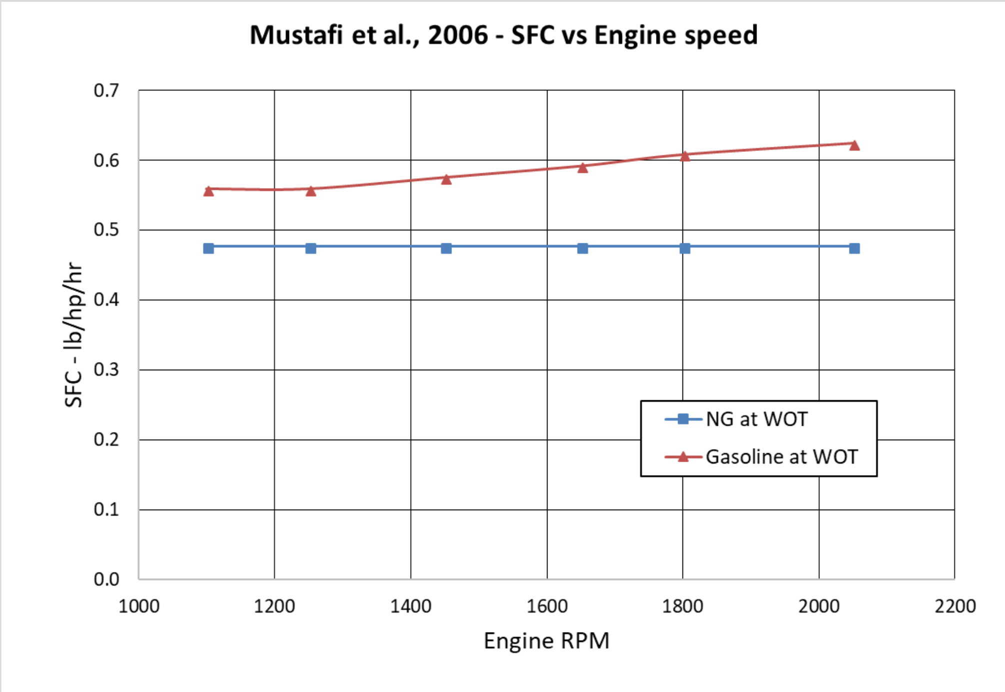



Figure 64. Mustafi et al. (2006) - SFC vs engine RPM for natural gas and gasoline fuels

## 2.6 RANGE PERFORMANCE ESTIMATES

### 2.6.1 Overview

Range performance estimates for the Cessna 421B aircraft modified with the LNG belly tank are provided using the Breguet range equation and data provided previously in this section. These range performance estimates were determined at three altitudes, sea level, 10000 ft and 20000 ft for an engine maximum cruise setting of 73.5% BHP.

A spreadsheet-based analysis was undertaken that divided the flight into various flight segments corresponding to the fuel used in that segment as follows:

- Range segment 1 – Cruise - LNG fuselage belly tank – 230 lbs fuel (104 kg)
- Range segment 2 – Cruise – AVGAS – Wing tip tanks – 600 lbs fuel (272 kg)
- Range segment 3 – Cruise – AVGAS – Wing tanks – 190 lbs fuel (86 kg)

The total fuel loading of 1020 lbs (463 kg) equated to the certificated maximum fuel load for the Cessna 421B aircraft. The distribution of fuel loading was such that maximum LNG fuel load was used in the belly tank, maximum fuel load in the wing tip tanks (for structural inertial relief) and the remaining balance in the wing tanks.

Taxi, take-off, climb, descent and landing segments were not considered in this analysis as range performance data was not required for these segments. The estimation of range performance for these segments was also not required for range-payload predictions.

At stated above, the Breguet range equation for piston engine aircraft was used to calculate range for each range segment. The Breguet range equation as given by Lan & Roskam (2008) is:

$$R = 326\,\eta_p \frac{C_L}{C_D} \frac{1}{SFC} \ln\left(\frac{W_{begin}}{W_{end}}\right) \qquad \text{Equation 37}$$

Where $\eta_p$ – Is the average propeller efficiency, which is assumed to be 0.8 for the purposes of this analysis. Selection of $\eta_p$ = 0.8 is based on typical values used in Lan & Roskam (2008) and Hale (1984).

$C_L/C_D$ – Is the lift to drag ratio determined at the corresponding altitude condition. The baseline $C_L/C_D$ value is determined from Cessna Aircraft Company, 1974, C421B aircraft owner's manual data as described in Section 2.4.3.1. Note that this baseline $C_L/C_D$ value was modified by the increment in drag coefficient due to the LNG fuel tank installation. Therefore, the new drag coefficient was based on the old drag coefficient determined at the corresponding cruise flight condition plus the drag increment due to the LNG fuel tank installation thus:

$C_{Dnew} = C_{Dold} + C_{DLNG}$ where $C_{D\,LNG}$ is determined from Section 2.4.5.

$SFC$ – Is the relevant Specific Fuel Consumption for the appropriate flight segment. That is $SFC$ = 0.45 lbs/hp/hr for segments flown on AVGAS, and $SFC$ = 0.38 lbs/hp/hr as determined from Section 2.5.

$W_{begin}$ – Is the weight of the aircraft at the start of cruise.

$W_{end}$ – Is the weight of the aircraft at the end of cruise.

Note that these two weights are corrected for the increase in empty weight of the aircraft due to the LNG belly tank installation.

### 2.6.2 Results

The results of this range performance analysis are shown in Table 55. This table presents range estimates using $C_L/C_D$ ratio corrected for the LNG belly tank drag increment and corrected SFC data using the Breguet's range equation. This analysis has been further extended to include a sensitivity study of $C_L/C_D$ and *SFC* to account for uncertainties associated with the estimates in these two quantities. In this case, $C_L/C_D$ coefficient was varied by decreasing this quantity by 5% to account for uncertainties implicit in the analysis of data derived from the Aircraft operator's manual and estimates of the LNG belly tank installation. LNG Specific Fuel Consumption (*SFC)* was increased by 10% to account for uncertainties or errors associated with the original estimate of this quantity as described in Section 2.5. These two quantities were used to calculate range distance separately as shown by *CL/CD-5%* and *SFC+10%* as shown in Table 55. In addition, the two quantities were combined to provide a range distance of *Dist_combined* for each applicable range segment.

It should be noted that Range segment 1 provides two range estimates. The first corresponds to range calculated using LNG fuel with the belly tank installation i.e. *SFC* =0.38. The second is provided for comparison, and corresponds to range if AVGAS is used on this range segment, and no belly tank is installed. In this case a *SFC* = 0.45 is used.

Accordingly, the remaining Range segments 2 and 3 are flown using AVGAS fuel, so range values are determined using the corresponding AVGAS fuel weights as described above, and *SFC* = 0.45.

In general, it can be observed from Range segment 1 that that the range results for the LNG/AVGAS bi-fuel option outperform the AVGAS configuration by a small margin of about 23 NM (43 km) over the altitudes analysed. However, these results are based on of $C_L/C_D$ ratio and LNG *SFC* estimates, which are subject to uncertainties implicit in the analysis of aerodynamic data and SFC data as discussed earlier. The sensitivity study undertaken on these two quantities therefore attempts to rationalise these uncertainties, showing that the range results for the LNG/AVGAS bi-fuel option falls short of the AVGAS only configuration by a very small margin of about 3 NM (5.6 km) over the same altitudes. The fact that the range benefits are only realised if the nominal LNG SFC is achieved. These range benefits are marginal, and other operational factors such as changing winds, flight profiles and engine performance may mask the effect of any range improvement.

The range results for the complete AVGAS/LNG bi-fuel mission are shown in the lower portion of Table 55, where these results are compared to the AVGAS (clean configuration) baseline in Section 1.9. However, it can be seen that the sensitivity study results show a decrease in range results when compared to the nominal. This reduction in range equates to about 50 NM (93 km) over a typical maximum range mission. It is important to note that the sensitivity study selected a ±5% variation in the respective quantity as this provided a 'round' number for the analysis. In reality it is expected that drag coefficient may be significantly more that 5% value, due to inaccuracies in estimation and real-world aerodynamic design inefficiencies. Therefore, it is expected that these results to be an optimistic estimate of range for an AVGAS/LNG modified commuter aircraft.

Lastly, it is important to note that this prediction is subject to limitations associated with conceptual design methods and data approximations. Specifically, high accuracy/confidence flight testing or wind tunnel testing data was not available. This is further discussed in Section 1.9 where payload-range performance is analysed in context with other design metrics.

## Table 55. Range performance

**Drag calculation - NG belly tank installation**

| | |
|---|---|
| CD_NG belly tank installation | 0.0019 |
| Weight_NG belly tank installation (lbs) | 65.4 |

| | CL/CD | CD | CL | CD_new | CL/CD_new |
|---|---|---|---|---|---|
| Alt = 0 ft at %73.5HP | 8.71 | 0.033 | 0.287 | 0.0349 | 8.23 |
| Alt = 10000 ft at %73.5HP | 9.55 | 0.046 | 0.441 | 0.0481 | 9.17 |
| Alt = 20000 ft at %73.5HP | 10.46 | 0.067 | 0.705 | 0.0693 | 10.18 |

| | |
|---|---|
| Wempty (lbs) | 4847.4 |
| MTOW (lbs) | 7450 |

| | Fuel load | |
|---|---|---|
| AVGAS_wing tip tanks (lbs) | 600 | |
| AVGAS_wing tanks (lbs) | 190 | Note max quantity to remain with 1020 lbs fuel load limit |
| LNG_belly tank (lbs) | 230 | |
| Total fuel (lbs) | 1020 | |

**Range segment 1 - Cruise_LNG - Fuselage belly tank**

| | |
|---|---|
| Weight_begin | 7450 |
| Weight_end | 7220 |

| | Alt (ft) | Prop eff | SFC | Wbegin | Wend | CL/CD | Dist_1 | CL/CD-5% | Dist_1_delCL/CD | SFC+10% | Dist_1_SFC | Dist_1_combined |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| %73.5 HP | 0 | 0.8 | 0.38 | 7450 | 7220 | 8.23 | 176.9 | 7.82 | 168.0 | 0.42 | 160.8 | 152.8 |
| | 10000 | 0.8 | 0.38 | 7450 | 7220 | 9.17 | 197.1 | 8.71 | 187.2 | 0.42 | 179.2 | 170.2 |
| | 20000 | 0.8 | 0.38 | 7450 | 7220 | 10.18 | 218.7 | 9.67 | 207.7 | 0.42 | 198.8 | 188.8 |

| | Alt (ft) | Prop eff | SFC | Wbegin | Wend | CL/CD | Dist_1 |
|---|---|---|---|---|---|---|---|
| %73.5 HP | 0 | 0.8 | 0.45 | 7450 | 7220 | 8.71 | 158.2 |
| | 10000 | 0.8 | 0.45 | 7450 | 7220 | 9.55 | 173.6 |
| | 20000 | 0.8 | 0.45 | 7450 | 7220 | 10.46 | 190.2 |

**Range segment 2 - Cruise_AVGAS - Wing tip tank fuel**

| | |
|---|---|
| Weight_begin | 7220 |
| Weight_end | 6620 |

| | Alt (ft) | Prop eff | SFC | Wbegin | Wend | CL/CD | Dist_2 | CL/CD-5% | Dist_2_delCL/CD |
|---|---|---|---|---|---|---|---|---|---|
| %73.5 HP | 0 | 0.8 | 0.45 | 7220 | 6620 | 8.23 | 414.0 | 7.82 | 393.3 |
| | 10000 | 0.8 | 0.45 | 7220 | 6620 | 9.17 | 461.3 | 8.71 | 438.2 |
| | 20000 | 0.8 | 0.45 | 7220 | 6620 | 10.18 | 511.8 | 9.67 | 486.2 |

**Range segment 3 - Cruise_AVGAS - Wing tank fuel**

| | |
|---|---|
| Weight_begin | 6620 |
| Weight_end | 6430 |

| | Alt (ft) | Prop eff | SFC | Wbegin | Wend | CL/CD | Dist_3 | CL/CD-5% | Dist_3_delCL/CD |
|---|---|---|---|---|---|---|---|---|---|
| %73.5 HP | 0 | 0.8 | 0.45 | 6620 | 6430 | 8.23 | 139.0 | 7.82 | 132.0 |
| | 10000 | 0.8 | 0.45 | 6620 | 6430 | 9.17 | 154.8 | 8.71 | 147.1 |
| | 20000 | 0.8 | 0.45 | 6620 | 6430 | 10.18 | 171.8 | 9.67 | 163.2 |

**Mission totals - LNG/AVGAS**

| Alt (ft) | Range (NM) | Range_delCL/CD | Range_delSFC | Range_delComb |
|---|---|---|---|---|
| 0 | 730 | 693 | 714 | 678 |
| 10000 | 813 | 773 | 795 | 755 |
| 20000 | 902 | 857 | 882 | 838 |
| | | 5% decrease in CL/CD | 10% increase in LNG SFC | Combined CL/CD & LNG SFC impact |

## 2.7 CRUISE SPEED PERFORMANCE

### 2.7.1 Overview

Change in cruise speed performance resulting from the LNG belly tank drag increment is calculated in this section by comparing the unmodified and LNG modified configurations. In this comparison, the impact of power changes resulting from the differences in engine power output caused by the usage of AVGAS or LNG fuels is not considered. This aspect of engine power change impacts is considered to be managed by the Engine Control Unit (ECU). In this case the ECU shall be designed to compensate for LNG-related engine power outputs through increasing turbocharger boost thus maintaining comparable power performance with AVGAS. This of course is a non-trivial design exercise and therefore the development and certification of such an ECU would be a challenge which is accounted for in the change propagation analysis shown in Appendix 1.

### 2.7.2 Approach

Cruise speed performance differences has been calculated using the relationship that constant thrust = drag at a design airspeed condition. In this case the simple relationship given by the drag equation is used to determine drag of the unmodified aircraft which is then used to determine cruise speed of the LNG modified aircraft. The output of this process is shown in Table 56 for a sea level altitude and a maximum cruise power setting.

Table 56. Cruise speed difference at sea level and 73.5% HP

| Altitude (ft) | Baseline - $C_D$ | Airspeed - KCAS | Cruise speed - ft/s | sigma | Density - slugs/ft$^3$ | Drag | LNG mod - $C_D$ new | New cruise Airspeed - ft/s | Airspeed - KCAS | Delta (kt) |
|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 0.033 | 182 | 307.2 | 1.00 | 0.00238 | 797.1 | 0.0349 | 298.7 | 177 | **-5** |

### 2.7.3 Results and discussion

Table 56 shows a 5-knot reduction in cruise airspeed as a result of the LNG fuel tank installation drag increment as described above. This reduction in cruise speed is comparable to other similar configurations such as an external cargo pod on other aircraft. An example of an external cargo pod installation is an option on the Cessna Caravan 208B aircraft as illustrated in Figure 65. The cargo pod installation is shown beneath the fuselage.

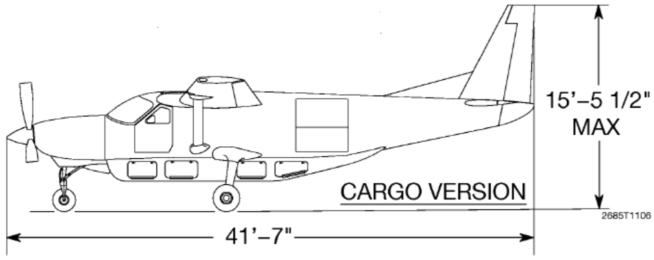



Figure 65. Cessna Caravan C208B with cargo pod

A comparison of Cessna Caravan performance data provided in the Cessna Model C208B Information Manual (2008) shows cruise speed data for both configurations with and without cargo pod. These data show a 9-knot reduction in cruise speed at maximum cruise power setting and 10000 ft (3000 m) altitude with cargo pod fitted. It should be noted that this Cessna cargo pod installation is of different aerodynamic profile compared to the aerodynamically faired LNG tank and is also adversely affected by the propeller flow field. Nevertheless, this reduction in Cessna Caravan aircraft cruise airspeed is comparable to that predicted here for the belly mounted LNG fuel tank installation within limitations of the analytical methods applied. This difference however highlights the differences in real world aerodynamic data and analytical predictions of the same, and therefore justifies the sensitivity study as described earlier.

# Appendix 3. Electric propulsion system – Case study

## 3.1 BACKGROUND

This case study implements the matrix-based conceptual design methodology framework involving a small aircraft Electric Propulsion (EP) system modification. The underpinning theory associated with this framework has been detailed in Chapter 4, and is similar to that outlined in Appendix 1. This case study illustrates implementation of conceptual design from the requirements management stage through several steps to the final development of the initial systems specification. As stated above, this matrix-based methodology enables a structured approach, where the outputs of one stage forms the basis of the next. As stated in Chapter 4 this conceptual design methodology accounts for the aircraft system lifecycle through (1) development and management of requirements, (2) generation of concepts, (3) concept selection validation, (4) evaluation of design changes, (5) evaluation of design impacts and certification impacts, and (6) evaluation of performance. Each of the matrices developed have outputs which are either used as feedback for earlier steps or are used as inputs to the next steps as described in ensuing sections. Given that the methodology is the same or very similar to that described in Appendix 1 (natural gas fuels case study), the underpinning implementation will not be described in detail. Rather this case study will focus on the outputs of each step and the differences encountered as a result of this case study technology, mission and systems architecture.

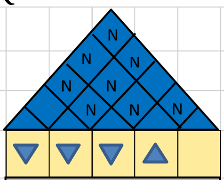
## 3.2 REQUIREMENTS – STEP 1

The Quality Functional Deployment (QFD) matrix as described in Chapter 4 is a method intended to help in the design of complex products, by taking various customer requirements into account. This is accomplished using a selection matrix that helps evaluate the impact of the various customer needs and requirements on areas such as the engineering development. As described in Chapter 4, a QFD matrix is a specialised matrix, designed to convert customer requirements into a numeric score that helps define areas for conceptual design. The QFD matrix is formulated for this EP modification using the same methods as described in Chapter 4, and Appendix 1.

### 3.2.1 Quality function deployment matrix

As stated in Appendix 1 it is assumed that customer surveys have been collected supporting the development of the EP system modification with needs established along with desired requirements. In this case, a simplified set of requirements have been determined which are consistent with those as described in Appendix 1. As described previously in Appendix 1, the actual QFD matrix would almost certainly have more than seven requirements. However, these requirements are limited for the purposes of this case study. Like the natural gas case study, the needs are for a high performance, safe, inexpensive, environmentally friendly, compatible, and spacious (internal fuselage volume for skydivers) system modification, providing a skydiving aircraft capability. It is assumed that the customer survey has requested that potential customers rate the corresponding requirements using values between 1 (not important) and 5 (very important) as depicted in Table 57. The weightings are assigned to engineering challenges, as defined in Table 58, and are then used to populate the main body of the QFD matrix in Table 57. The weightings as applied to the lower line totals show Life Cycle Costs (LCC) as the highest score, closely followed by weight and power impacts. It is these parameters, or metrics, that receive the greatest attention during the conceptual design phase, noting that skydiving missions are primarily dependent on aircraft climb performance, which is directly related to aircraft weight and engine power.

Table 57. Aircraft Electric propulsion system modification – QFD matrix



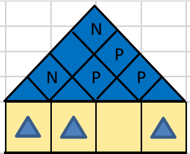

| Customer needs | Aircraft Mod Spec Requirements | Importance 1 (not) - 5 (very) | | Modification engineering challenges: Drag impact | Weight impact | LCC impact | Power impact | Size impact | |
|---|---|---|---|---|---|---|---|---|---|
| High performance - Time of climb | The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load. | 4 | 16.0% | 9 | 9 | 9 | 9 | 3 | |
| High performance - Useful load | The modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | 5 | 20.0% | 9 | 9 | 9 | 9 | 3 | |
| Low cost | The modification lifecycle costs (LCC) shall be kept to a minimum. | 4 | 16.0% | 3 | 9 | 9 | 9 | 3 | |
| Safety & airworthiness | The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 3 | 12.0% | 1 | 1 | 3 | 1 | 1 | |
| Emissions | The modification shall minimise emissions. | 3 | 12.0% | 3 | 1 | 1 | 1 | 1 | |
| Compatibility | The modification shall be compatible with a range of single engine light aircraft types. | 3 | 12.0% | 1 | 1 | 1 | 1 | 1 | |
| Spacious | The modification shall not impact passenger, baggage or cargo space. | 3 | 12.0% | 3 | 3 | 3 | 3 | 9 | |
| | SUM = | 25 | 100.0% | 4.68 | 5.4 | 5.64 | 5.4 | 3 | 24.12 |
| | | | | 19.4 | 22.4 | 23.4 | 22.4 | 12.4 | |

Table 58. Aircraft EP engineering challenges definitions

| | Engineering challenges definitions |
|---|---|
| Drag impact | This relates to the impact of the modification on overall aircraft drag and performance |
| Weight impact | This relates to the impact of the modification on empty weight and hence useful load capability |
| Life Cycle Cost impact | This relates to the impact of the modification on aircraft LCC |
| Power impact | This relates to the impact of the modification on propulsion system power output and/or related changes |
| Size impact | This relates to the impact of the modification on aircraft internal volume and/or space affecting payload capability |

As stated earlier in this thesis, the installation of an EP system modification on a small aircraft not only affects the aircraft segment, but also the ground segment. This ground segment comprises the ground charging subsystem which provides the recharging capability for the propulsion system batteries. The ground segment QFD matrix is analysed in the same way as the aircraft segment described above. Table 59 shows the QFD matrix for the ground charging segment, using the same sub-matrices as described above. In this particular case, the 'battery replacement time', 'recharge time' and 'costs' are the three critical technical performance measures, or metrics, that should receive the greatest attention during the conceptual design phase.

Table 59. EP ground charging segment – QFD matrix




| Customer needs | Ground infrastructure requirements | Importance 1 (not) - 5 (very) | | Engineering challenges: Recharge time | Battery replacement time | Transportability | Low running costs | |
|---|---|---|---|---|---|---|---|---|
| Battery recharge time | The propulsion system batteries shall have the capability to be recharged to 100% capacity within 1 hour. | 4 | 20.0% | 1 | 9 | 3 | 3 | |
| Battery exchange time | The ground charging subsystem infrastructure shall allow replacement of all propulsion system batteries within 15 minutes of aircraft shutdown. | 4 | 20.0% | 3 | 3 | 3 | 3 | |
| Recharging capacity | The ground charging subsystem shall have the capacity to recharge one complete set of propulsion system batteries in one charging cycle. | 4 | 20.0% | 3 | 9 | 9 | 3 | |
| Safety | The ground charging subsystem shall comply with Australian Standards for electrical wiring and Electric Vehicle charging systems. | 5 | 25.0% | 9 | 3 | 1 | 3 | |
| Low cost | The ground charging subsystem shall be designed to minimise capital and operating costs. | 3 | 15.0% | 9 | 3 | 3 | 9 | |
| | SUM = | 20 | 100.0% | 5 | 5.4 | 3.7 | 3.9 | 18 |
| | | | | 27.8 | 30 | 20.6 | 21.7 | |

Like the aircraft segment QFD analysis the definitions of these engineering challenge focus areas are provided in Table 60. It can be seen that the assumptions implicit in this analysis place priority on recharge time, or battery exchange time, and low running costs.

Table 60. EP ground charging segment engineering challenge definitions

| | Engineering challenges definitions |
|---|---|
| Recharge time | This relates to achieving the required recharge time as determined by flight duty cycle. |
| Battery replacement time | This relates to the time taken to replace batteries as determined by flight duty cycle. |
| Low running costs | This relates to the design of the ground charging station to minimise operation costs and maintenance costs. |

### 3.2.2 System sizing

#### *3.2.2.1 Weight & Balance aspects*

The modification as considered in this case study involves the removal of the existing Cessna 182P powerplant and related accessories, and the installation of an equivalent EP system. The main items removed from the aircraft are components located forward of the firewall comprising the engine, propeller and associated assemblies, various items in the cabin area, and a selection of miscellaneous unrequired fuel system components. Table 61 shows the component weights and moment arms associated with items removed from the aircraft as part of this EP modification. This data shown in Table 61 is obtained from Cessna Aircraft Company – Cessna 182P Pilots Operating Handbook (1976), with those exceptions denoted as estimates (est.) in the Equipment List Description column. This weight data is used to support quantisation of the morphological matrix as described in the following sections.

Table 61. Cessna 182P Pilot Operating Handbook Items Removed

| Item No | Equipment List Description | Weight (lbs) | Weight (kg) | Arm (inches) | Arm (mm) | Moment (lb.in) | Moment (kg.mm) |
|---|---|---|---|---|---|---|---|
| | **POWERPLANT & ACCESSORIES** | | | | | | |
| A01-R | Engine, Continental O-470-S Spec | 442.0 | 200.5 | -17.5 | -444.5 | -7735.0 | -89117.8 |
| | Two magnetos with impulse coupling | 12.9 | 5.9 | -12.0 | -304.8 | -154.8 | -1783.5 |
| | Oil cooler-Harrison | 4.6 | 2.1 | -31.5 | -800.1 | -144.9 | -1669.4 |
| | Twelve spark plugs | 2.8 | 1.3 | -19.0 | -482.6 | -53.2 | -612.9 |
| | Starter 12 volt | 17.8 | 8.1 | -4.5 | -114.3 | -80.1 | -922.9 |
| A05-R | Filter carbuettor | 1.0 | 0.5 | -33.0 | -838.2 | -33.0 | -380.2 |
| A09-R | Alternator 14 volt 60 AMP | 11.5 | 5.2 | -5.5 | -139.7 | -63.3 | -728.7 |
| A21-A | Filter installation | 4.5 | 2.0 | -3.4 | -86.4 | -15.3 | -176.3 |
| | Adapter assembly | 1.5 | 0.7 | -4.2 | -106.7 | -6.3 | -72.6 |
| | Filter can assembly | 1.8 | 0.8 | -3.0 | -76.2 | -5.4 | -62.2 |
| | Filter element kit | 0.3 | 0.1 | -3.0 | -76.2 | -0.9 | -10.4 |
| A61-S | Vacuum system, engine driven | 4.5 | 2.0 | 0.0 | 0.0 | 0.0 | 0.0 |
| A70-A | Priming system | 1.0 | 0.5 | -15.0 | -381.0 | -15.0 | -172.8 |
| | Oil - 13 quarts | 24.4 | 11.1 | -17.5 | -444.5 | -427 | -4919.6 |
| | Engine mounts (est.) | 44.1 | 20.0 | -12 | -304.8 | -529.1 | -6095.7 |
| | **SUB-TOTALS** | **574.7** | **260.7** | | | -9263.2 | -106725.0 |
| | **PROPELLER** | | | | | | |
| A33-R | Propeller, McCauley 2A34C203/90DCA-8 | 51.4 | 23.3 | -41.6 | -1056.6 | -2138.2 | -24635.4 |
| A37-R | Governor, Propeller | 3.0 | 1.4 | -32.5 | -825.5 | -97.5 | -1123.3 |
| A41-R | Spinner installation, Propeller | 3.0 | 1.4 | -42.0 | -1066.8 | -126.0 | -1451.7 |
| | **SUB-TOTALS** | **57.4** | **26.0** | | | -2361.7 | -27210.5 |
| | **INSTRUMENTS** | | | | | | |
| D22-A | Gage, carbuettor air temperature | 1.0 | 0.5 | 5.5 | 139.7 | 5.5 | 63.4 |
| D34-R | Instrument cluster engine & fuel | 0.7 | 0.3 | 8.2 | 208.3 | 5.7 | 66.1 |
| D73-R | Gage, manifold pressure | 0.9 | 0.4 | 15.8 | 401.3 | 14.2 | 163.8 |
| D85-R | Tachometer installation engine | 0.9 | 0.4 | 13.8 | 350.5 | 12.4 | 143.1 |
| | **SUB-TOTALS** | **3.5** | **1.6** | | | 37.9 | 436.4 |
| | **CABIN ACCOMODATIONS** | | | | | | |
| E07-O | Seat, Co-pilot articulating | 24.0 | 10.9 | 41.5 | 1054.1 | 996 | 11475.3 |
| E09-S | Seat, 2nd row bench | 23.0 | 10.4 | 80.5 | 2044.7 | 1851.5 | 21331.8 |
| E23-O | Belt & shoulder harness assy, co-pilot (Aust) | 1.6 | 0.7 | 37.0 | 939.8 | 59.2 | 682.1 |
| E27-O2 | Belt & shoulder harness assy, 2nd row (Aust) | 3.2 | 1.5 | 74.0 | 1879.6 | 236.8 | 2728.3 |
| | **SUB-TOTALS** | **51.8** | **23.5** | | | 3143.5 | 36217.4 |
| | **OTHER** | | | | | | |
| | Fuel system bladders (est.) | 28.0 | 12.7 | 46.0 | 1168.4 | 1288 | 14839.5 |
| C01-R | Battery, 12 volt, 33 amp hr | 26.5 | 12.0 | 130.5 | 3314.7 | 3458.25 | 39843.8 |
| C04-R | Regulator, 14 volt, 60 amp alternator | 0.5 | 0.2 | -0.7 | -17.8 | -0.35 | -4.0 |
| C07-R | Ground service plug receptacle | 3.2 | 1.5 | -2.6 | -66.0 | -8.32 | -95.9 |
| | **SUB-TOTALS** | **58.2** | **26.4** | | | 4737.58 | 54583.4 |
| | **GRAND-TOTAL** | **745.6** | **338.2** | | | -1344.3 | -15487.8 |

Table 62 shows a sample of component weights and moment arms associated with items added to the aircraft as part of this EP modification. The weights and arms are estimates of electric propulsion system components based on data determined from product literature, and parametric estimates provided in scientific papers and journals with those exceptions denoted as estimates (est.) in the Equipment List Description column. This weight and moment arm information is used in the morphological matrix as described in the following sections, with actual weights corresponding to motor and propeller type selected in the quantisation as required.

### 3.2.2.2 Motor weights

It should be noted that the weights of most electric motors as stated in Table 68 is based on configurations which were not developed for aeronautical applications. The exceptions were the Siemens AG260D motor and the Contra-Electric 2X YASA 750 axial flux series motor, both of which have been specifically developed for aviation. These motors therefore have appropriate modifications incorporated to react propeller thrust and torque loads and to allow for effective cooling at high power settings. These modifications add to the installed motor weight. Corrections for motor installed weight were not applied, as not enough was known about the specifics of each motor, such as the configuration and layout, the mounting and installation requirements, or cooling setup. Rather this case study adopted the motor weight data as presented in Table 68 without correction. Accordingly, the design methodology would note the motor installed weight uncertainty as a potential risk, to be added to the risk matrix described in Table 94. In addition, this motor installed weight attribute would form a requirement of Design Specification Document (DSD), which would be updated or refined in the next design phase. Specific details of the DSD is discussed in Chapter 4.

Table 62. Cessna 182P Items Added – Sample

| | POWERPLANT & ACCESSORIES | | | | | | |
|---|---|---|---|---|---|---|---|
| **Item No** | **Equipment List Description** | **Weight (lbs)** | **Weight (kg)** | **Arm (inches)** | **Arm (mm)** | **Moment (lb.in)** | **Moment (kg.mm)** |
| | Motor | 110.2 | 50.0 | -25.0 | -635.0 | -2755.0 | -31741.4 |
| | Controller and inverter | 33.0 | 15.0 | -20.0 | -508.0 | -660.0 | -7604.1 |
| | Cabling | 22.0 | 10.0 | -20.0 | -508.0 | -440.0 | -5069.4 |
| | Batteries | 497.4 | 225.6 | -10.0 | -254.0 | -4974.0 | -57307.3 |
| | Propeller/spinner - conventional | 44.0 | 20.0 | -41.6 | -1056.6 | -1830.4 | -21088.7 |
| | Engine mounts (est.) | 44.1 | 20.0 | -12 | -304.8 | -529.1 | -6095.7 |
| | **SUB-TOTALS** | **750.7** | **340.5** | | | -11188.5 | -128906.6 |
| | **INSTRUMENTS** | | | | | | |
| | State of Charge/State of Health indicator | 20.0 | 9.1 | 5.5 | 139.7 | 110.0 | 1267.4 |
| | **SUB-TOTALS** | **20.0** | **9.1** | | | 110.0 | 1267.4 |
| | **CABIN ACCOMODATIONS** | | | | | | |
| | Slipper power pod batteries | TBD | TBD | TBD | TBD | TBD | TBD |
| | **SUB-TOTALS** | **0.0** | **0.0** | | | 0 | 0.0 |
| | **OTHER** | | | | | | |
| C07-R | Ground recharging plug receptacle | 3.2 | 1.5 | -2.6 | -66.0 | -8.32 | -95.9 |
| | **SUB-TOTALS** | **3.2** | **1.5** | | | -8.32 | -95.9 |
| | | | | | | | |
| | **GRAND-TOTAL** | **773.9** | **351.0** | | | -11086.8 | -127735.1 |

### 3.2.2.3 Battery weights

Storage of electrical energy in electric vehicles is one of the greatest challenges of these types of propulsion systems. In recent years, battery technology has steadily improved with the development of lithium-ion type battery storage as detailed by Patterson et al. (2012). Lithium-ion polymer batteries, polymer lithium ion or more commonly lithium polymer rechargeable batteries have technologically very high energy densities, which why these types have been used in aviation applications.

Fehrenbacher et al. (2011) has undertaken an extensive analysis of battery types in 2011, and it is expected that in this time the specific energy density (which is a term used in aeronautical power applications) associated with these battery types has improved 3-5% per year. For example, Fehrenbacher et al. (2011) quotes Lithium Polymer batteries specific energy densities ranging from 129 to 142 Watt-hour/kg (Whr/kg). Patterson et al. (2012) makes predictions for battery specific energy density at the year 2015 and 2035 timeframes. Table 63 shows these technology assumptions presented by Patterson et al. (2012).

Table 63. Electric propulsion technology assumptions

| | **Technology year** | | |
|---|---|---|---|
| | **2015** | **2035** | **2050** |
| Motor peak specific power (HP/lb) | 4 | 6 | 12.5 |
| Motor peak specific power (kW/kg) | 6.6 | 9.9 | 20.6 |
| Motor nominal specific power (HP/lb) | 3 | 4.5 | 9.375 |
| Motor nominal specific power (kW/kg) | 4.9 | 7.4 | 15.4 |
| Motor efficiency without gearbox | 0.95 | 0.97 | 0.98 |
| Motor efficiency with gearbox | 0.925 | 0.95 | 0.97 |
| Controller specific weight (lb/HP) | 0.05 | 0.05 | 0.05 |
| Controller specific weight (kg/kW) | 0.03 | 0.03 | 0.03 |
| Controller efficiency | 0.98 | 0.99 | 0.99 |
| Battery specific energy, <5°C (Whr/kg) | 200 | 600 | 1200 |
| Battery specific energy, >5°C, <20°C (Whr/kg) | 150 | 450 | 900 |
| Battery specific energy, >20°C, <60°C (Whr/kg) | 100 | 300 | 600 |
| Battery efficiency | 0.98 | 0.98 | 0.99 |

Patterson et al. (2012)

A summary of the various batteries and associated technical characteristics is presented in Table 64. Some of these battery systems are used as the basis for specific energy density limits in the concepts considered in this EP case study.

Table 64. Battery specifications and characteristics

| Aircraft/battery description | Power capacity (kWh) | Mass (kg) | Specific energy density (Wh/kg) | Ave charge time (mins) | Cost $/Wh | Reference | Comments |
|---|---|---|---|---|---|---|---|
| **Silent Club** | 1.4 | 40.0 | 35.0 | 40 | ---- | Muntwyler & Vezzino (2015) | |
| **Silent 2 Targa Electro** | 4.3 | 31.0 | 138.7 | 150 | ---- | Muntwyler & Vezzino (2015) | |
| **Antares 20E** | 11.6 | 77.0 | 150.6 | 540 | ---- | Muntwyler & Vezzino (2015) | |
| **Yuneec E430** | 13.3 | 83.5 | 159.3 | 210 | ---- | Muntwyler & Vezzino (2015) | |
| **E-Spyder** | 4.66 | 30.0 | 155.3 | 210 | ---- | Muntwyler & Vezzino (2015) | |
| **Magnus eFusion** | 8.8 | 65.6 | 134.1 | ---- | ---- | Refueling an electric aircraft - Accessed at https://www.youtube.com/watch?v=al8OriHmd60 | 8 battery modules assumed |
| **Electric Power Systems EPiC n42 Liquid Cooled Lithium Battery** | 21.0 | 175.0 | 120.0 | 420 | ---- | EP Systems (2018) accessed at http://ep-sys.net/wp-content/uploads/2017/05/EPiC-n42.pdf | 10 battery modules assumed |
| **Electric Power Systems EPiC t32 Liquid Cooled Lithium Titanate Battery** | 6.3 | 85.0 | 74.1 | 200 | ---- | EP Systems (2018) accessed at http://ep-sys.net/wp-content/uploads/2017/07/EPiC-t32.pdf | 10 battery modules assumed |
| **Pipistrel Alpha Electro - Kokam Company LLC** | 21.0 | 122.0 | 172.1 | 90 | ---- | Pipistrel, DOO & Ajdovščina (2017) - Pilots Operating Handbook<br>Horne Thomas A. (2015) - Amping up the light single accessed at https://www.aopa.org/news-and-media/all-news/2015/october/pilot/f_pipistrel | 2 battery modules |
| **Contra-electric power system** | 33.8 | 125.0 | 270.0 | ---- | ---- | ContraElectric Propulsion Ltd (2018) accessed at http://www.contraelectric.com/innovation/crps-specification | This mass includes all battery cells, inverters, controllers, cabling and electronics in a suitable package for mounting. |
| **Parametric battery cost** | ---- | ---- | ---- | ---- | 0.2 | Stoll AM and Veble Mikic G (2016) | |

#### *3.2.2.4 Electrical controller weights*

An electrical controller is a device that is used to modulate the performance of an electric motor as described by Patterson et al. (2012). The functions of an electrical controller may include a means for overload and fault protection, starting or stopping the motor, selection of the rotation direction, regulation of the motor speed and the regulation of torque. Patterson et al. (2012) provides controller specific weights equating to 0.03 kg/kW (0.05 lb/HP) at the technology year 2015.

Fehrenbacher et al. (2011) quotes a mass of 31.8 kg (70 lbs) for the electric motor controller associated with the Cessna 172K electric propulsion conversion case study. This controller specific weight equates to 0.27 kg/kW (0.44 lb/HP) based on the 31.8 kg (70 lbs) weight for the 120 kW (160 HP) peak power output motor.

This case study has assumed a conservative electrical controller weight of 15 kg based on the specific weight of 0.03 kg/kW predicted by Patterson et al. (2012) combined with a factor of 2 to account for predicted uncertainty.

### 3.2.3 Climb performance model

An aircraft climb performance model was developed to support the conceptual sizing and quantisation of the morphological matrix as described in the following sections. A two-step process was followed where the outputs of this model were compared to Cessna Aircraft Company – Cessna 182P Pilots Operating Handbook (1976) data at the same climb conditions in order to validate the model. The second major step used the same model, modified to exclude altitude density power effects, to provide estimates of electric aircraft climb rate and time and climb energy requirements.

Table 65 provides the mission requirements, operational data and Cessna 182P specifications which were inputs to this aircraft climb performance model. The Cessna Aircraft Company – Cessna 182P Pilots Operating Handbook (1976) presents climb data as performance charts at maximum takeoff weight, flaps up, 2600 engine RPM and standard temperature.

Table 65. Cessna 182P mission and specification data

| Mission requirements | |
|---|---|
| Altitude (ft) | 14000 |
| Flights/charge | 1 |
| MTOW (lbs) | 2950 |
| MTOW (kg) | 1338 |
| ROC - mins to FL 140 | 10 |
| Average ROC (fpm) | 1400 |
| | |
| **Current operational weights** | |
| Skydiver weight (kg) - 4 jumpers | 340 |
| Pilot (kg) | 85 |
| Fuel - 60 litres | 43.2 |
| Total load (kg) | 468.2 |
| | |
| **Cessna Skylane II - C182P data - POH data** | |
| Empty weight (lbs) | 1771 |
| Empty weight (kg) | 803 |
| MTOW (lbs) | 2950 |
| MTOW (kg) | 1338 |
| Useful load (lbs) | 1179 |
| Useful load (kg) | 535 |
| Engine power max (HP) | 230 |
| Engine power max (kW) | 171.5 |
| Wing area (ft^2) | 174 |
| Wing area (m^2) | 16.2 |

Table 66 shows the climb data from the Cessna Aircraft Company – Cessna 182P Pilots Operating Handbook (1976), with an additional three columns added to account for climb time interval and to calculate energy requirements for each 1000 ft altitude interval. The last two columns shown in Table 66 are calculated from engine power (Power – Alt) at the given altitude and the time taken to climb to that altitude (delT). This is presented in HP.h or kW.h respectively. The total energy requirements to climb to the 14000 ft altitude is shown by the totals for each altitude interval and is provided in each column. This calculation method was used for all EP system climb energy predictions described in this section.

The energy requirement during the climb was dependent on engine power output and climb time interval. Therefore, engine power output, which was dependent on density effect, was predicted using the Wright equation provided in Raymer (2003). This density effect on engine power output is given by the relationship:

$$P = P_0\left[\frac{\rho}{\rho_0}\right] - \frac{1-\frac{\rho}{\rho_0}}{7.55} \qquad \text{Equation 38}$$

Where:

$P$ is the engine power at altitude

$P_0$ is the engine power at sea level

$\rho$ is the air density

$\rho_0$ is the air density at sea level

Table 66. Baseline - Cessna 182P Pilot Operating Handbook Climb Data

| Altitude (ft) | KIAS - POH | ROC (fpm) - POH | Time (min) | delT (sec) | sigma | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|
| 0 | 80 | 890 | 0 | 0 | 1.0000 | 230 | 0.0 | 0.0 |
| 1000 | 80 | 845 | 1 | 67 | 0.9711 | 222 | 4.2 | 3.1 |
| 2000 | 79 | 800 | 2 | 71 | 0.9428 | 215 | 4.2 | 3.2 |
| 3000 | 78 | 755 | 4 | 75 | 0.9151 | 208 | 4.3 | 3.2 |
| 4000 | 78 | 710 | 5 | 79 | 0.8881 | 201 | 4.4 | 3.3 |
| 5000 | 77 | 665 | 7 | 85 | 0.8617 | 194 | 4.6 | 3.4 |
| 6000 | 76 | 620 | 8 | 90 | 0.8359 | 187 | 4.7 | 3.5 |
| 7000 | 75 | 575 | 10 | 97 | 0.8107 | 181 | 4.9 | 3.6 |
| 8000 | 75 | 535 | 12 | 104 | 0.7860 | 174 | 5.1 | 3.8 |
| 9000 | 74 | 490 | 14 | 112 | 0.762 | 168 | 5.2 | 3.9 |
| 10000 | 73 | 445 | 16 | 122 | 0.7385 | 162 | 5.5 | 4.1 |
| 11000 | 73 | 400 | 18 | 135 | 0.7156 | 156 | 5.8 | 4.3 |
| 12000 | 72 | 355 | 21 | 150 | 0.6932 | 150 | 6.3 | 4.7 |
| 13000 | 71 | *310* | | 194 | 0.6713 | 144 | 7.8 | 5.8 |
| 14000 | 70 | *266* | | 226 | 0.6500 | 139 | 8.7 | 6.5 |
| | | | | 1607 | s | | **75.6** | **56.3** |
| | | | To 14000ft | **27** | min | | | |
| | | | | 903 | s | | | |
| | | | To 10000ft | **15** | min | | | |

An equivalent aircraft climb performance model was developed using the rate of climb expression for naturally aspirated IC engines given by Hale (1984). This model used the same baseline Cessna 182P mission and specification data as shown in Table 65.

Hale (1984) presents an equation for fastest climb as follows:

$$R/C_{max} = 550\eta_p\sigma\left(\frac{HP}{W}\right) - \frac{V_{Pmin}}{0.866\sigma^{1/2}E_m} \qquad \text{Equation 39}$$

Where

$\eta_p$ is the propeller efficiency

*(HP/W)* is the aircraft power loading

$V_{Pmin}$ is the minimum drag-power airspeed

σ is the atmospheric density ratio

$E_m$ is the maximum lift to drag ratio, $C_L/C_D$

The predicted climb performance is shown in Table 67, and this provides estimates Rate of Climb (ROC), time of climb (delT), and total energy requirements using the climb expression as noted above. The same general methods are used to generate these parameters as presented above in Table 66. Given that no data could be obtained for Cessna 182P propeller efficiency $\eta_p$, values ranging from 0.65 to 0.70 were assumed for the climb profile based on the matching of rate of climb data at various altitude intervals. The coefficient of lift ($C_L$) values at each altitude were calculated using the lift equation and the trim condition at each climb altitude. The drag coefficient ($C_D$) was calculated from the drag polar for the Cessna 182 aircraft as presented Chaun-Tau, Lan & Roskam (2008). This drag polar was defined by the following relationship:

$$C_D = 0.0293+0.0506C_L^2 \quad \text{Equation 40}$$

Again, the density effect on engine power output was predicted using the Wright equation provided in Raymer (2003).

The rate of climb and time of climb values shown in Table 67 compare favourably with the Cessna Aircraft Company – Cessna 182P Pilots Operating Handbook (1976) data shown in Table 66, noting that it is difficult to resolve these figures to any greater accuracy using flight test techniques. More importantly the selection of propeller efficiency $\eta_p$, values, based on rate of climb data as described above, matched total climb duration to 10000 ft and 14000 ft altitude results respectively. Therefore, this aircraft climb performance model was considered adequate to predict electric propulsion system performance with the appropriate changes and corrections as described in the following section.

Table 67. Predicted performance using Hale (1984) R/$C_{max}$ method for aspirated engines

| Altitude (ft) | sigma | rho | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 80 | 135 | 0.65 | 0.78 | 0.060 | 13.0 | 952 | 0 | 230 | 0.0 | 0.0 |
| 1000 | 0.9711 | 0.002308 | 80 | 135 | 0.65 | 0.81 | 0.062 | 13.0 | 892 | 63 | 222 | 3.9 | 2.9 |
| 2000 | 0.9428 | 0.002241 | 79 | 133 | 0.66 | 0.85 | 0.066 | 12.9 | 852 | 67 | 215 | 4.0 | 3.0 |
| 3000 | 0.9151 | 0.002175 | 78 | 132 | 0.66 | 0.90 | 0.070 | 12.8 | 809 | 70 | 208 | 4.1 | 3.0 |
| 4000 | 0.8881 | 0.002111 | 78 | 132 | 0.66 | 0.93 | 0.073 | 12.7 | 748 | 74 | 201 | 4.1 | 3.1 |
| 5000 | 0.8617 | 0.002048 | 77 | 130 | 0.67 | 0.98 | 0.078 | 12.6 | 703 | 80 | 194 | 4.3 | 3.2 |
| 6000 | 0.8359 | 0.001987 | 76 | 128 | 0.67 | 1.04 | 0.084 | 12.4 | 656 | 85 | 187 | 4.4 | 3.3 |
| 7000 | 0.8107 | 0.001927 | 75 | 127 | 0.67 | 1.10 | 0.090 | 12.2 | 596 | 91 | 181 | 4.6 | 3.4 |
| 8000 | 0.7860 | 0.001868 | 75 | 127 | 0.68 | 1.13 | 0.094 | 12.0 | 542 | 101 | 174 | 4.9 | 3.6 |
| 9000 | 0.762 | 0.001811 | 74 | 125 | 0.68 | 1.20 | 0.102 | 11.7 | 489 | 111 | 168 | 5.2 | 3.8 |
| 10000 | 0.7385 | 0.001755 | 73 | 123 | 0.68 | 1.27 | 0.111 | 11.4 | 424 | 123 | 162 | 5.5 | 4.1 |
| 11000 | 0.7156 | 0.001701 | 73 | 123 | 0.69 | 1.31 | 0.117 | 11.3 | 366 | 142 | 156 | 6.1 | 4.6 |
| 12000 | 0.6932 | 0.001648 | 72 | 122 | 0.69 | 1.39 | 0.128 | 10.9 | 305 | 164 | 150 | 6.8 | 5.1 |
| 13000 | 0.6713 | 0.001596 | 71 | 120 | 0.69 | 1.48 | 0.140 | 10.6 | 232 | 197 | 144 | 7.9 | 5.9 |
| 14000 | 0.6500 | 0.001545 | 70 | 118 | 0.70 | 1.57 | 0.154 | 10.2 | 165 | 258 | 139 | 10.0 | 7.4 |
| | | | | | | | | | | 1627 | s | **75.8** | **56.5** |
| | | | | | | | | | To 14000ft | **27** | min | | |
| | | | | | | | | | | 866 | s | | |
| | | | | | | | | | To 10000ft | **14** | min | | |

#### 3.2.3.1 *Electric propulsion climb performance and energy estimates*

A similar computational approach as described in the previous section was used to undertake electric propulsion system initial sizing, climb performance predictions and energy estimates. In this case the relevant power characteristics of each electric motor combination was determined from Table 68.

##### 3.2.3.1.1 *Selection of motors*

The sample of motors selected in this analysis is based on existing motor technology, and not motors in development at time of writing. It has been noted that there are several motors in development for specific aviation applications, all showing promising performance. However, these motors have been excluded, as insufficient data exists at this time.

##### 3.2.3.1.2 *Motor performance aspects*

It was highlighted in the review of electric motor performance by Glassock (2018, pers. comm., 8 October) that electric motors shown in Table 68 may be subject to time limitations at the maximum power setting. No data could be obtained to determine the time that maximum power could be sustained in a typical aircraft installation, noting that the skydiving mission profile climb segment requirement necessitates maximum power for a duration of 10 minutes. As discussed earlier, most motors presented in Table 68 were not developed for aeronautical applications, with the exceptions being the Siemens AG260D, and the Contra-Electric 2X YASA 750 axial flux series motors. Both of these motors have been specifically developed for aviation, with the Siemens AG260D motor being flight proven at time of writing (Siemens AG 2016).

The remaining motors shown in Table 68 have not been modified or adapted for aviation applications, Furthermore the maximum power performance characteristics of these motors will be dependent on the specifics of the aircraft installation and cooling arrangements, which is not known in this early conceptual design phase. Given these uncertainties, this case study has adopted the **continuous** power figures in this analysis as recommended in correspondence with Glassock (2018, pers. comm., 8 October). Nevertheless, the time limits at maximum power will need to be established in the next design phase, where the specifics of the installation will be defined, and prototype testing undertaken to determine the actual motor performance.

For this reason, the conceptual design methodology would note the potential uncertainty in motor power performance as a risk, to be added to the risk matrix described in Table 94. In addition, this motor continuous power performance attribute would also form a requirement of Design Specification Document (DSD), which would be updated or refined in the next design phase.

Table 68. Electric motor specifications and characteristics

| Aircraft/Motor description | Peak/max power (kW) | Mass (kg) | Peak/max specific power (kW/kg) | Torque (Nm) | Torque density (Nm/kg) | Continuous power (kW) | Efficiency (%) | Cost ($US)/kW | Reference | Comments |
|---|---|---|---|---|---|---|---|---|---|---|
| **Contra-Electric 2X YASA 750 axial flux series** | 225 | 66.0 | 3.41 | 790 | 12.0 | **150** | >95 | ---- | ContraElectric Propulsion Ltd (2018) accessed at http://www.contraelectric.com/innovation/crps-specification | This total mass 95 kg includes 2X motors as part of the contra-rotating system. Contra-rotating prop is 29 kg |
| **Emrax 268** | 230 | 20.3 | 11.33 | 500 | 24.6 | **100** | 98 | ---- | EMRAX Innovative E-Motors, (2018) accessed at http://emrax.com/products/emrax-268/ | |
| **Emrax 348** | 300 | 40 | 7.50 | 1000 | 25.0 | **150** | 98 | ---- | EMRAX Innovative E-Motors, (2018) accessed at http://emrax.com/products/emrax-348/ | |
| **Extra 330LE - Siemens AG SP260D** | 260 | 50 | 5.20 | 1000 | 20.0 | **230** | 95 | ---- | Siemens AG (2016) accessed at https://www.siemens.com/press/pool/de/feature/2015/corporate/2015-03-electromotor/factsheet-erstflug-weltrekordmotor-d.pdf<br>Siemens AG (2016) accessed at https://www.siemens.com/press//pool/de/pressemitteilungen/2016/corporate/PR2016120105COEN.pdf | |
| **Parametric motor cost** | ---- | ---- | ---- | ---- | ---- | ---- | ---- | 50 | Stoll AM and Veble Mikic G (2016) | |

The equivalent aircraft climb performance model was developed using the rate of climb expression for turbocharged IC engines operated below the critical altitude as given by Hale (1984). The difference in this expression is the omission of the density ratio σ in the first term of this equation.

Hale (1984) presents an equation for fastest climb as follows:

$$R/C_{max} = 550\eta_p\left(\frac{HP}{W}\right) - \frac{V_{Pmin}}{0.866\sigma^{1/2}E_m} \quad \text{Equation 41}$$

Where

$\eta_p$ is the propeller efficiency

*(HP/W)* is the aircraft power loading

$V_{Pmin}$ is the minimum drag-power airspeed

$E_m$ is the maximum lift to drag ratio, $C_L/C_D$

This model used the same baseline Cessna 182P mission and specification data as shown in Table 65 and the same calculation methods along with the same climb speed profiles at standard day atmospheric conditions.

Corrections were applied to the climb performance data in three areas where applicable. These corrections consisted of a contrarotating propeller approximation for increased propulsive efficiency, an allowance for recuperative power generation using a specially developed propeller, and a drag increment resulting from the addition of the Slipper power pod (pod mounted underneath the fuselage). These corrections were applied to the applicable climb performance prediction results either within the applicable spreadsheet, or within the quantisation matrix. In the case of the Slipper power pod the impact of the drag increment was applied to total drag and incorporated in the climb performance model. Climb performance predictions therefore reflected this drag increment where applicable. The other impact to drag which was modelled was the cooling drag reduction resulting from the removal of IC engine. As this cooling drag was the same for all configurations it was not incorporated into the actual climb performance modelling. Rather it was presented in the quantisation to account for this drag decrement.

#### 3.2.3.2 Contrarotating propeller efficiency corrections

Contrarotating propellers can offer significant increase in propulsive efficiency, particularly when high activity factors and large numbers of blades can be employed. However, they do result in an increase in weight and this is estimated in the quantisation of the morphological matrix by a 30% increase over the standard propeller installation weight. There is limited design data for contrarotating propeller setups. However, Nicolai & Carichner (2010) state that the induced efficiency of a contrarotating propeller may be found by using the expression:

$$\eta_i = \eta_i' + \Delta\eta_i + 0.6\frac{P_R}{P} \qquad \text{Equation 42}$$

Where the term $\eta_i' + \Delta\eta_i$ is the actual induced efficiency and is incorporated in the existing propeller efficiency estimates. The latter term $0.6\frac{P_R}{P}$ can be evaluated using the efficiency correction chart for dual rotation propellers provided in Nicolai & Carichner (2010). This chart is reproduced here as Figure 66.

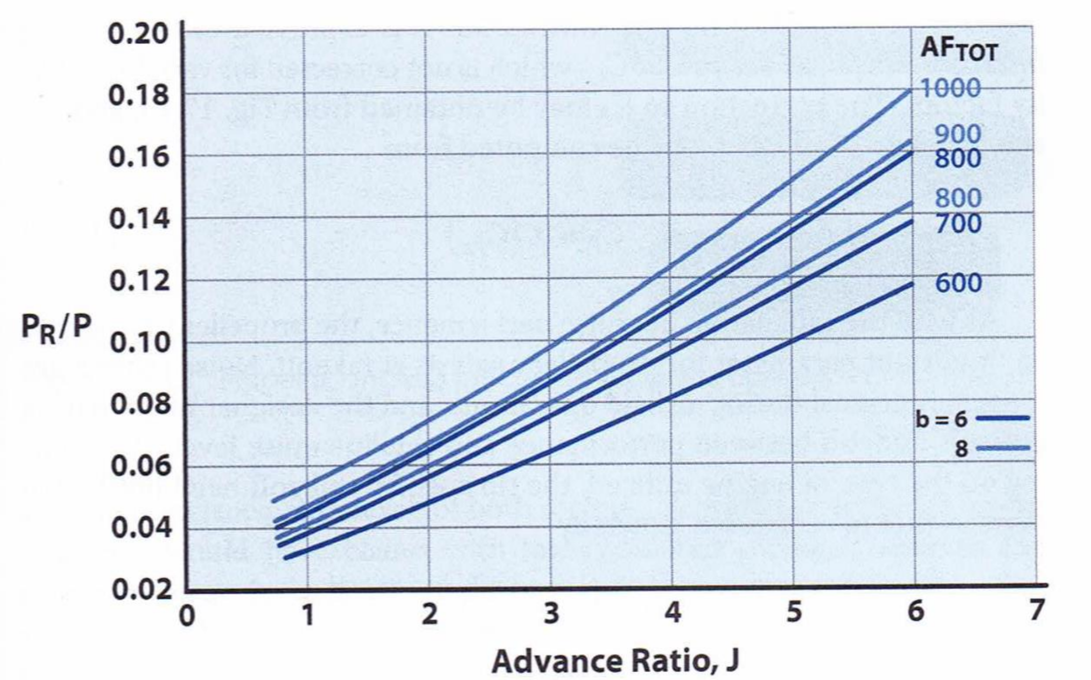



Figure 66. Efficiency correction for dual rotation propellers
Nicolai & Carichner (2010)

It is assumed that a 6-blade setup be used in contrarotating configurations consisting of 3 blades on each rotating element. Furthermore, it is assumed that the

Activity Factor (*AF*) is 100 for a standard fixed pitch propeller configuration (Chuan-Tau, Lan & Roskam, 2008). This provides a total *AF* = 100 × 6 = 600. The advance ratio (*J*) is determined by the relationship:

$$J = \frac{V}{nD} \qquad \text{Equation 43}$$

Where

*V* is the flight velocity at the associated operating condition

*n* is the propeller revolutions per second

*D* is the propeller diameter

Therefore $J = \frac{135}{45 \times 5.7} = 0.53$, based on a 5.7 ft (1.74 m) diameter propeller operating at 2700 RPM, at 80 KCAS climb speed.

Therefore, based on Figure 66, the $P_R/P$ ratio is determined as follows:

$$0.6\frac{P_R}{P} = 0.6 \times 0.03 = 0.018$$

This correction is applied to contrarotating setup propeller efficiency figures corresponding to the Contra Electric – 2xYASA 750 Series Axial Flux configurations. Further modelling would be required in order to establish accurate contrarotating propeller propulsive efficiencies, which could be achieved via testing of actual setups. However, this is out of scope of this thesis.

#### *3.2.3.3 Recuperative propeller energy correction*

The second correction involved an energy allowance for recuperative power generation using a specially developed propeller. The paper by Erzen et al. (2018) details the design and operation of a propeller that exploits the in-flight power recuperation of power on an electric aircraft during the descent phases of flight. This paper described an optimised propeller that could act as a wind turbine as well as operate as a conventional fixed-pitch propeller. It was found from this study that this propeller showed a 19% reduction in energy consumption within climb/descent manoeuvres. This net energy reduction was calculated and applied in the quantisation as a percentage of total climb energy as a *simple* approximation and presented in the climb energy results. The 19% reduction in energy was applied in the morphological matrix quantisation depending on whether a conventional propeller or recuperative propeller concept was selected.

### *3.2.3.4 Installation drag increment*

The third correction involves a drag increment resulting from the Slipper power pod installation. This Slipper power pod was installed beneath the fuselage between the main landing gear in an arrangement similar to the fuselage belly pod described in Appendix 2. Figure 67 highlights the Slipper power pod mounted beneath the fuselage. In order to simplify this analysis, it is assumed that the drag increment due to this Slipper power pod is the same as the fuselage belly tank as described in Appendix 2 which is:

Average LNG fuel tank drag contribution, $C_{DLNG}$ = **0.0019**

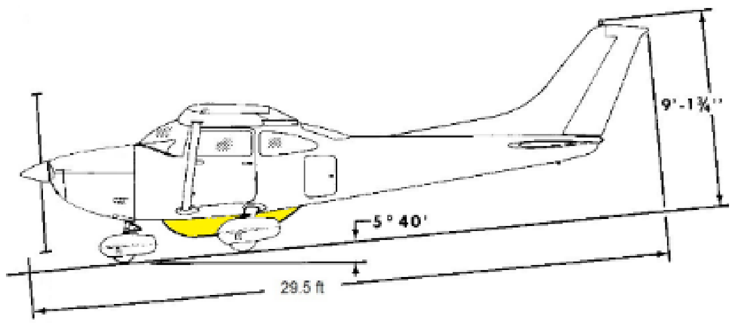



Figure 67. Slipper power pod installation beneath fuselage

As before the Cessna 421B wing reference area $S_{ref}$ = 215 ft$^2$

The Cessna 182P wing reference area $S_{ref}$ = 174 ft$^2$

Based on aircraft reference areas the $C_D$ of the Slipper power pod can be calculated:

Slipper power pod drag coefficient $C_{Dpod} = C_{DLNG} \times \left(\frac{S_{ref\_C182P}}{S_{ref\_C421B}}\right)$ Equation 44

$$C_{Dpod} = 0.0019 \times \left(\frac{174}{215}\right)$$

$$C_{Dpod} = \underline{0.0015}$$

This can be expressed in terms of $D/q$ = 0.0015 × 174 = 0.2676 ft$^2$

The removal of the IC engine and its replacement with the equivalent electric motor results in a reduction in cooling drag. As stated by Raymer (2012) cooling drag represents the momentum loss of the air taken into the cowling and passed over the engine for cooling. It is noted however that there remains a miscellaneous drag increment corresponding to electric motor, speed controller and battery cooling requirements. However, at this early stage in conceptual design the actual increment in miscellaneous cooling drag resulting from the electric propulsion system installation is difficult to estimate. Although this drag increment is omitted in this analysis it may not be negligible, and therefore the recommended next design phase estimate this drag component.

Raymer (2012) provides an expression for cooling drag as follows:

$$(D/q)_{cooling} = (4.9 \times 10^{-7})\frac{HP.T^2}{\sigma V} \qquad \text{Equation 45}$$

Where:

$T$ is the air temperature in K

$V$ is the velocity in ft/s

$HP$ is the engine horsepower

$\sigma$ is the atmospheric density ratio

The engine cooling drag contribution resulting from removal of the 230 HP engine can be evaluated at each altitude, as follows:

$$(D/q)_{cooling} = (4.9 \times 10^{-7})\frac{230 \times 288.2^2}{1 \times 135}$$

$$(D/q)_{cooling} = 0.0693 \text{ ft}^2$$

This value of cooling drag is evaluated at each altitude increment and the average value presented in the morphological matrix quantisation.

A computational approach as described in the previous section was used to undertake electric propulsion system initial sizing, climb performance predictions and energy estimates, and incorporated these corrections to each configuration, where applicable. The results of this modelling are shown by Table 69 through Table 76 for the range of electric motor types summarised in Table 68, propeller types, and Slipper power pod installation. As stated earlier the energy recovery characteristics resulting from the recuperative propeller installation is calculated as a 19% of climb energy for all predictions and is applied in the quantisation matrix where applicable.

The contrarotating propeller propulsive efficiency increment was incorporated as a function of the total climb energy component and added to the quantisation matrix where applicable to this propeller type.

It is necessary that reserve energy requirements are incorporated into this model as this adds additional energy requirements and battery weight. This reserve energy estimate was based on aircraft operational holding requirements for a skydiving mission which comprised 45 minutes reserve power holding at the best *L/D* ratio airspeed. This was a constant energy requirement added to the total climb energy and was calculated by determining the trim coefficient of lift $C_L$ at the best *L/D* ratio airspeed at maximum weight, which resulted in:

$C_L$ = 0.78 at best *L/D* ratio airspeed 80 KCAS

Drag coefficient can then be calculated using the drag polar shown in Chaun-Tau, Lan & Roskam (2008) giving:

$C_D = 0.0293+0.0506C_L^2$

$C_D = 0.0602$

Drag can be calculated for this airspeed as follows:

D = 227.3 lbs

And Thrust = Drag = 227.3 lbs, where power can be calculated by the following:

$$P = \frac{T.V}{550} = \frac{227.3\times135}{550} = 55.8 \text{ HP or } 41.6 \text{ kW}$$

$$E_{reserve} = 41.6\times0.75 = 31.2 \text{ kWh}$$

Table 69 through Table 76 shows the electric propulsion system initial sizing, climb performance predictions and energy estimates, incorporating these corrections and estimates as described above.

### 3.2.4 Initial sizing performance outcomes

Only the Siemens AG SP260D electric motor /propeller combination satisfied the sub 10-minute time to climb to 14000 ft altitude requirement. The Contra Electric – 2 X YASA 750 Series Axial Flux motor combination, and the Emrax 348 motor provided climb performance comparable to the IC engine powered Cessna 182 aircraft. However, they do not offer any significant improvements that could be considered to justify adoption of these options. Given that the Emrax 268 motor exhibited much lower climb performance due to the lower continuous power output, this motor was omitted from the morphological analysis.

Although only one motor/propeller combination satisfied the time of climb requirement, three motor combinations are analysed in the morphological matrix to illustrate the methodology.

Table 69. Climb performance prediction – Siemens AG SP260D

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.67 | 0.78 | 0.060 | 13.0 | 1591 | 0 | 308.4 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.67 | 0.81 | 0.063 | 13.0 | 1577 | 38 | 308.4 | 3.2 | 2.4 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.67 | 0.85 | 0.066 | 12.9 | 1574 | 38 | 308.4 | 3.3 | 2.4 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.67 | 0.90 | 0.070 | 12.8 | 1567 | 38 | 308.4 | 3.3 | 2.4 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.67 | 0.93 | 0.073 | 12.7 | 1552 | 38 | 308.4 | 3.3 | 2.4 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.67 | 0.98 | 0.078 | 12.6 | 1541 | 39 | 308.4 | 3.3 | 2.5 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.67 | 1.04 | 0.084 | 12.4 | 1527 | 39 | 308.4 | 3.3 | 2.5 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.67 | 1.10 | 0.090 | 12.2 | 1511 | 39 | 308.4 | 3.4 | 2.5 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.67 | 1.13 | 0.094 | 12.0 | 1489 | 40 | 308.4 | 3.4 | 2.5 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.67 | 1.20 | 0.102 | 11.7 | 1468 | 40 | 308.4 | 3.5 | 2.6 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.67 | 1.27 | 0.111 | 11.4 | 1443 | 41 | 308.4 | 3.5 | 2.6 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.67 | 1.31 | 0.117 | 11.3 | 1416 | 42 | 308.4 | 3.6 | 2.7 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.67 | 1.39 | 0.128 | 10.9 | 1386 | 42 | 308.4 | 3.6 | 2.7 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.67 | 1.48 | 0.140 | 10.6 | 1352 | 43 | 308.4 | 3.7 | 2.8 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.67 | 1.57 | 0.154 | 10.2 | 1315 | 44 | 308.4 | 3.8 | 2.8 |
| | | | | | | | | | | | 562 | s | **48.1** | **35.8** |
| | | | | | | | | | | To 14000ft | **9** | min | Recuperation | -6.9 |
| | | | | | | | | | | | 390 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **6** | min | **Total** | **60.1** |

Table 70. Climb performance prediction – Contra Electric – 2 X YASA 750 Series Axial Flux

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.69 | 0.78 | 0.060 | 13.0 | 827 | 0 | 201.2 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.69 | 0.81 | 0.063 | 13.0 | 813 | 73 | 201.2 | 4.1 | 3.0 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.69 | 0.85 | 0.066 | 12.9 | 811 | 74 | 201.2 | 4.1 | 3.1 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.69 | 0.90 | 0.070 | 12.8 | 804 | 74 | 201.2 | 4.1 | 3.1 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.69 | 0.93 | 0.073 | 12.7 | 788 | 75 | 201.2 | 4.2 | 3.1 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.69 | 0.98 | 0.078 | 12.6 | 777 | 76 | 201.2 | 4.3 | 3.2 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.69 | 1.04 | 0.084 | 12.4 | 763 | 77 | 201.2 | 4.3 | 3.2 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.69 | 1.10 | 0.090 | 12.2 | 747 | 79 | 201.2 | 4.4 | 3.3 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.69 | 1.13 | 0.094 | 12.0 | 725 | 80 | 201.2 | 4.5 | 3.3 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.69 | 1.20 | 0.102 | 11.7 | 704 | 83 | 201.2 | 4.6 | 3.4 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.69 | 1.27 | 0.111 | 11.4 | 680 | 85 | 201.2 | 4.8 | 3.5 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.69 | 1.31 | 0.117 | 11.3 | 652 | 88 | 201.2 | 4.9 | 3.7 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.69 | 1.39 | 0.128 | 10.9 | 622 | 92 | 201.2 | 5.1 | 3.8 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.69 | 1.48 | 0.140 | 10.6 | 589 | 96 | 201.2 | 5.4 | 4.0 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.69 | 1.57 | 0.154 | 10.2 | 551 | 102 | 201.2 | 5.7 | 4.2 |
| | | | | | | | | | | | 1154 | s | **64.5** | **48.0** |
| | | | | | | | | | | To 14000ft | **19** | min | Recuperation | -9.2 |
| | | | | | | | | | | | 775 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **13** | min | **Total** | **70.0** |

Table 71. Climb performance prediction – Emrax 268

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.67 | 0.78 | 0.060 | 13.0 | 284 | 0 | 134.1 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.67 | 0.81 | 0.063 | 13.0 | 270 | 211 | 134.1 | 7.9 | 5.9 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.67 | 0.85 | 0.066 | 12.9 | 268 | 222 | 134.1 | 8.3 | 6.2 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.67 | 0.90 | 0.070 | 12.8 | 260 | 224 | 134.1 | 8.3 | 6.2 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.67 | 0.93 | 0.073 | 12.7 | 245 | 230 | 134.1 | 8.6 | 6.4 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.67 | 0.98 | 0.078 | 12.6 | 234 | 245 | 134.1 | 9.1 | 6.8 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.67 | 1.04 | 0.084 | 12.4 | 220 | 256 | 134.1 | 9.6 | 7.1 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.67 | 1.10 | 0.090 | 12.2 | 204 | 272 | 134.1 | 10.1 | 7.6 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.67 | 1.13 | 0.094 | 12.0 | 182 | 294 | 134.1 | 11.0 | 8.2 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.67 | 1.20 | 0.102 | 11.7 | 161 | 329 | 134.1 | 12.3 | 9.1 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.67 | 1.27 | 0.111 | 11.4 | 137 | 372 | 134.1 | 13.9 | 10.3 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.67 | 1.31 | 0.117 | 11.3 | 109 | 439 | 134.1 | 16.3 | 12.2 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.67 | 1.39 | 0.128 | 10.9 | 79 | 548 | 134.1 | 20.4 | 15.2 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.67 | 1.48 | 0.140 | 10.6 | 46 | 755 | 134.1 | 28.1 | 21.0 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.67 | 1.57 | 0.154 | 10.2 | 8 | 1313 | 134.1 | 48.9 | 36.4 |
| | | | | | | | | | | | 5713 | s | **212.8** | **158.5** |
| | | | | | | | | | | To 14000ft | **95** | min | Recuperation | -30.5 |
| | | | | | | | | | | | 2657 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **44** | min | **Total** | **159.2** |

Table 72. Climb performance prediction – Emrax 348

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.67 | 0.78 | 0.060 | 13.0 | 787 | 0 | 201.2 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.67 | 0.81 | 0.063 | 13.0 | 772 | 76 | 201.2 | 4.3 | 3.2 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.67 | 0.85 | 0.066 | 12.9 | 770 | 78 | 201.2 | 4.3 | 3.2 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.67 | 0.90 | 0.070 | 12.8 | 763 | 78 | 201.2 | 4.4 | 3.2 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.67 | 0.93 | 0.073 | 12.7 | 748 | 79 | 201.2 | 4.4 | 3.3 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.67 | 0.98 | 0.078 | 12.6 | 737 | 80 | 201.2 | 4.5 | 3.3 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.67 | 1.04 | 0.084 | 12.4 | 723 | 81 | 201.2 | 4.6 | 3.4 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.67 | 1.10 | 0.090 | 12.2 | 706 | 83 | 201.2 | 4.6 | 3.5 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.67 | 1.13 | 0.094 | 12.0 | 685 | 85 | 201.2 | 4.7 | 3.5 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.67 | 1.20 | 0.102 | 11.7 | 664 | 88 | 201.2 | 4.9 | 3.6 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.67 | 1.27 | 0.111 | 11.4 | 639 | 90 | 201.2 | 5.1 | 3.8 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.67 | 1.31 | 0.117 | 11.3 | 612 | 94 | 201.2 | 5.2 | 3.9 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.67 | 1.39 | 0.128 | 10.9 | 582 | 98 | 201.2 | 5.5 | 4.1 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.67 | 1.48 | 0.140 | 10.6 | 548 | 103 | 201.2 | 5.8 | 4.3 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.67 | 1.57 | 0.154 | 10.2 | 511 | 109 | 201.2 | 6.1 | 4.6 |
| | | | | | | | | | | | 1223 | s | **68.3** | **50.9** |
| | | | | | | | | | | To 14000ft | **20** | min | Recuperation | -9.8 |
| | | | | | | | | | | | 818 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **14** | min | **Total** | **72.3** |

Table 73. Climb performance prediction – Siemens AG SP260D with Slipper power pod

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.67 | 0.78 | 0.062 | 12.7 | 1573 | 0 | 308.4 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.67 | 0.81 | 0.064 | 12.6 | 1558 | 38 | 308.4 | 3.3 | 2.4 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.67 | 0.85 | 0.067 | 12.6 | 1557 | 38 | 308.4 | 3.3 | 2.5 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.67 | 0.90 | 0.072 | 12.5 | 1551 | 39 | 308.4 | 3.3 | 2.5 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.67 | 0.93 | 0.074 | 12.5 | 1536 | 39 | 308.4 | 3.3 | 2.5 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.67 | 0.98 | 0.079 | 12.3 | 1525 | 39 | 308.4 | 3.3 | 2.5 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.67 | 1.04 | 0.085 | 12.2 | 1512 | 39 | 308.4 | 3.4 | 2.5 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.67 | 1.10 | 0.092 | 12.0 | 1497 | 40 | 308.4 | 3.4 | 2.5 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.67 | 1.13 | 0.096 | 11.8 | 1475 | 40 | 308.4 | 3.4 | 2.6 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.67 | 1.20 | 0.104 | 11.6 | 1455 | 41 | 308.4 | 3.5 | 2.6 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.67 | 1.27 | 0.113 | 11.3 | 1431 | 41 | 308.4 | 3.5 | 2.6 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.67 | 1.31 | 0.118 | 11.1 | 1404 | 42 | 308.4 | 3.6 | 2.7 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.67 | 1.39 | 0.129 | 10.8 | 1375 | 43 | 308.4 | 3.7 | 2.7 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.67 | 1.48 | 0.142 | 10.4 | 1342 | 44 | 308.4 | 3.7 | 2.8 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.67 | 1.57 | 0.156 | 10.1 | 1305 | 45 | 308.4 | 3.8 | 2.9 |
| | | | | | | | | | | | 567 | s | **48.6** | **36.2** |
| | | | | | | | | | | To 14000ft | **9** | min | Recuperation | -7.0 |
| | | | | | | | | | | | 394 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **7** | min | **Total** | **60.4** |

Table 74. Climb performance prediction – Contra Electric – 2 X YASA 750 Series Axial Flux with Slipper power pod

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.69 | 0.78 | 0.062 | 12.7 | 809 | 0 | 201.2 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.69 | 0.81 | 0.064 | 12.6 | 795 | 74 | 201.2 | 4.1 | 3.1 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.69 | 0.85 | 0.067 | 12.6 | 794 | 75 | 201.2 | 4.2 | 3.1 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.69 | 0.90 | 0.072 | 12.5 | 787 | 76 | 201.2 | 4.2 | 3.1 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.69 | 0.93 | 0.074 | 12.5 | 772 | 76 | 201.2 | 4.3 | 3.2 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.69 | 0.98 | 0.079 | 12.3 | 762 | 78 | 201.2 | 4.3 | 3.2 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.69 | 1.04 | 0.085 | 12.2 | 749 | 79 | 201.2 | 4.4 | 3.3 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.69 | 1.10 | 0.092 | 12.0 | 733 | 80 | 201.2 | 4.5 | 3.3 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.69 | 1.13 | 0.096 | 11.8 | 712 | 82 | 201.2 | 4.6 | 3.4 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.69 | 1.20 | 0.104 | 11.6 | 691 | 84 | 201.2 | 4.7 | 3.5 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.69 | 1.27 | 0.113 | 11.3 | 668 | 87 | 201.2 | 4.8 | 3.6 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.69 | 1.31 | 0.118 | 11.1 | 641 | 90 | 201.2 | 5.0 | 3.7 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.69 | 1.39 | 0.129 | 10.8 | 611 | 94 | 201.2 | 5.2 | 3.9 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.69 | 1.48 | 0.142 | 10.4 | 578 | 98 | 201.2 | 5.5 | 4.1 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.69 | 1.57 | 0.156 | 10.1 | 541 | 104 | 201.2 | 5.8 | 4.3 |
| | | | | | | | | | | | 1176 | s | **65.7** | **49.0** |
| | | | | | | | | | | To 14000ft | **20** | min | Recuperation | -9.4 |
| | | | | | | | | | | | 791 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **13** | min | **Total** | **70.7** |

## Table 75. Climb performance prediction – Emrax 268 with Slipper power pod

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.67 | 0.78 | 0.062 | 12.7 | 266 | 0 | 134.1 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.67 | 0.81 | 0.064 | 12.6 | 252 | 226 | 134.1 | 8.4 | 6.3 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.67 | 0.85 | 0.067 | 12.6 | 251 | 238 | 134.1 | 8.9 | 6.6 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.67 | 0.90 | 0.072 | 12.5 | 244 | 239 | 134.1 | 8.9 | 6.6 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.67 | 0.93 | 0.074 | 12.5 | 229 | 246 | 134.1 | 9.2 | 6.8 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.67 | 0.98 | 0.079 | 12.3 | 219 | 262 | 134.1 | 9.8 | 7.3 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.67 | 1.04 | 0.085 | 12.2 | 206 | 274 | 134.1 | 10.2 | 7.6 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.67 | 1.10 | 0.092 | 12.0 | 190 | 291 | 134.1 | 10.9 | 8.1 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.67 | 1.13 | 0.096 | 11.8 | 169 | 315 | 134.1 | 11.7 | 8.7 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.67 | 1.20 | 0.104 | 11.6 | 148 | 356 | 134.1 | 13.2 | 9.9 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.67 | 1.27 | 0.113 | 11.3 | 125 | 404 | 134.1 | 15.1 | 11.2 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.67 | 1.31 | 0.118 | 11.1 | 98 | 481 | 134.1 | 17.9 | 13.3 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.67 | 1.39 | 0.129 | 10.8 | 68 | 615 | 134.1 | 22.9 | 17.1 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.67 | 1.48 | 0.142 | 10.4 | 35 | 879 | 134.1 | 32.7 | 24.4 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.67 | 1.57 | 0.156 | 10.1 | -2 | 1706 | 134.1 | 63.5 | 47.3 |
| | | | | | | | | | | | 6533 | s | **243.3** | **181.2** |
| | | | | | | | | | | To 14000ft | **109** | min | Recuperation | -34.8 |
| | | | | | | | | | | | 2852 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **48** | min | **Total** | **177.6** |

## Table 76. Climb performance prediction – Emrax 348 with Slipper power pod

| Altitude (ft) | sigma | rho | Temp (K) | KIAS - POH | Vp (fps) | n_p | CL | CD | CL/CD | ROC (fpm) | delT (sec) | Power - Alt (HP) | HP.h | kWh |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0 | 1.0000 | 0.002377 | 288.2 | 80 | 135 | 0.67 | 0.78 | 0.062 | 12.7 | 768 | 0 | 201.2 | 0.0 | 0.0 |
| 1000 | 0.9644 | 0.002292 | 286.2 | 80 | 135 | 0.67 | 0.81 | 0.064 | 12.6 | 754 | 78 | 201.2 | 4.4 | 3.2 |
| 2000 | 0.9428 | 0.002241 | 284.2 | 79 | 133 | 0.67 | 0.85 | 0.067 | 12.6 | 753 | 80 | 201.2 | 4.4 | 3.3 |
| 3000 | 0.9151 | 0.002175 | 282.2 | 78 | 132 | 0.67 | 0.90 | 0.072 | 12.5 | 747 | 80 | 201.2 | 4.5 | 3.3 |
| 4000 | 0.8881 | 0.002111 | 280.2 | 78 | 132 | 0.67 | 0.93 | 0.074 | 12.5 | 732 | 80 | 201.2 | 4.5 | 3.3 |
| 5000 | 0.8617 | 0.002048 | 278.3 | 77 | 130 | 0.67 | 0.98 | 0.079 | 12.3 | 721 | 82 | 201.2 | 4.6 | 3.4 |
| 6000 | 0.8359 | 0.001987 | 276.3 | 76 | 128 | 0.67 | 1.04 | 0.085 | 12.2 | 708 | 83 | 201.2 | 4.6 | 3.5 |
| 7000 | 0.8107 | 0.001927 | 274.3 | 75 | 127 | 0.67 | 1.10 | 0.092 | 12.0 | 693 | 85 | 201.2 | 4.7 | 3.5 |
| 8000 | 0.7860 | 0.001868 | 272.3 | 75 | 127 | 0.67 | 1.13 | 0.096 | 11.8 | 671 | 87 | 201.2 | 4.8 | 3.6 |
| 9000 | 0.762 | 0.001811 | 270.3 | 74 | 125 | 0.67 | 1.20 | 0.104 | 11.6 | 651 | 89 | 201.2 | 5.0 | 3.7 |
| 10000 | 0.7385 | 0.001755 | 268.3 | 73 | 123 | 0.67 | 1.27 | 0.113 | 11.3 | 627 | 92 | 201.2 | 5.2 | 3.8 |
| 11000 | 0.7156 | 0.001701 | 266.4 | 73 | 123 | 0.67 | 1.31 | 0.118 | 11.1 | 600 | 96 | 201.2 | 5.3 | 4.0 |
| 12000 | 0.6932 | 0.001648 | 264.4 | 72 | 122 | 0.67 | 1.39 | 0.129 | 10.8 | 571 | 100 | 201.2 | 5.6 | 4.2 |
| 13000 | 0.6713 | 0.001596 | 262.4 | 71 | 120 | 0.67 | 1.48 | 0.142 | 10.4 | 538 | 105 | 201.2 | 5.9 | 4.4 |
| 14000 | 0.6500 | 0.001545 | 260.4 | 70 | 118 | 0.67 | 1.57 | 0.156 | 10.1 | 501 | 112 | 201.2 | 6.2 | 4.6 |
| | | | | | | | | | | | 1248 | s | **69.7** | **51.9** |
| | | | | | | | | | | To 14000ft | **21** | min | Recuperation | -10.0 |
| | | | | | | | | | | | 836 | s | Reserve | 31.2 |
| | | | | | | | | | | To 10000ft | **14** | min | **Total** | **73.1** |

## 3.3 CONCEPT GENERATION – STEP 2

### 3.3.1 Overview

The structured generation of concepts within the design space is a challenge well suited to the morphological method as previously described in Chapter 4, Chapter 5 and Appendix 1. This approach can be augmented by quantisation of the matrix using properties of the key technical performance measures previously described as key metrics determined in the previous QFD step. This structured approach explores the complete design space, and down-selects the ‘best’ potential solutions using simple relationships based on minimisation and maximising these metrics as a Figure of Merit (*FoM*).

### 3.3.2 Morphological matrix

As described in Chapter 5 the morphological matrix is a technique that supports design synthesis through assisting the design team to identify and generate combinations of systems, subsystems or components. The morphological matrix is created by decomposing the main functions of the modification into sub-functions. These sub-functions are listed on the vertical axis of the matrix. The morphological matrix for an aircraft fuel system is shown in Table 77. In this the main functional decomposition comprises the propeller type (propulsor), which is shown on the first line of the morphological matrix.

Table 77. Electric propulsion system modification morphological matrix

| | Modification function | Selected solution | Alternative solutions | | | |
|---|---|---|---|---|---|---|
| **Aicraft segment** | Propulsor configuration | Single motor | Single motor | Two motor - contra-rotating | 0 | 0 |
| | Effector configuration | Conventional propellor(s) | Conventional propellor(s) | Recuperative propeller(s) | 0 | 0 |
| | Energy storage location 1 | Firewall forward | Firewall forward | None | 0 | 0 |
| | Energy storage location 2 | None | Fuselage slipper power pod | None | 0 | 0 |
| | Energy regulation & control | Single electric speed controller | Single electric speed controller | Dual electric speed controller | 0 | 0 |
| | Heat exchange | Speed controller cooling | Speed controller cooling | Speed ctrl & motor cooling | 0 | 0 |
| | Energy safety system | Circuit breakers | Circuit breakers | None | 0 | 0 |
| | Energy enable/disable | Electrical Master Switch (EMS) | Electrical Master Switch (EMS) | None | 0 | 0 |
| | Energy storage state monitoring | State of Charge (SOC) indicator | State of Charge (SOC) indicator | None | 0 | 0 |
| | Energy health monitoring | State of Health (SOH) indicator | State of Health (SOH) indicator | None | 0 | 0 |
| | Energy exchange/interface | Battery exchange | Recharge receptacle | Battery exchange | 0 | 0 |
| | | | | | | |
| **Ground infrastructure and delivery segment** | Energy supply type | Mains supply | Mains supply | 0 | 0 | 0 |
| | Energy transfer/charge | In aircraft charging | In aircraft charging | Battery exchange - External charge | 0 | 0 |
| | Energy charge function | AC Level 1 | AC Level 1 | AC Level 2 | DC Fast Charging (DCFC) | 0 |
| | Energy discharge | Discharge function | Discharge function | 0 | 0 | 0 |
| | Energy safety system | Circuit breakers | Circuit breakers | 0 | 0 | 0 |
| | Energy enable/disable | Power On/Off | Power On/Off | 0 | 0 | 0 |
| | Energy storage state monitoring | State of Charge (SOC) indicator | State of Charge (SOC) indicator | Nil | 0 | 0 |
| | Energy storage monitoring | State of Health (SOH) indicator | State of Health (SOH) indicator | Nil | 0 | 0 |
| | Energy charge/discharge interface | Electrical cable & connector | Electrical cable & connector | None | 0 | 0 |

### 3.3.3 Quantisation of the aircraft morphological matrix

As stated in Chapter 5, the major shortfall of the morphological matrix technique is the potentially large number of candidate solutions. For this reason, the morphological matrix has been restricted to EP systems only rather than include Hybrid-Electric concepts. Hybrid-Electric configurations are possible solutions. However, this analysis has been limited to EP system concepts only as the technology has the potential to fulfil the mission needs using established technologies with lower accompanying integration costs. This is a client design choice which limits the number of configurations in accordance with customer needs.

The approach taken in this thesis quantifies the morphological matrix using a methodology which is based on that described by Gavel et al. (2008) and as described in Chapter 5. In the framework presented here, the focus is on properties that can be quantified, such as propulsion system weights, aerodynamic drag and system cost.

The quantified matrix gives access to approximated properties of the complete system, with every potential sub-solution modelled either by physical or statistical equations, or a combination, as described in Chapter 5. The mathematical basis underpinning this methodology is described in Chapter 4, where the approach involves identifying the concept solution with the lowest weight ($W$), installation drag ($D$), and cost ($C$), whilst providing the highest the best useful load climb performance. In this case the models used is one of simple summation to provide a Figure of Merit and also a parametric Useful load-time to climb score as described below.

Table 78 through Table 81 show the respective quantified morphological matrices relating to conventional and recuperative propeller modification concepts. Although it is usual to present this information in one matrix, it has been separated here in four separate tables to facilitate presentation in this thesis. The upper rows of these matrices provide configuration details of each concept which are determined from the morphological matrices as described earlier. In this case the yellow highlighted region on these aircraft diagrams depict the Slipper power pod installation and the double propeller line denoted the contrarotating propeller arrangement. These diagrams are intended to provide a quick visual means to define each configuration concept.

#### *3.3.3.1 Metrics*

Rather than attempting to represent each metric as an objective function comprising all system components as described Gavel et al. (2008), the metrics as described in Table 78 through Table 81 have been simplified based on QFD requirements. Each requirement attribute is therefore represented as a metric and modelled to approximate the properties of the complete aircraft segment. These metrics equate to the EP system modification weight, aerodynamic drag and cost impacts, with these quantities either directly or indirectly derived from the QFD matrix. In this instance weight is further broken down into powerplant, propeller spinner and accessories, electric controller cabling and equipment, batteries and other. It should be noted that the QFD engineering attributes comprising *Power impact* and *Size Impact* are implicit in the design concept selection. That is *Power impact* will be a function of motor selection. *Size Impact* is also dependent on conceptual design selection where batteries are located to ensure no impact on passenger, baggage, or cargo space. Therefore, given that these attributes are implicit in concept selection these are not quantified here as metrics.

#### *3.3.3.2 Normalisation of metrics*

This quantisation is based on the metrics as shown in the left most column of each matrix shown by Table 78 to Table 81. These metrics are grouped into blocks which presents the data relevant to each metric such as itemised weight, weight removed/added, drag increments or costs. The final score relating to each metric, shown in **<u>bold underline</u>**, is then determined which shows the change impact. This score is expressed as a ratio in case of weight which is normalised to the configuration baseline. The configuration baseline used to normalise useful load, installed drag and cost metrics is Configuration 1, shown in Table 78, which equates to the Siemens AG SP260D concept. This same Configuration 1 baseline is used also for recuperative propeller series concepts shown in Table 80. Therefore, all scores are referenced to a common baseline.

#### *3.3.3.3 Metric weighting*

Each metric is weighted to account for importance in accordance with those weightings derived in the QFD matrix shown in Table 57. These weightings are shown on the left-hand side of the matrices shown in Table 78 through Table 81 and have been assigned to weight, installed drag and cost respectively. It should be noted that this weighting has focussed on the three highest rated engineering challenges as shown in the QFD matrix as described earlier.

#### *3.3.3.4 Weight estimates*

The weight estimates as shown in this quantisation rely on the traditional weight and balance (W&B) accounting methods for items added or removed as part of the EP modification. This process follows the basis of Table 61 and Table 62 where items added or removed are presented within categories related to powerplant weight, propeller/spinner and accessories weight, electric controller weight, battery weight and lastly other miscellaneous weights. These weights are presented in the quantisation of the morphological matrix by Table 78 through Table 81.

The weights of the various items are determined from actual component weights in case of the motor, as specified in Table 68, or are estimated from parametric relationship such as that for the batteries as specified in Table 64. The quantisation matrix as shown by Table 78 through Table 81 includes a “pull-down” option box to specify battery specific energy density, which can be varied between 150 to 270 Wh/kg. For the purposes of this thesis, the value of 250 Wh/kg is used as a near term **maxima**. This specific energy density coupled with the energy requirements for climb and reserves is used to calculate the battery weight for this skydiving mission.

Although the ideal solution is to the place the battery installation forward of the engine firewall, actual calculated battery, motor and electric controller weight exceeds the IC powerplant and accessories removed. This therefore introduces a W&B issue which needs to be addressed. Rather than undertake detailed W&B calculations as part of this quantisation, the approach has been to determine the residual battery weight (the weight that exceeds the powerplant weight removed) and then place these batteries close to the Centre-of-Gravity (CG) within the Slipper power pod. The quantisation includes a calculation that determines the residual battery weight installed in the Slipper power pod. Further it should be noted that those concepts that do not include the Slipper power pod for W&B are accordingly not viable solutions. These are denoted by the red text Figure of Merit values shown in the quantisation matrix.

#### *3.3.3.5 Installed drag*

The impact of additional drag resulting from the modification is determined by the drag contribution of cooling drag and Slipper power pod drag. Cooling drag is determined by inspection to be the main contributor to drag for this modification. The Slipper power pod drag contributions are determined from an analysis which accounts for location (beneath the fuselage) using the natural gas approximations as described in Appendix 2. Again, the drag contribution metric is accounted for as a normalised score compared to the baseline as described above.

#### *3.3.3.6 Propulsion system costs*

Installation cost is represented by the cost of the motor, propeller and batteries. Batteries are determined by inspection to be the most expensive contributor to the EP system costs, as this component is likely to be a custom developed solution. Battery costs have been factored by the specific energy density where batteries with lower specific energy density are proportionally lower cost. These cost estimates are determined by parametric values in Table 68 and Table 64 where the paper by Stoll and Veble Mikic G (2016) details the costs of electric motors and batteries. It should be noted that this quantised analysis does not include the battery lifecycle costs. The battery lifecycle costs are discussed in the flight costs evaluation step shown in Section 3.9.6.3.

The cost of electric controllers is also detailed by Stoll and Veble Mikic G (2016) as parametric cost relationship for such equipment. Propeller costs are determined as percentage of the motor cost and factored by 1.5 to account for the complexity of contrarotating propellers. Again, the cost contribution metric is accounted for as a normalised score compared to the baseline as described above.

#### *3.3.3.7 Normalised climb performance score*

In order to encapsulate flight performance associated with this quantisation analysis, an additional measure has been formulated that combines useful load and time of climb characteristics. This climb performance score is determined by the ratio of the useful load to time of climb, and is then normalised using the Configuration 1 baseline as described above. It is important to note that the normalised climb performance measure does not incorporate cost. Therefore, the normalised climb performance measure may not directly align with the Figure of Merit results described below.

#### 3.3.3.8 *Figure of Merit*

As described above these metrics are quantised by use of simple mathematical models allowing an approximation of the complete aircraft segment modification. These metrics can then be evaluated by a *FoM* which is established on a relationship that accounts for a requirement to minimise or maximise these quantities respectively. This approach involves identifying the concept solution with the highest useful load, lowest installation drag, and lowest cost solution. Therefore, *FoM* is formulated to provide a ranking score using these requirements and is reflected in Table 78 and Table 80 using the following relationship which is based on normalised scores:

$$FoM = \frac{W_{useful\,load}}{D_{score}+C_{score}} \qquad \text{Equation 46}$$

Where

$W_{useful\ loal}$ – Normalised useful load score of each concept solution.

$D_{score}$ – Normalised installed drag score for each concept solution.

$C_{score}$ – Normalised score EP system cost for each concept solution.

Based on this *FoM* relationship the highest-ranking solution can be determined and compared for over the aircraft segment solution space. The subsequent approach therefore can select the best solution or solutions to in order to conduct a Pugh pairwise comparison analysis as the next step in this methodology.

Table 78. EP system conventional propeller series quantisation – Part 1

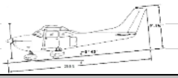
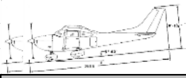
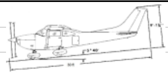
| | | Electric - Conventional propeller series | | |
|---|---|---|---|---|
| **Sub-system function** | Weighting | **Config 1** | **Config 2** | **Config 3** |
| **Energy storage location 1** | | Firewall forward | Firewall forward | Firewall forward |
| **Energy storage location 2** | | None | None | None |
| | |  |  |  |
| **Propeller(s)** | | Conventional | Conventional | Contra-rotating conventional |
| **Motor(s)** | | Siemens AG SP260D | EMRAX 348 | 2x YASA 750 series axial flux |
| **Powerplant weight** | | | | |
| Motor (kg) - Added | | 50.0 | 40.0 | 66.0 |
| Motor mount (est.) (kg) - Added | | 20.0 | 20.0 | 20.0 |
| Powerplant & accessories (kg) - Removed | | -260.7 | -260.7 | -260.7 |
| Total (kg) | | **-190.7** | **-200.7** | **-174.7** |
| **Propeller/spinner & accessories weight** | | | | |
| Propeller(s) (kg) - Added | | 20.0 | 24.0 | 26.0 |
| Propeller(s) (kg) - Removed | | -26.0 | -26.0 | -26.0 |
| Total (kg) | | **-6.1** | **-2.0** | **0.0** |
| **Electric controller/cabling/equipment weight** | | | | |
| Electric controller (kg) - Added | | 15.0 | 15.0 | 15.0 |
| Cabling & recharge receptacle (kg) - Added | | 11.4 | 11.4 | 11.4 |
| Total (kg) | | **26.4** | **26.4** | **26.4** |
| **Battery weight** | | | | |
| Climb power (kWh) - Sizing model | | 35.8 | 50.9 | 48.0 |
| Reserve power (kWh) - 45 mins reserve | | 31.2 | 31.2 | 31.2 |
| Recuperation power - 19% of av climb energy | | 0.0 | 0.0 | 0.0 |
| Total mission power (kWh) | | 67.0 | 82.1 | 79.2 |
| Battery specific energy density (Wh/kg) - Variable | | 250.0 | 250.0 | 250.0 |
| Firewall forward batteries (kg) - Added | | 268.1 | 328.3 | 316.8 |
| Fuselage slipper power pod (kg) - Added | | 0.0 | 0.0 | 0.0 |
| Total batteries (kg) | | **268.1** | **328.3** | **316.8** |
| **Firewall forward weight summary** | | | | |
| Firewall forward weight (kg) - Removed | | -286.7 | -286.7 | -286.7 |
| Firewall forward weight (kg) - Added | | 384.5 | 438.7 | 455.2 |
| *Net firewall forward weight (kg) - Added/Removed* | | **97.8** | **152.0** | **168.5** |
| **Other weight** | | | | |
| Instruments (kg) - Added | | 9.1 | 9.1 | 9.1 |
| Instruments kg - Removed | | -1.6 | -1.6 | -1.6 |
| Cabin accomodations (kg) - Removed | | -23.5 | -23.5 | -23.5 |
| Fuel tank bladders, battery, other (kg) - Removed | | -26.4 | -26.4 | -26.4 |
| Total (kg) | | **-42.4** | **-42.4** | **-42.4** |
| **Net installed weight (kg)** | | | | |
| Total (kg) - Added/Removed | | **55.4** | **109.6** | **126.1** |
| Empty weight (kg) | | 803 | 803 | 803 |
| Adjusted empty weight (kg) | | **858.4** | **912.6** | **929.1** |
| Maximum takeoff weight (kg) | | 1338 | 1338 | 1338 |
| Useful load (kg) | | **479.6** | **425.4** | **408.9** |
| Normalised score - Useful load | 1.20 | **1.20** | **1.06** | **1.02** |
| **Installed drag** | | | | |
| Slipper power pod drag (D/q) (ft^2) - Added | | 0.0000 | 0.0000 | 0.0000 |
| Cooling drag (D/q) (ft^2) - Removed | | -0.0829 | -0.0829 | -0.0829 |
| Net drag | | **-0.0829** | **-0.0829** | **-0.0829** |
| Normalised score - Installed drag | | **1.00** | **1.00** | **1.00** |
| **Propulsion system cost estimate** | | | | |
| Motor power (kW) | | 260 | 300 | 225 |
| Motor(s) ($US) | | $13,000 | $15,000 | $11,250 |
| Propeller(s) ($US) (2 factor for contra-prop) | | $3,250 | $3,750 | $5,625 |
| Batteries ($US) | | $16,758 | $20,519 | $19,803 |
| Controllers ($US) (1.5 factor for contra-prop) | | $26,000 | $30,000 | $33,750 |
| Total cost ($US) | | **$59,008** | **$69,269** | **$70,428** |
| Normalised score - Propulsion system cost | 1.16 | **1.16** | **1.36** | **1.38** |
| **Normalised climb performance** | | | | |
| Useful load (kg) | | **479.6** | **425.4** | **408.9** |
| Time to climb (secs) | | **562** | **1223** | **1154** |
| **Normalised score - Time to climb** | 1.16 | **1.16** | **0.53** | **0.56** |
| **Normalised climb performance score** | | **0.85** | **0.35** | **0.35** |
| **Figure of Merit** | | **0.556** | **0.451** | **0.429** |

Table 79. EP system conventional propeller series quantisation – Part 2

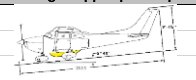
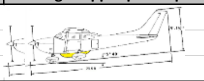
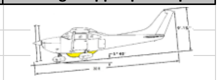
| | | Electric - Conventional propeller series | | |
|---|---|---|---|---|
| **Sub-system function** | Weighting | **Config 4** | **Config 5** | **Config 6** |
| **Energy storage location 1** | | Firewall forward | Firewall forward | Firewall forward |
| **Energy storage location 2** | | Fuselage slipper power pod | Fuselage slipper power pod | Fuselage slipper power pod |
| | |  |  |  |
| **Propeller(s)** | | Conventional | Conventional | Contra-rotating conventional |
| **Motor(s)** | | Siemens AG SP260D | EMRAX 348 | 2X YASA 750 series axial flux |
| **Powerplant weight** | | | | |
| Motor (kg) - Added | | 50.0 | 40.0 | 66.0 |
| Motor mount (est.) (kg) - Added | | 20.0 | 20.0 | 20.0 |
| Powerplant & accessories (kg) - Removed | | -260.7 | -260.7 | -260.7 |
| Total (kg) | | **-190.7** | **-200.7** | **-174.7** |
| **Propeller/spinner & accessories weight** | | | | |
| Propeller(s) (kg) - Added | | 20.0 | 24.0 | 26.0 |
| Propeller(s) (kg) - Removed | | -26.0 | -26.0 | -26.0 |
| Total (kg) | | **-6.1** | **-2.0** | **0.0** |
| **Electric controller/cabling/equipment weight** | | | | |
| Electric controller (kg) - Added | | 15.0 | 15.0 | 15.0 |
| Cabling & recharge receptacle (kg) - Added | | 11.4 | 11.4 | 11.4 |
| Total (kg) | | **26.4** | **26.4** | **26.4** |
| **Battery weight** | | | | |
| Climb power (kWh) - Sizing model | | **36.2** | **51.9** | **49.0** |
| Reserve power (kWh) - 45 mins reserve | | 31.2 | 31.2 | 31.2 |
| Recuperation power - 19% of av climb energy | | 0.0 | 0.0 | 0.0 |
| Total mission power (kWh) | | 67.4 | 83.1 | 80.2 |
| Battery specific energy density (Wh/kg) - Variable | | 250.0 | 250.0 | 250.0 |
| Firewall forward batteries (kg) - Added | | 170.4 | 176.3 | 148.3 |
| Fuselage slipper power pod (kg) - Added | | 99.1 | 156.2 | 172.3 |
| Total batteries (kg) | | **269.5** | **332.5** | **320.6** |
| **Firewall forward weight summary** | | | | |
| Firewall forward weight (kg) - Removed | | -286.7 | -286.7 | -286.7 |
| Firewall forward weight (kg) - Added | | 286.7 | 286.7 | 286.7 |
| *Net firewall forward weight (kg) - Added/Removed* | | 0.0 | 0.0 | 0.0 |
| **Other weight** | | | | |
| Instruments (kg) - Added | | 9.1 | 9.1 | 9.1 |
| Instruments kg - Removed | | -1.6 | -1.6 | -1.6 |
| Cabin accomodations (kg) - Removed | | -23.5 | -23.5 | -23.5 |
| Fuel tank bladders, battery, other (kg) - Removed | | -26.4 | -26.4 | -26.4 |
| Total (kg) | | **-42.4** | **-42.4** | **-42.4** |
| **Net installed weight (kg)** | | | | |
| Total (kg) - Added/Removed | | **56.7** | **113.8** | **129.9** |
| Empty weight (kg) | | 803 | 803 | 803 |
| Adjusted empty weight (kg) | | **859.7** | **916.8** | **932.9** |
| Maximum takeoff weight (kg) | | 1338 | 1338 | 1338 |
| Useful load (kg) | | **478.3** | **421.2** | **405.1** |
| Normalised score - Useful load | 1.20 | **1.20** | **1.05** | **1.01** |
| **Installed drag** | | | | |
| Slipper power pod drag (D/q) (ft^2) - Added | | 0.2676 | 0.2676 | 0.2676 |
| Cooling drag (D/q) (ft^2) - Removed | | -0.0829 | -0.0829 | -0.0829 |
| Net drag | | **0.1847** | **0.1847** | **0.1847** |
| Normalised score - Installed drag | | **2.23** | **2.23** | **2.23** |
| **Propulsion system cost estimate** | | | | |
| Motor power (kW) | | 260 | 300 | 225 |
| Motor(s) ($US) | | $13,000 | $15,000 | $11,250 |
| Propeller(s) ($US) (2 factor for contra-prop) | | $3,250 | $3,750 | $5,625 |
| Batteries ($US) | | $16,758 | $20,519 | $19,803 |
| Controllers ($US) (1.5 factor for contra-prop) | | $26,000 | $30,000 | $33,750 |
| Total cost ($US) | | **$59,008** | **$69,269** | **$70,428** |
| Normalised score - Propulsion system cost | 1.16 | **1.16** | **1.36** | **1.38** |
| **Normalised climb performance** | | | | |
| Useful load (kg) | | **478.3** | **421.2** | **405.1** |
| Time to climb (secs) | | **567** | **1248** | **1176** |
| **Normalised score - Time to climb** | 1.16 | **1.15** | **0.52** | **0.55** |
| **Normalised climb performance score** | | **0.84** | **0.34** | **0.34** |
| **Figure of Merit** | | **0.353** | **0.294** | **0.281** |

Table 80. EP system recuperative propeller series quantisation – Part 1

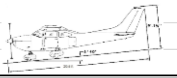
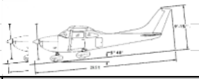
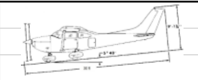
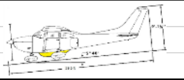
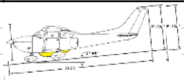
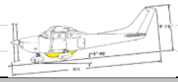
| | | Electric - Recuperative propeller series | | |
|---|---|---|---|---|
| **Sub-system function** | **Weighting** | **Config 7** | **Config 8** | **Config 9** |
| **Energy storage location 1** | | Firewall forward | Firewall forward | Firewall forward |
| **Energy storage location 2** | | None | None | None |
| | |  |  |  |
| **Propeller(s)** | | Recuperative | Recuperative | Contra-rotating recuperative |
| **Motor(s)** | | Siemens AG SP260D | EMRAX 348 | 2x YASA 750 series axial flux |
| **Powerplant weight** | | | | |
| Motor (kg) - Added | | 50.0 | 40.0 | 66.0 |
| Motor mount (est.) (kg) - Added | | 20.0 | 20.0 | 20.0 |
| Powerplant & accessories (kg) - Removed | | -260.7 | -260.7 | -260.7 |
| Total (kg) | | **-190.7** | **-200.7** | **-174.7** |
| **Propeller/spinner & accessories weight** | | | | |
| Propeller(s) (kg) - Added | | 20.0 | 24.0 | 26.0 |
| Propeller(s) (kg) - Removed | | -26.0 | -26.0 | -26.0 |
| Total (kg) | | **-6.1** | **-2.0** | **0.0** |
| **Electric controller/cabling/equipment weight** | | | | |
| Electric controller (kg) - Added | | 15.0 | 15.0 | 15.0 |
| Cabling & recharge receptacle (kg) - Added | | 11.4 | 11.4 | 11.4 |
| Total (kg) | | **26.4** | **26.4** | **26.4** |
| **Battery weight** | | | | |
| Climb power (kWh) - Sizing model | | 35.8 | 50.9 | 48.0 |
| Reserve power (kWh) - 45 mins reserve | | 31.2 | 31.2 | 31.2 |
| Recuperation power - 19% of av climb energy | | -6.8 | -9.7 | -9.1 |
| Total mission power (kWh) | | 60.2 | 72.4 | 70.1 |
| Battery specific energy density (Wh/kg) - Variable | | 250.0 | 250.0 | 250.0 |
| Firewall forward batteries (kg) - Added | | 240.9 | 289.6 | 280.4 |
| Fuselage slipper power pod (kg) - Added | | 0.0 | 0.0 | 0.0 |
| Total (kg) | | **240.9** | **289.6** | **280.4** |
| **Firewall forward weight summary** | | | | |
| Firewall forward weight (kg) - Removed | | -286.7 | -286.7 | -286.7 |
| Firewall forward weight (kg) - Added | | 357.3 | 400.0 | 418.8 |
| *Net firewall forward weight (kg) - Added/Removed* | | **70.5** | **113.3** | **132.0** |
| **Other weight** | | | | |
| Instruments (kg) - Added | | 9.1 | 9.1 | 9.1 |
| Instruments kg - Removed | | -1.6 | -1.6 | -1.6 |
| Cabin accomodations (kg) - Removed | | -23.5 | -23.5 | -23.5 |
| Fuel tank bladders, battery, other (kg) - Removed | | -26.4 | -26.4 | -26.4 |
| Total (kg) | | **-42.4** | **-42.4** | **-42.4** |
| **Net installed weight (kg)** | | | | |
| Total (kg) - Added/Removed | | **28.1** | **70.9** | **89.6** |
| Empty weight (kg) | | 803 | 803 | 803 |
| Adjusted empty weight (kg) | | **831.1** | **873.9** | **892.6** |
| Maximum takeoff weight (kg) | | 1338 | 1338 | 1338 |
| Useful load (kg) | | **506.9** | **464.1** | **445.4** |
| Normalised score - Useful load | 1.20 | **1.27** | **1.16** | **1.11** |
| **Installed drag** | | | | |
| Slipper power pod drag (D/q) (ft^2) - Added | | 0.0000 | 0.0000 | 0.0000 |
| Cooling drag (D/q) (ft^2) - Removed | | -0.0829 | -0.0829 | -0.0829 |
| Net drag | | **-0.0829** | **-0.0829** | **-0.0829** |
| Normalised score - Installed drag | | **1.00** | **1.00** | **1.00** |
| **Propulsion system cost estimate** | | | | |
| Motor power (kW) | | 260 | 300 | 225 |
| Motor(s) ($US) | | $13,000 | $15,000 | $11,250 |
| Propeller(s) ($US) (2 factor for contra-prop) | | $3,250 | $3,750 | $5,625 |
| Batteries ($US) | | $15,056 | $18,102 | $17,522 |
| Controllers ($US) (1.5 factor for contra-prop) | | $26,000 | $30,000 | $33,750 |
| Total cost ($US) | | **$57,306** | **$66,852** | **$68,147** |
| Normalised score - Propulsion system cost | 1.16 | **1.13** | **1.31** | **1.34** |
| **Normalised climb performance** | | | | |
| Useful load (kg) | | **506.9** | **464.1** | **445.4** |
| Time to climb (secs) | | **562** | **1223** | **1154** |
| **Normalised score - Time to climb** | 1.16 | **1.16** | **0.53** | **0.56** |
| **Normalised climb performance score** | | **0.90** | **0.38** | **0.39** |
| **Figure of Merit** | | **0.596** | **0.502** | **0.476** |

Table 81. EP system recuperative propeller series quantisation – Part 2

| | | Electric - Recuperative propeller series | | |
|---|---|---|---|---|
| **Sub-system function** | Weighting | **Config 10** | **Config 11** | **Config 12** |
| **Energy storage location 1** | | Firewall forward | Firewall forward | Firewall forward |
| **Energy storage location 2** | | Fuselage slipper power pod | Fuselage slipper power pod | Fuselage slipper power pod |
| | |  |  |  |
| **Propeller(s)** | | Recuperative | Recuperative | Contra-rotating recuperative |
| **Motor(s)** | | Siemens AG SP260D | EMRAX 348 | 2X YASA 750 series axial flux |
| **Powerplant weight** | | | | |
| Motor (kg) - Added | | 50.0 | 40.0 | 66.0 |
| Motor mount (est.) (kg) - Added | | 20.0 | 20.0 | 20.0 |
| Powerplant & accessories (kg) - Removed | | -260.7 | -260.7 | -260.7 |
| Total (kg) | | **-190.7** | **-200.7** | **-174.7** |
| **Propeller/spinner & accessories weight** | | | | |
| Propeller(s) (kg) - Added | | 20.0 | 24.0 | 26.0 |
| Propeller(s) (kg) - Removed | | -26.0 | -26.0 | -26.0 |
| Total (kg) | | **-6.1** | **-2.0** | **0.0** |
| **Electric controller/cabling/equipment weight** | | | | |
| Electric controller (kg) - Added | | 15.0 | 15.0 | 15.0 |
| Cabling & recharge receptacle (kg) - Added | | 11.4 | 11.4 | 11.4 |
| Total (kg) | | **26.4** | **26.4** | **26.4** |
| **Battery weight** | | | | |
| Climb power (kWh) - Sizing model | | **36.2** | 51.9 | 49.0 |
| Reserve power (kWh) - 45 mins reserve | | 31.2 | 31.2 | 31.2 |
| Recuperation power - 19% of av climb energy | | -6.9 | -9.9 | -9.3 |
| Total mission power (kWh) | | 60.5 | 73.3 | 70.9 |
| Battery specific energy density (Wh/kg) - Variable | | 250.0 | 250.0 | 250.0 |
| Firewall forward batteries (kg) - Added | | 170.4 | 176.3 | 148.3 |
| Fuselage slipper power pod (kg) - Added | | 71.7 | 116.8 | 135.1 |
| Total (kg) | | **242.0** | **293.1** | **283.4** |
| **Firewall forward weight summary** | | | | |
| Firewall forward weight (kg) - Removed | | -286.7 | -286.7 | -286.7 |
| Firewall forward weight (kg) - Added | | 286.7 | 286.7 | 286.7 |
| *Net firewall forward weight (kg) - Added/Removed* | | 0.0 | 0.0 | 0.0 |
| **Other weight** | | | | |
| Instruments (kg) - Added | | 9.1 | 9.1 | 9.1 |
| Instruments kg - Removed | | -1.6 | -1.6 | -1.6 |
| Cabin accomodations (kg) - Removed | | -23.5 | -23.5 | -23.5 |
| Fuel tank bladders, battery, other (kg) - Removed | | -26.4 | -26.4 | -26.4 |
| Total (kg) | | **-42.4** | **-42.4** | **-42.4** |
| **Net installed weight (kg)** | | | | |
| Total (kg) - Added/Removed | | **29.2** | **74.3** | **92.7** |
| Empty weight (kg) | | 803 | 803 | 803 |
| Adjusted empty weight (kg) | | **832.2** | **877.3** | **895.7** |
| Maximum takeoff weight (kg) | | 1338 | 1338 | 1338 |
| Useful load (kg) | | **505.8** | **460.7** | **442.3** |
| Normalised score - Useful load | 1.20 | **1.27** | **1.15** | **1.11** |
| **Installed drag** | | | | |
| Slipper power pod drag (D/q) (ft^2) - Added | | 0.2676 | 0.2676 | 0.2676 |
| Cooling drag (D/q) (ft^2) - Removed | | -0.0829 | -0.0829 | -0.0829 |
| Net drag | | **0.1847** | **0.1847** | **0.1847** |
| Normalised score - Installed drag | | **2.23** | **2.23** | **2.23** |
| **Propulsion system cost estimate** | | | | |
| Motor power (kW) | | 260 | 300 | 225 |
| Motor(s) ($US) | | $13,000 | $15,000 | $11,250 |
| Propeller(s) ($US) (2 factor for contra-prop) | | $3,250 | $3,750 | $5,625 |
| Batteries ($US) | | $15,056 | $18,102 | $17,522 |
| Controllers ($US) (1.5 factor for contra-prop) | | $26,000 | $30,000 | $33,750 |
| Total cost ($US) | | **$57,306** | **$66,852** | **$68,147** |
| Normalised score - Propulsion system cost | 1.16 | **1.13** | **1.31** | **1.34** |
| **Normalised climb performance** | | | | |
| Useful load (kg) | | **505.8** | **460.7** | **442.3** |
| Time to climb (secs) | | **567** | **1248** | **1176** |
| **Normalised score - Time to climb** | 1.16 | **1.15** | **0.52** | **0.55** |
| **Normalised climb performance score** | | **0.89** | **0.37** | **0.38** |
| **Figure of Merit** | | **0.377** | **0.325** | **0.310** |

### 3.3.4 Quantisation of the ground charging morphological matrix

Table 82 shows the quantified morphological matrix relating to the ground changing station solution space. It is presented in a similar format as described earlier in this Appendix. The upper rows of this matrix provide configuration details of each concept which are determined from the corresponding ground charging infrastructure morphological matrix as described earlier. For example, the upper rows present energy supply type, charging mode and energy transfer/charge option for each configuration concept.

Table 82 incorporates infrastructure costs, recharge time and flight duty cycle metrics to approximate the main attributes of the ground charging infrastructure segment. These quantities are either directly or indirectly derived from the corresponding ground infrastructure QFD matrix shown in Table 59.

This quantisation analysis is based on Electric Vehicle (EV) infrastructure data described by Kettles (2015) and Smith & Castellano (2015). Kettles (2015) describes the types and standards associated with electric vehicle charging stations, with these stations are classified by Level, as follows:

- Level 1 is a domestic socket type recharging station with a protection device usually built into the cable. The vehicle is connected to the main power grid via a household socket-outlet, with charging achieved using a single-phase or three-phase network.
- Level 2 is a specific socket on a dedicated circuit, where the vehicle is connected directly to the electrical network. A control and protection function may be installed permanently in this installation. The charging schedule may be used to optimise the electric vehicle charging time and can also allow load shedding so that other electrical equipment can be operated during charging.
- DCFC relies on a Direct current (DC) connection for fast recharging. The control and protection functions are permanently installed. Typical charging times are less than 30 minutes for fast charging, or less than 10 minutes for ultra-high charging (Dsdmip.qld.gov.au, 2018).

#### *3.3.4.1 Quantisation and normalisation of metrics*

As described earlier, quantisation is based on the metrics as shown in the left most column of this matrix shown in Table 82. These metrics are grouped into blocks which present the data relevant to each metric given by costs and the related performance metrics relating to recharge time and flight duty cycle. The final score relating to each metric, is shown in **<u>bold underline</u>**, and is normalised to the configuration baseline, or as a direct score.

#### *3.3.4.2 Metrics weighting*

Each metric is weighted to account for importance in accordance with those weightings derived in the QFD matrix shown by Table 59. These weightings are shown on the left-hand side of the matrix and have been assigned to cost and recharge time metrics.

#### *3.3.4.3 Costs*

The cost metrics associated with the ground charging station can be divided into two broad areas. These areas relate to the ground charging system equipment installation costs, and the other associated with the costs of additional battery units used for exchange with the system installed on the aircraft. Given that the objective function is to minimise costs, then the number of additional battery sets has been minimised. However, it is noted that some ground charging systems do not provide adequate recharge times for this benefit to be realised. This is discussed further below.

The average ground charging system equipment and installation costs per unit are determined from Smith & Castellano (2015). Additional battery sets are determined from the aircraft quantisation matrix based on an average installation cost. This is determined to be $US 15000 per set. Battery set quantity are determined in accordance with a cost minimisation objective function as discussed above. However, the recharging time associated with some ground charging stations do not provide significant benefits without additional charging battery sets and associated additional costs. The boundary condition constraining additional battery sets is flight duty cycle, where the number of additional flights achieved is limited by the pre-charged batteries installed in the aircraft and the number of spare battery sets. In the case of Configurations 4 and 5, two additional battery sets can only achieve 3 flights per day, which is two more than that achieved for a Configurations 1 and 2 (no spare battery sets). Additional battery sets to achieve an acceptable number of flights per day obviously will add to overall system cost, making this operating concept untenable.

#### *3.3.4.4 Recharge time*

The maximum battery recharge time metric is determined from Kettles (2015) for the corresponding charging station levels as noted. It should be noted that the maximum charging time assumes an average recharge on an EV. Kettles (2015) states the actual recharging time is subject to a combination of factors such State-of-Charge (SOC), Charging Level and battery size as described below.

- The State-Of-Charge (SOC) is a measure of the battery charge level at the beginning of the charging cycle. A battery with a low SOC will take longer to recharge, regardless of the level of charging applied.
- Charging level is a measure of the power that is supplied to a battery during recharging, with the most common measurement being maximum current. Various levels of charging are available. Kettles (2015) states that Level 1 provides up to 15 amps of alternating current, Level 2 provides up to 40 amps of alternating current, and DC Fast Charge provides up to 125 amps of direct current.
- Battery size requirements are proportional to the physical size and weight of the vehicle and the desired range of travel. Accordingly, larger batteries take longer to recharge, unless faster charge rates are used.

#### *3.3.4.5 Flight duty cycle*

The flight duty cycle is the number of flights that can be achieved during an assumed 8-hour day. It is determined from the recharge time for each system charging level and the spare battery sets available for exchange after each flight. It is assumed that the day starts with a charged battery set in the aircraft which can be exchanged at the conclusion of the first flight. Battery sets are exchanged accordingly but were in some cases limited by the recharge rate performance of some charge stations.

#### *3.3.4.6 Figure of Merit*

As described earlier for the aircraft segment, metrics are quantised by use of simple mathematical models allowing an approximation of the ground charging station. These metrics can then be evaluated by a *FoM* which is established on a relationship that accounts for a requirement to minimise costs and minimise recharge time and maximise flight duty cycles.

This *FoM* is formulated to provide a ranking score using these requirements and is reflected in Table 82 as follows:

$$FoM = \frac{Flight\ duty\ cycle}{T_{score}+C_{score}} \qquad \text{Equation 47}$$

Where the following definitions apply:

$T_{score}$ – Normalised recharge time score.

$C_{score}$ – Normalised cost score.

*Flight duty cycle* – This is a normalised score based on one achievable flight for Configuration 1.

Based on this *FoM* relationship the highest-ranking solution can be determined and compared for the ground infrastructure segment solution space.

### 3.3.5 Ground infrastructure segment results

The lower row in Table 82 shows the *FoM* scores determined for all potential ground charging station solutions. On the basis of recharge time, cost and flight duty cycle metrics, the battery exchange DC Fast Charging Concept (Configuration 6) providing 12 flights per day using an additional battery set ranked highest. This was followed by the similar DC Fast Charging Concept (Configuration 3) providing 8 flights per day using the in-aircraft charging. What is not considered is the time and effort associated with exchanging approximately 250 kg of battery at the conclusion of each flight. This would be an onerous and arduous activity that would require several persons to complete in an acceptable length of time. Furthermore, the battery exchange concept would also introduce the possibility of damage to batteries or the aircraft. For this reason, battery exchange on module by module basis may not be practical for the weight of batteries considered here. Therefore, development of automated ground handling equipment to exchange batteries as a complete set may be required. Correspondence with Glassock (2018, pers. comm., 8 October) has indicated that an automated battery exchange system that could remove and replace the entire battery unit may be a viable solution.

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Table 82. Electric Propulsion ground charging station quantisation

| | | Ground charging system | | | | | |
|---|---|---|---|---|---|---|---|
| **Sub-system function** | **Weighting** | **Config 1** | **Config 2** | **Config 3** | **Config 4** | **Config 5** | **Config 6** |
| Energy supply type | | Mains supply | Mains supply | Mains supply | Mains supply | Mains supply | Mains supply |
| Charging mode | | AC Level 1 | AC Level 2 | DC Fast Charging (DCFC) | AC Level 1 | AC Level 2 | DC Fast Charging (DCFC) |
| Energy transfer/charge | | **In aircraft charging** | **In aircraft charging** | **In aircraft charging** | **Battery exchange** | **Battery exchange** | **Battery exchange** |
| **Description** | | Basic AC charging using OEM charger 1 phase | Basic AC charging - Hardwired charger 1 or 3 Phase | Dedicated DC fast charge station | Basic AC charging using OEM charger 1 phase | Basic AC charging - Hardwired charger 1 or 3 Phase | Dedicated DC fast charge station |
| **Costs** | | | | | | | |
| Ground charging system - per unit average ($US) | | $1,000 | $3,500 | $25,000 | $1,000 | $3,500 | $25,000 |
| Installation average per unit ($US) | | $1,500 | $6,700 | $27,500 | $1,500 | $6,700 | $27,500 |
| Additional battery sets (#) | | 0 | 0 | 0 | 2 | 2 | 1 |
| Battery set cost ($US) - $15000 /set | | $0 | $0 | $0 | $30,000 | $30,000 | $15,000 |
| Total | | $2,500 | $10,200 | $52,500 | $32,500 | $40,200 | $67,500 |
| Normalised score - Costs | 1.2 | **1.19** | **0.29** | **0.06** | **0.09** | **0.07** | **0.04** |
| **Recharge time** | | | | | | | |
| Max battery recharge time (mins) | | 1200 | 420 | 30 | 1200 | 420 | 30 |
| Normalised score - Recharge time | 1.2 | **1.20** | **0.42** | **0.03** | **1.20** | **0.42** | **0.03** |
| **Flight duty cycle** | | | | | | | |
| No of flights/day based on recharge time and/or exchange | | **1** | **1** | **8** | **3** | **3** | **12** |
| | | | | | | | |
| **Figure of Merit** | | **0.4** | **1.4** | **92.3** | **2.3** | **6.1** | **162.0** |

## 3.4 CONCEPT SELECTION VALIDATION – STEP 3

### 3.4.1 Overview

The next step in this methodology involves the application of the Pugh Matrix to validate the candidate concept solution as determined from the previous morphological matrix analysis. However, in this case study there is only one solution concept that meets climb and useful load requirements, with the remaining solution concepts, providing no significant advantage over existing IC engine climb performance. Therefore, it is not necessary to apply the Pugh Matrix to validate this concept solution, given that there is only one satisfactory candidate.

### 3.4.2 Candidate design concept

The solution concept that meets climb and useful load required is given by **Configuration 10** (*FoM* = 0.377), which comprises a firewall forward battery installation with balance of batteries installed in Power slipper pod. The Siemens AG SP260D motor is used, and is fitted with a recuperative propeller. It is important to note that Configuration 10 was selected over Configuration 7, due to configuration weight and balance impacts. In the case of Configuration 7, the battery weight added forward of the firewall exceeded the weight removed from this compartment. This results in a significant weight imbalance that must be corrected by adding these batteries to the Slipper pod. This in effect becomes Configuration 10, which is the preferred candidate solution. The corresponding Configurations 1 and 4 were discounted as they did not provide the small efficiency and weight benefits provided by the recuperative propeller.

### 3.4.3 Effect of motor continuous power vs maximum/peak power

It should be noted that this case study also investigated the effect of selecting motor maximum/peak power as the basis of climb performance prediction (Williams, 2018b). The selection of the motor maximum/peak power parameter in contrast to continuous power, changed the preferred candidate selection. In this particular case the preferred solution was the Emrax 268 motor-based solution (*FoM* = 0.403), as this motor possessed high specific peak power (i.e. possessed very low weight) in comparison to the other motors. This was closely followed by the Siemens AG SP260D motor (*FoM* = 0.377) and the Emrax 348 motor (*FoM* = 0.371). Therefore, application of the quantified morphological matrix has emphasised differences in motor power characteristics, and how this impacts the selection of a preferred candidate design concept. This outcome also aligns with motor data presented in Table 68, and provides substantiation that validates this QMM approach.

As described earlier, the low continuous power output of the Emrax 268 motor exhibited much lower climb performance, and for this reason, the Emrax 268-based configuration was omitted from the subsequent morphological analysis.

## 3.5 ASSESSMENT OF PROPAGATION EFFECTS AND CHANGE OPTIONS – STEP 4

### 3.5.1 Overview

The assessment of engineering change propagation effects is applied as an intermediate step of this conceptual design methodology. As discussed in Chapter 4 and Chapter 5 the conduct of this step provides inputs to the engineering assessment step. In reality, these propagation effect design synthesis activities would be conducted as a parallel iterative process.

The underpinning theory associated with this method is described by Koh et al. (2012) with this method building on the QFD matrix technique. The QFD matrix is an early step as described in this conceptual design methodology, and precedes the more detailed change propagation analysis as outlined in the next section.

### 3.5.2 Selection of best change option

The revised performance ratings applicable to this EP system modification are determined by a four-step process as described in Chapter 5 and also implemented in Appendix 1. Again, the Fields denoted by A to E in Figure 42 are developed to determine the change dependencies shown in the resultant Multiple-Domain Matrix (MDM) at Table 83. Full details of the underpinning analysis methods associated with this change option approach is provided in Chapter 5.

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Table 83. EP system change dependencies MDM

| | Electric motor(s) | Propeller(s) | Batteries - Firewall forward | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | Decrease mod installation drag | Decrease mod installation weight | Reduce propulsion system related LCC | Maintain power output | Maintain aircraft payload volume and space capacity | The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load. | TThe modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | The modification lifecycle costs (LCC) shall be kept to a minimum | The modification shall comply with airworthiness standards and minimise impact to the TC basis. | The modification shall minimise emissions. | The modification shall be compatible with a range of single engine light aircraft types. | The modification shall not impact passenger, baggage or cargo space. |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Electric motor(s) | 1 | 0.2 | 0.4 | 0.4 | 0 | 0.8 | 0 | 0 | 0 | 0 | 0 | 1 | 1 | | 1 | | | | | | | | |
| Propeller(s) | 0.8 | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 1 | 1 | | 1 | | | | | | | | |
| Batteries - Firewall forward | 0.4 | 0.4 | 1 | 0.2 | 0 | 0 | 0 | 0 | 0.2 | 0.2 | 0.2 | | 1 | | 1 | | | | | | | | |
| Batteries - Slipper power pod | 0.4 | 0.4 | 0.2 | 1 | 0 | 0 | 0 | 0 | 0.2 | 0.2 | 0.2 | 1 | 1 | | 1 | 1 | | | | | | | |
| Electric controller - Throttle | 0 | 0 | 0 | 0 | 1 | 0.5 | 0 | 0 | 0 | 0 | 0 | | 1 | | 1 | | | | | | | | |
| Speed ctrl & motor cooling | 0.8 | 0.2 | 0.4 | 0.4 | 0.4 | 1 | 0 | 0.2 | 0 | 0 | 0 | | 1 | | 1 | | | | | | | | |
| Circuit breakers | 0.1 | 0 | 0.1 | 0.1 | 0 | 0.1 | 1 | 0.1 | 0.1 | 0.1 | 0.1 | | 1 | | | | | | | | | | |
| Master switch | 0 | 0 | 0 | 0 | 0 | 0 | 0.1 | 1 | 0 | 0 | 0 | | 1 | | | | | | | | | | |
| State of Charge (SOC) instrumentation | 0 | 0 | 0.5 | 0.5 | 0 | 0 | 0.1 | 0 | 1 | 0.4 | 0 | | 1 | | | | | | | | | | |
| State of Health (SOH) Instrumentation | 0 | 0 | 0.5 | 0.5 | 0 | 0 | 0.1 | 0 | 0.4 | 1 | 0 | | 1 | | | | | | | | | | |
| Recharge receptacle/exchange | 0 | 0 | 0.2 | 0.2 | 0 | 0 | 0 | 0 | 0 | 0 | 1 | | 1 | | | | | | | | | | |
| Decrease mod installation drag | 1 | 1 | 0 | 0 | 0 | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0.5 | -0.4 | 0 | -0.2 | | | | | | | |
| Decrease mod installation weight | 1 | 1 | 1 | 1 | 0 | 1 | 0 | 0 | 1 | 1 | 0 | 0.3 | 0 | -0.4 | 0 | -0.2 | | | | | | | |
| Reduce propulsion system related LCC | 1 | 1 | 1 | 1 | 0 | 1 | 0 | 0 | 1 | 1 | 0 | -0.4 | 0.1 | 0 | -0.4 | -0.2 | | | | | | | |
| Maintain power output | 1 | 1 | 1 | 1 | 0 | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | -0.6 | 0 | 0 | | | | | | | |
| Maintain aircraft payload volume and space capacity | 0 | 0 | 0 | 0 | 1 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0.2 | 0 | 0 | | | | | | | |
| The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load.. | | | | | | | | | | | | 4 | 4 | 4 | 5 | 3 | | | | | | | |
| TThe modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | | | | | | | | | | | | 4 | 4 | 4 | 5 | 3 | | | | | | | |
| The modification lifecycle costs (LCC) shall be kept to a minimum | | | | | | | | | | | | 3 | 3 | 5 | 3 | 3 | | | | | | | |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | | | | | | | | | | | | 1 | 1 | 2 | 0 | 0 | | | | | | | |
| The modification shall minimise emissions. | | | | | | | | | | | | 3 | 2 | 0 | 1 | 1 | | | | | | | |
| The modification shall be compatible with a range of single engine light aircraft types. | | | | | | | | | | | | 1 | 1 | 3 | 1 | 1 | | | | | | | |
| The modification shall not impact passenger, baggage or cargo space. | | | | | | | | | | | | 1 | 1 | 2 | 1 | 5 | | | | | | | |

### 3.5.3 Results

Based on the revised performance ratings generated in the previous section, the different change options can be evaluated in terms of their expected attributes with the best change option selected for further development. As stated by Koh et al. (2012) there are different ways to undertake the assessment. One approach is to select the change option that has the best performance rating against a given modification requirement. Table 84 through Table 87 are provided as summaries of an analysis of Field A (refer Figure 42) of the MDM as shown in Table 83.

For example, it is apparent from Table 84 that the two requirements dealing with 'shall provide a time of climb….' and 'shall provide a useful load' are important to the success of the modification. This corresponds to the change option 'Decrease installation weight', which in this case, both possess values of '4.5'. Therefore, the change option 'Decrease installation weight' would be an ideal or best solution as it has the highest performance ratings of '4.5'.

An alternative approach is to select the change option with the best overall attributes. This can be carried out by comparing the sum of each column. For instance, the second change option comprising 'Decrease installation weight' has the highest total performance rating of '18.6', suggesting that it has the best overall product attributes. Hence, from an overall perspective, the second change option is the best solution for further development closely followed by the first being to 'Decrease installation drag' with total performance rating of '18.0'. Therefore, both of these change options comprising weight reduction and drag reduction are closely matched. This result aligns with climb performance theory where highest climb rate is achieved with the lowest weight and drag for a constant thrust.

Table 84. Dependencies without weighting – EP System Mod

| **Field A* without weighting** | Decrease mod installation drag | Decrease mod installation weight | Reduce propulsion system related LCC | Maintain power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load.. | 4.4 | 4.5 | 2.9 | 5.0 | 3.0 |
| The modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | 4.4 | 4.5 | 2.9 | 4.7 | 3.0 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 3.0 | 3.8 | 4.2 | 2.6 | 3.0 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 1.0 | 1.2 | 1.9 | -0.1 | 0.0 |
| The modification shall minimise emissions. | 3.5 | 2.3 | -0.5 | 1.0 | 1.0 |
| The modification shall be compatible with a range of single engine light aircraft types. | 0.8 | 1.2 | 2.7 | 0.8 | 1.0 |
| The modification shall not impact passenger, baggage or cargo space. | 1.0 | 1.2 | 1.7 | 0.9 | 5.0 |
| | 18.0 | 18.6 | 15.8 | 14.8 | 16.0 |

Another approach is to assign weightings to each row so as to better reflect the importance of each product requirement as shown in Table 85. This is analogous to the use of weighted score during concept scoring as described by Gudmundsson (2014) as applied in the QFD matrix shown by Table 57.

For example, given that the requirement to ‘shall provide a useful load ….’ is important, a higher weighting is assigned to emphasise its importance. In this case, the significance is deemed as 20% of the overall product requirements, and hence the performance ratings can be adjusted as shown. Again, the change option with the best overall attributes is, ‘Decrease installation weight’, with a score of ‘2.8’, followed by ‘Decrease installation drag’ with a score of ‘2.7’. Again, both attributes are closely matched.

Table 85. Dependencies with weighting – EP System Mod

| **Field A* with weighting** | Decrease mod installation drag | Decrease mod installation weight | Reduce propulsion system related LCC | Maintain power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load.. | 0.7 | 0.7 | 0.5 | 0.8 | 0.5 |
| TThe modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | 0.9 | 0.9 | 0.6 | 0.9 | 0.6 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 0.5 | 0.6 | 0.7 | 0.4 | 0.5 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | 0.1 | 0.1 | 0.2 | 0.0 | 0.0 |
| The modification shall minimise emissions. | 0.4 | 0.3 | -0.1 | 0.1 | 0.1 |
| The modification shall be compatible with a range of single engine light aircraft types. | 0.1 | 0.1 | 0.3 | 0.1 | 0.1 |
| The modification shall not impact passenger, baggage or cargo space. | 0.1 | 0.1 | 0.2 | 0.1 | 0.6 |
| | 2.7 | 2.8 | 2.2 | 2.4 | 1.8 |

Another approach is to compare the simple difference between Field A and Field A* to get a simple summary of the change propagation effects as shown by Table 86. In this case, a positive difference implies that the relevant attributes are affected in a favourable way (e.g. ‘Decrease installation drag’ or ‘Decrease installation weight’ on ‘shall provide a time of climb….’) while a negative difference suggests opposite (e.g. ‘Reduce propulsion system LCC’ on ‘shall provide a time of climb….’).

Table 86. Difference between Field A and A* - EP System Mod

| **Simple difference between Field A and A*** | Decrease mod installation drag | Decrease mod installation weight | Reduce propulsion system related LCC | Maintain power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load.. | 0.4 | 0.5 | -1.1 | 0.0 | 0.0 |
| TThe modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | 0.4 | 0.5 | -1.1 | -0.3 | 0.0 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 0.0 | 0.8 | -0.8 | -0.4 | 0.0 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis | 0.0 | 0.2 | -0.1 | -0.1 | 0.0 |
| The modification shall minimise emissions. | 0.5 | 0.3 | -0.5 | 0.0 | 0.0 |
| The modification shall be compatible with a range of single engine light aircraft types. | -0.2 | 0.2 | -0.3 | -0.2 | 0.0 |
| The modification shall not impact passenger, baggage or cargo space. | 0.0 | 0.2 | -0.3 | -0.1 | 0.0 |
| | 1.0 | 2.6 | -4.2 | -1.2 | 0.0 |

Change options that are more sensitive to change propagation can be highlighted as well by calculating the absolute difference between Field A and A* as shown by Table 87. For example, change option 'Decrease installation weight' is the most change sensitive option for requirement 'shall provide a time of climb….' as it has the greatest absolute difference along the first row of the matrix (absolute difference of '0.5'). Conversely, the change option 'Decrease propulsion system LCC' is the most change sensitive option overall as it has the greatest absolute difference in total (assuming no weightings were assigned to the ratings).

It should be noted that the analysis seeks to support the design team by highlighting change propagation effects in a structured manner. Koh et al. (2012) states that, even though change propagation can sometimes result in better product attributes, unplanned change propagation can be risky and detrimental to the success of a design modification. Hence, the implementation of change options should always be properly managed even if the assessment suggests a favourable potential outcome.

Table 87. Absolute difference between Field A and A* - EP System Mod

| **Absolute difference between Field A and A*** | Decrease mod installation drag | Decrease mod installation weight | Reduce propulsion system related LCC | Maintain power output | Maintain aircraft payload volume and space capacity |
|---|---|---|---|---|---|
| The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load.. | 0.4 | 0.5 | 1.1 | 0.0 | 0.0 |
| TThe modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | 0.4 | 0.5 | 1.1 | 0.3 | 0.0 |
| The modification lifecycle costs (LCC) shall be kept to a minimum | 0.0 | 0.8 | 0.8 | 0.4 | 0.0 |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis | 0.0 | 0.2 | 0.1 | 0.1 | 0.0 |
| The modification shall minimise emissions. | 0.5 | 0.3 | 0.5 | 0.0 | 0.0 |
| The modification shall be compatible with a range of single engine light aircraft types. | 0.2 | 0.2 | 0.3 | 0.2 | 0.0 |
| The modification shall not impact passenger, baggage or cargo space. | 0.0 | 0.2 | 0.3 | 0.1 | 0.0 |
| | 1.6 | 2.6 | 4.2 | 1.2 | 0.0 |

## 3.6 EVALUATION OF DESIGN CHANGE IMPACTS – STEP 5

### 3.6.1 Overview

A project that modifies or retrofits an aircraft with a new subsystem or component will invoke changes to that segment. Indeed, the integration of a propulsion system modification such as given in this EP system case study will also impact external interfaces such as ground infrastructure. Therefore, the conceptual design phase of this project will need to embrace a systems view of change and how it propagates through what is a complex system comprising an aircraft segment and its systems, subsystems and components; and the ground segment comprising the ground charging systems and subsystems.

This section of this EP system modification case study focusses on the aircraft segment. Although the conceptual design methodology incorporates the aircraft and ground infrastructure, the change impact of the EP system modification on the ground charging station infrastructure is minimal insofar that changes in the EP system will not invoke a redesign of ground charging station architecture. The extent of change is restricted to interfaces such as the charging receptacle. In this instance it is assumed that charging connectors and receptacles currently in use for EVs would be adapted without little or no change for aviation applications. Nevertheless, the major change in the system is the provision of the DC Fast Charger mains supply infrastructure, which does not currently exist at airports. However, the provision of mains supply infrastructure is a standard electrical installation activity, and as such is outside the scope of this study. Therefore, the impact of the EP system modification is limited here to the aircraft segment only. If in other cases an analysis of change propagation is required across aircraft-ground segments then the methods and techniques shown here can be extended as another MDM, as described below.

### 3.6.2 Change propagation analysis

This approach to change propagation is based on the work undertaken by various researchers (Clarkson et al. 2001, Eckert et al. 2001, Eckert et al. 2009, Eckert et al. 2006, Keller et al. 2005 a & b, Koh & Clarkson 2009 and Koh et al. 2012), as described in Chapter 5 and Appendix 1. In this context the changes resulting from the EP system modification results in changes not only to the interfaces but also to other aircraft systems, subsystems and components. This section deals with complexity of the product, whereas the broader conceptual design methodology described in this thesis deals with the design process and complexity of the design description.

This section focuses on the methods used to model the dependencies between system, subsystems and components of this EP system modification solution. This method can be used to support the risk/impact assessment of the solution which fits into the broader change management and project management processes within a design organisation.

Clarkson et al. (2001), outlines a method that predicts change propagation in complex design. The method outlined is referred to as the Change Propagation Method (CPM), which is fully described in Chapter 4, Chapter 5 and implemented in Appendix 1. This Section illustrates the practical application of this method which has been simplified and adapted to fit within a broader matrix-based conceptual design methodology and is based on a combination of Design Structure Matrices (DSM) as described by Clarkson et al. 2001), MDM and Domain Mapping Matrices (DMM) as described by Koh et al. (2012). As described in Chapter 4, Chapter 5 and Appendix 1 these matrices make up the change propagation analysis element of this methodology and examine the impact of change using direct dependency relationship which are then used to derive risks for further consideration by the design team.

The first step in this change propagation analysis follows the general process outlined by Clarkson et al. (2001) which is based on a Design Structure Matrices (DSM) combined with a propagation model. This change propagation model is derived from the product itself in this case being an indicative EP system as illustrated in Figure 68. This EP system can be decomposed into subsystems and components and their related dependencies. Furthermore, the existing aircraft propulsion and fuel system can be decomposed into subsystems and components. Again, the existing AVGAS propulsion and fuel system will possess dependencies although the design objective will ensure that the existing aircraft powerplant and subsystems and components will be removed from the firewall forward location and replaced with the EP system functional equivalent. This illustrated conceptually in Figure 68 where the EP system components replace the existing SI aircraft engine. It should be noted however that the modification also comprises electric motor and battery management instrumentation which is installed in the cockpit, along with motor throttle controls. Therefore, this EP system modification is a relatively simple physical integration engineering activity at the subsystems level.

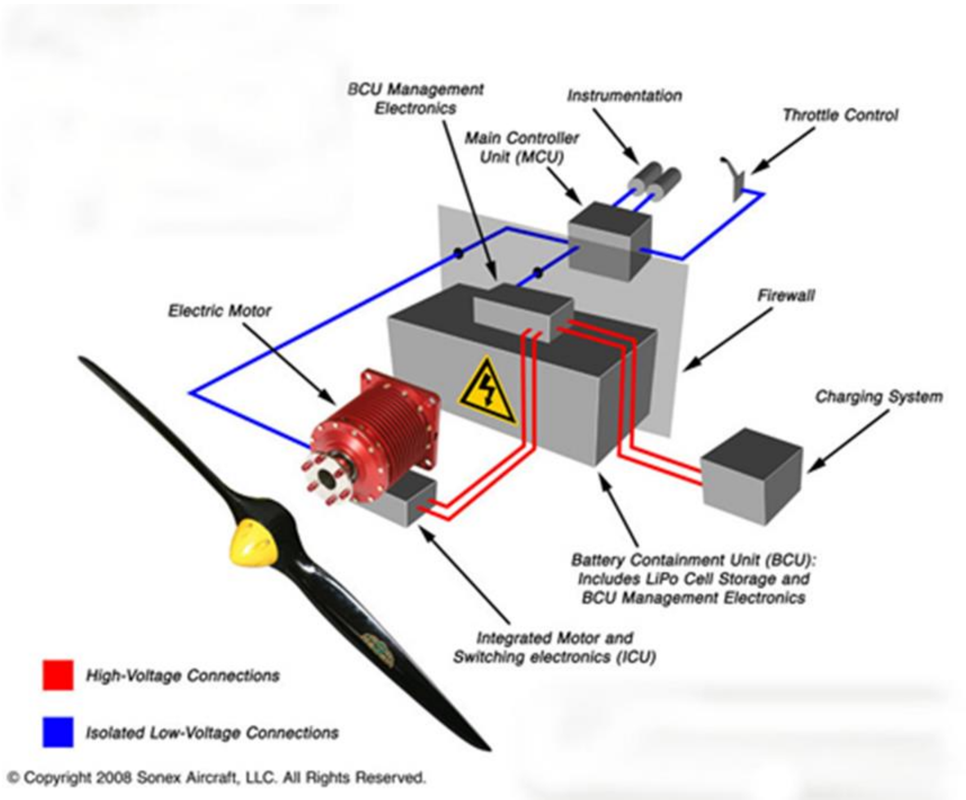


Figure 68. Indicative electric propulsion system schematic
Source: Sonex Aircraft LLC (2008)

### *3.6.2.1 Modification risk elements*

As described previously in Chapter 5, the Change Propagation Method (CPM) as outlined by Clarkson et al. (2001), relies on systems decomposition and dependencies as described above, with the dependencies obtained from an analysis of the aircraft architecture. These are defined in terms of the likelihood that the redesign of the subsystem or component will force the redesign of another and the subsequent impact, or extent, of that redesign. This is characterised in the likelihood definitions provided in Table 88.

Therefore, all parameters in a complex system are connected with each other through the interaction of subsystems and components to generate the desired behaviour of the entire system. The degree that this interaction occurs is determined by the linking parameters types and their attributes as defined in Table 88, and the direct impact definitions provided in Table 89. The likelihood definitions are derived from linking parameters as described by Eckert et al. (2004) and Koh et al. (2012), and are further described in Chapter 5.

Table 88. Direct likelihood definitions (*l*)

| Direct likelihood | | Definitions |
|---|---|---|
| Frequent | 0.9 | Multiple linkage types comprising mechanical, electrical and informational flows |
| Probable | 0.6 | Multiple linkage types, however omitting a linkage type(s) (e.g. mech static/mech dynamics) |
| Occasional | 0.3 | Single linkage or single type (e.g. mech static, mech dynamic, spatial, fluid flow, fluid flow dynamic) |

As described earlier the second element of this process is the assessment of the design change and the subsequent impact, or extent, of that redesign. Table 89 provides definitions of the consequences of design changes on the extent of engineering redesign required to address the change or propagation impacts.

Table 89. Direct impact definitions (*i*)

| Direct Impact | | Definitions |
|---|---|---|
| High | 0.9 | High level of redesign work to address change or change propagation impacts. |
| Medium | 0.6 | Medium level of redesign work to address change or change propagation impacts. |
| Low | 0.3 | Minimal level of redesign work to address change or change propagation impacts. |

These two relationships are combined to represent a scale of impact severity as shown in Table 90, where the impact severity is defined as the product of the likelihood (*l*) of the change and the impact (*i*) or consequence of the subsequent change.

Table 90. Change impact severity matrix

| IMPACT CATEGORY | HIGH 0.9 | MEDIUM 0.6 | LOW 0.3 |
|---|---|---|---|
| FREQUENT 0.9 | 0.8 | 0.5 | 0.3 |
| PROBABLE 0.6 | 0.5 | 0.4 | 0.2 |
| OCCASIONAL 0.3 | 0.3 | 0.2 | 0.1 |

These likelihood (*l*) and impact (*i*) values may be derived from previous design modifications and from the experience of senior designers. For this reason, the change propagation methods described here should be conducted as a design team activity as described in Appendix 1.

The final outcome of this process is to represent impact severity (*S*) of the change in one subsystem or component resulting in a change in a neighbouring subsystem or component by propagation over a common interface. The theory underpinning this process is described at Chapter 5 and Appendix 1, where the resultant impact severity (*S*) can be calculated using the relationship as described by Clarkson et al. (2001).

#### *3.6.2.2 Modification subsystem to aircraft subsystems dependencies*

The Change Propagation Method (CPM) as outlined by Clarkson et al. (2001), relies on systems decomposition and dependencies involving a two-stage product model analysis of the aircraft architecture. Firstly, a number of aircraft subsystems or components are identified, which together represent the whole of the aircraft segment. This aircraft subsystem has been identified in the direct likelihood Design Structure Matrix (DSM) and follows the same subsystem functional composition as described in Appendix 1. The same structures and approaches have been applied to relate EP subsystems and components to aircraft subsystems. This is not repeated here for brevity. Furthermore, the process descriptions and associated with deriving Direct likelihood (*l)* and Direct Impact (*i*) matrices are not presented. Rather the resulting Mod severity risk matrix is provided as the final output of this change impact analysis process.

Table 91 shows the change severity/risk as a result of the modification subsystems impacting the aircraft subsystems. These values of change severity are colour coded in accordance with Table 90 to allow easy identification. For example, the results of this change severity matrix in Table 91 shows three occurrences of significant risk of change propagation. This relates to the dependencies associated with the motor, propeller and energy subsystem (batteries) where a change in motor configuration or size may require considerable redesign or rigorous application of design controls to mitigate change propagation risk to the other subsystems. The other occurrence of high change propagation risk is that associated with energy subsystem changes impacting aircraft fire protection subsystems. In this case the existing aircraft fire protection subsystems may be inadequate for a battery-based fire. Therefore, there are significant change propagation risks that impact the aircraft propulsion subsystems which then affects certification. This is further discussed in Section 3.8 which deals with certification Domain Mapping Matrices (DMM).

Table 91. Mod change severity/risk matrix – Mod subsystems to Aircraft subsystems

| **Mod Risk Matrix - Mod Subsystem to Aircraft Subsystem** | Fuselage | Engines/motors | Wings | Nacelles | Stabilisers | Propellers | Hydraulic system | Energy system | Engine auxiliaries | Flight Control System | Avionics | Auxiliary electrical | Cabling & Piping (C&P) | Landing gear | Equipment and Furnsihings (E&F) | Fire protection systems | Environmental Control Systems (ECS) | Ice and rain protection | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Fuselage | 0 | 0.5 | | 0.5 | | | | 0.5 | | | | | 0.3 | | 0.5 | | | | 2.4 |
| Engines/motors | 0.5 | 0 | | | | 0.3 | | | | | | | 0.3 | | | | | | 1.1 |
| Wings | | | 0 | | | | | | | | | | | | | | | | |
| Nacelles & fairings | 0.5 | 0.5 | | 0 | | 0.4 | | 0.5 | | | | | 0.3 | | | | | | 2.3 |
| Stabilisers | | | | | 0 | | | | | | | | | | | | | | |
| Propellers | 0.5 | 0.8 | | 0.5 | | 0 | | | | | | | | | | | | | 1.9 |
| Hydraulic system | | | | | | | 0 | | | | | | | | | | | | |
| Energy system | 0.5 | 0.8 | | | | | | 0 | | | | | 0.2 | | | | | | 1.5 |
| Engine auxiliaries | 0.3 | 0.3 | | | | 0.1 | | | 0 | | | | 0.3 | | | | | | 0.9 |
| Flight Control System | | | | | | | | | | 0 | | | | | | | | | |
| Avionics | 0.3 | | | | | 0.2 | | 0.4 | | | 0 | | 0.3 | | 0.5 | | | | 1.6 |
| Auxiliary electrical | 0.3 | 0.3 | | | | | | 0.2 | | | 0.1 | 0 | 0.3 | | 0.3 | | | | 1.4 |
| Cabling & Piping (C&P) | 0.5 | 0.3 | | 0.1 | | | | 0.2 | | | 0.2 | | 0 | | 0.3 | | | | 1.5 |
| Landing gear | | | | | | 0.2 | | | | | | | | 0 | | | | | 0.2 |
| Equipment and Furnishings (E&F) | 0.1 | | | | | | | | | | | | 0.1 | | 0 | | | | 0.2 |
| Fire protection systems | 0.5 | 0.1 | | 0.5 | | | | 0.8 | | | | | 0.1 | | | 0 | | | 2.1 |
| Environmental Control Systems (ECS) | | | | | | | | | | | | | | | | | 0 | | |
| Ice and rain protection | | | | | | | | | | | | | | | | | | 0 | |
| | 4.1 | 3.6 | | 1.7 | | 1.1 | | 2.6 | | | 0.3 | | 2.0 | | 1.6 | | | | |

An extension of the change propagation related DSM work described by Clarkson et al. (2001) includes the provision of an additional column and row providing totals of change severity. The additional column shown on the right-hand side of the change severity matrix shows the change severity total of each aircraft subsystem impacted by the EP system modification subsystems as described above. These totals highlight the degree that aircraft subsystems are affected by the EP system modification.

In a similar way, the additional row shown on the lower part of the change severity matrix shows the change severity total of each EP system modification related aircraft subsystem impacted by each aircraft subsystems. These totals highlight the EP modification subsystems that are highly impacted by changes to the aircraft subsystems.

#### 3.6.2.3 Modification components to aircraft subsystems dependencies

This DSM-based approach can be extended to other dependencies as described below using a slightly different mapping. In this case rather than group EP subsystems and components relative to aircraft subsystems, it is advantageous to assess EP components separately against the aircraft subsystems. In this way the direct change dependencies can be assessed for each EP component initiating a change against an aircraft subsystem. Again, the outcome of this process, being the EP system modification change severity matrix, is presented here as Table 92. For example, the results of this change severity matrix show two occurrences of significant risk of change propagation. These occurrences highlight the change propagation risks associated with integration of the electric motor to the propeller facilitating adequate propeller diameter; and the development and integration of electric controller to allow adequate cooling airflow within the cowling.

Again, the additional rows and columns to the right-hand side and lower part of the change severity matrix highlight those areas of severity for each impacted aircraft system and EP components respectively. As expected, all EP components impact the aircraft fuselage as part of the installation. While the electric motor impacted numerous aircraft subsystems.

Table 92. Mod change/severity risk matrix – Mod components to Aircraft subsystems

| **Mod Change Risk Matrix - Mod Components to Aircraft Subsystems** | Electric motor(s) | Propeller(s) | Batteries - Firewall forward | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Fuselage | 0.5 | | 0.5 | 0.5 | 0.4 | 0.4 | 0.1 | 0.1 | 0.2 | 0.2 | 0.4 | 3.2 |
| Engines/Motors | | 0.1 | 0.4 | 0.4 | 0.1 | 0.8 | | | 0.1 | 0.1 | | 1.9 |
| Wings | | | | | | | | | | | | |
| Nacelles & fairings | 0.5 | 0.4 | 0.5 | 0.5 | | 0.4 | | | | | 0.4 | 2.7 |
| Stabilisers | | | | | | | | | | | | |
| Propellers | 0.8 | | 0.4 | 0.4 | | | | | | | | 1.5 |
| Hydraulic system | | | | | | | | | | | | |
| Energy system | 0.5 | | 0.5 | 0.5 | | 0.5 | 0.1 | 0.1 | | | 0.1 | 2.4 |
| Engine auxiliaries | 0.3 | 0.1 | | | 0.3 | 0.5 | | | | | | 1.2 |
| Flight Control System | | | | | | | | | | | | |
| Avionics | | 0.2 | 0.1 | 0.1 | 0.1 | 0.4 | 0.2 | 0.2 | 0.4 | 0.4 | | 1.9 |
| Auxiliary electrical | 0.3 | | 0.4 | 0.4 | 0.1 | 0.2 | 0.2 | 0.2 | 0.1 | 0.1 | 0.4 | 2.2 |
| Cabling & Piping (C&P) | 0.5 | | 0.5 | 0.5 | 0.3 | | 0.2 | 0.2 | 0.1 | 0.1 | 0.1 | 2.5 |
| Landing gear | | 0.5 | | 0.5 | | | | | | | | |
| Equipment and Furnishings (E&F) | | | | | 0.2 | | 0.1 | 0.1 | 0.4 | 0.4 | | 1.1 |
| Fire protection systems | 0.2 | | 0.5 | 0.5 | | 0.5 | | | | | | 1.8 |
| Environmental Control Systems (ECS) | | | | | | | | | | | | |
| Ice and rain protection | | | | | | | | | | | | |
| | 3.7 | 1.3 | 3.9 | 4.4 | 1.4 | 3.7 | 0.8 | 0.8 | 1.2 | 1.2 | 1.3 | |

### *3.6.2.4 Modification components to components dependencies*

This same DSM approach is applied to assess change dependencies associated with EP component change initiation and the impact of these changes with other EP components. These changes may be necessary to account changes in component ratings or settings which may be specific to the integrated solution. The outcome of this process, being the EP modification component change severity matrix, is presented as Table 93.

Table 93 shows the change severity matrix resulting from changes to EP modification components impacting other EP components. As stated earlier these values of change severity are colour coded in accordance with Table 90 to allow easy identification. The results of this change severity matrix show three high change severity/risks. These high change severity risks correspond to impact of the motor on propeller; motor on speed controller; and speed controller on the motor. These dependencies are a result of the highly integrated EP solution where a change in sizing, geometry or specification of these components will result in the propagation of changes to the other components.

Again, the additional rows and columns to the right-hand side and lower part of the change severity matrix highlight those areas of severity for each impacted EP component respectively. The right-hand column shows that main contributor to change severity/risk (value '2.3') is instigated by various EP components impacts the speed controller and cooling subsystems. This is expected, as changes in EP component geometry, sizing or location will impact the specification of this speed controller and the cooling of related components. In all likelihood the speed controller may be a custom developed solution matched to the motor and battery components. The lower row shows the main contributor to change severity/risk (value '2.4') is instigated by the motor impacting the various EP components as noted. As noted above two high change severity risks occur in this particular column.

Table 93. Mod change/severity risk matrix – Mod components to Mod components

| **Mod Change Risk Matrix - Mod Components to Mod Components** | Electric motor(s) | Propeller(s) | Batteries - Firewall forward | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Electric motor(s) | 0 | 0.2 | 0.4 | 0.4 | | 0.8 | | | | | | 1.7 |
| Propeller(s) | 0.8 | 0 | | | | | | | | | | 0.8 |
| Batteries - Firewall forward | 0.4 | 0.4 | 0 | 0.2 | | | | | 0.2 | 0.2 | 0.2 | 1.4 |
| Batteries - Slipper power pod | 0.4 | 0.4 | 0.2 | 0 | | | | | 0.2 | 0.2 | 0.2 | 1.4 |
| Electric controller - Throttle | | | | | 0 | 0.5 | | | | | | 0.5 |
| Speed ctrl & motor cooling | 0.8 | 0.2 | 0.4 | 0.4 | 0.4 | 0 | | 0.2 | | | | 2.3 |
| Circuit breakers | 0.1 | | 0.1 | 0.1 | | 0.1 | 0 | 0.1 | 0.1 | 0.1 | 0.1 | 0.7 |
| Master switch | | | | | | | 0.1 | 0 | | | | 0.1 |
| State of Charge (SOC)instrumentation | | | 0.5 | 0.5 | | | 0.1 | | 0 | 0.4 | | 1.5 |
| State of Health (SOH) Instrumentation | | | 0.5 | 0.5 | | | 0.1 | | 0.4 | 0 | | 1.5 |
| Recharge receptacle/exchange | | | 0.2 | 0.2 | | | | | | | 0 | 0.4 |
| | 2.4 | 1.1 | 2.3 | 2.3 | 0.4 | 1.4 | 0.3 | 0.3 | 0.8 | 0.8 | 0.5 | |

### 3.6.3 Change propagation risks and risk matrix

These DSM-based change propagation methods are useful in highlighting the severity of design change propagation risks as described fully in Appendix 1. These change propagation risks can be captured within a traditional project risk matrix, as shown by Table 94. Table 94 shows an example of such a risk matrix produced from the analysis of change propagation severity risks as described in this step.

Table 94. Electric propulsion system modification change propagation risks – sample

| Change Propagation Risk Matrix | | | | | |
|---|---|---|---|---|---|
| **Risk No.** | **Propagation Matrix** | **Risk Title** | **Description** | **Rated Risk** | **Actions & Comments** |
| 1 | Mod subsystem to Aircraft subsystem | Motor - Propeller | There is a chance that changes to the motor sizing or interfaces will propagate changes to the propeller system which will impact design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 2 | Mod subsystem to Aircraft subsystem | Motor - Energy system | There is a chance that changes to the motor sizing or interfaces will propagate changes to the battery energy system which will impact design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 3 | Mod subsystem to Aircraft subsystem | Energy system - Fire protection system | There is a chance that changes to the battery energy sizing or configuration will propagate changes to the fire protection/suppression system which will impact design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 4 | Mod subsystem to Aircraft subsystem | Motor - Fuselage | There is a chance that changes to the motor sizing or interfaces will propagate changes to the fuselage firewall and nose landing gear interfaces which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 5 | Mod subsystem to Aircraft subsystem | Motor - Nacelles & Fairings | There is a chance that changes to the motor sizing or interfaces will propagate changes to the engine nacelles and related fairings which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 6 | Mod subsystem to Aircraft subsystem | Fuselage - Motor | There is a chance that the current fuselage geometry or interfaces will propagate changes or constraints to motor sizing or interfaces which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 7 | Mod subsystem to Aircraft subsystem | Fuselage - Nacelles & Fairings | There is a chance that the current fuselage geometry or interfaces will propagate changes or constraints to nacelles or fairings which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 8 | Mod subsystem to Aircraft subsystem | Fuselage - Propeller | There is a chance that the current fuselage geometry or interfaces will propagate changes or constraints to propeller sizing or geometry which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |

| Change Propagation Risk Matrix | | | | | |
|---|---|---|---|---|---|
| **Risk No.** | **Propagation Matrix** | **Risk Title** | **Description** | **Rated Risk** | **Actions & Comments** |
| 9 | Mod subsystem to Aircraft subsystem | Fuselage - Energy system | There is a chance that the current fuselage geometry, volume or interfaces will propagate changes or constraints to the battery energy system which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 10 | Mod subsystem to Aircraft subsystem | Fuselage - Cabling & Piping | There is a chance that the current fuselage geometry, volume or interfaces will propagate changes or constraints to electrical cabling which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 11 | Mod subsystem to Aircraft subsystem | Fuselage - Fire protection system | There is a chance that the current fuselage geometry, volume or interfaces will propagate changes to the fire protection/suppression system which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 12 | Mod subsystem to Aircraft subsystem | Energy system - Nacelles & fairings | There is a chance that changes to the battery energy sizing or configuration will propagate changes to firewall forward nacelle and slipper power pod fairing which will impact design costs and schedule. | Medium | Medium-High level design controls required to mitigate impact of design change propagation |
| 13 | Mod components to Mod components | Electric motor - Propeller | There is a chance that changes to the electric motor sizing or configuration will propagate changes to propeller sizing or configuration which will impact design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 14 | Mod components to Mod components | Electric motor - Speed controller & cooling | There is a chance that changes to the electric motor sizing or configuration will propagate changes to speed controller sizing or configuration which will impact design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |
| 15 | Mod components to Mod components | Speed controller & cooling - Electric motor | There is a chance that changes to the electric motor sizing or configuration will propagate changes to speed controller sizing or configuration which will impact design costs and schedule. | High | Rigorous design controls required to mitigate impact of design change propagation |

## 3.7 ENGINEERING DOMAIN MAPPING MATRIX – STEP 6

### 3.7.1 Overview

In this section the methods used to evaluate the impact of changes to subsystems is extended into the engineering domain in order to evaluate the impact of design costs resulting from the EP system modification. This approach uses the results of the change propagation analysis described previously which is then used to estimate the impact on engineering development costs. The method is further extended to determine the impact of the EP system modification on the development of the system specification and also the design parameters on subsequent engineering development costs.

This distinction between engineering development and certification, as dealt with in the next section, is made here to ensure that the methodology accounts for the discrete activity that involves conceptual design and related cost estimates. These engineering activities, like other steps in the conceptual phase are iterative in nature and the use of these engineering Domain Mapping Matrices (DMM) would be undertaken within processes as described in the previous sections. As before one output of this DMM-based assessment are the draft Systems Specification document detailing the aircraft EP system modification and the related ground charging station infrastructure. The other outputs of this assessment were a Cost Breakdown Structure (CBS) which estimates engineering costs associated with the EP system modification and the ground charging system as well as the Requirements Allocation Sheet which documents the relationship between allocated function, performance and the physical system.

### 3.7.2 Design change impacts

Table 95 and Table 96 shows the modification design engineering DMM describing the relationship of the aircraft EP system modification components to requirements and design change parameters. This analysis evaluates the impact of the EP modification components on requirements and design changes parameters to facilitate a functional view and physical view of the modification. This aspect of the DMM provides a ‘reverse’ view of the functional analysis step undertaken in Systems Engineering. One output from this process is the provision of a draft Requirements Allocation Sheet which can be used in later design phases of the project. Again, it is not intended to produce this Requirements Allocation Sheet, as this Systems Engineering activity which is out of scope in this thesis.

Table 95 and Table 96 shows the EP system modification DMM. These two tables have been split into two parts to facilitate presentation in this thesis. In actual application, one combined DMM would be used to present modification design engineering data. Table 95 shows the EP system modification DMM which is based on design change propagation data determined from Section 3.6. Specifically, the DMM data shown are transposed from Table 92 and Table 93 modification change severity/risk results, where the colour coding provides a measure of engineering development change related risk impacting engineering cost. These DMM data are transposed to provide EP modification components as common columns to which cost data estimates can be aligned either from an aircraft system or as a modification component. The right-hand side of the DMM shown by Table 95 shows engineering costs derived from EP system modification impacts on aircraft subsystems. These costs are broken down into (1) engineering management (2) engineering design, and (3) engineering development and support, using a similar structure as described in Fabrycki & Blanchard (1991). The determination of these engineering costs is not within scope of this thesis. However, it is sufficient in this conceptual design phase to identify those costs impacted by the EP system modification as highlighted blue in Table 95. The basis of cost estimation is provided in Chapter 4.

## Table 95. EP System Mod design engineering DMM – Part 1

| Mod Components to Aircraft Subsystems | Electric motor(s) | Propeller(s) | Batteries - Firewall forward | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | Engineering Costs: Engineering management | Engineering Costs: Engineering design | Engineering Costs: Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Fuselage | 0.5 | | 0.5 | 0.5 | 0.4 | 0.4 | 0.1 | 0.1 | 0.2 | 0.2 | 0.4 | | | |
| Engines | | 0.1 | 0.4 | 0.4 | 0.1 | 0.8 | | | 0.1 | 0.1 | | | | |
| Wings | | | | | | | | | | | | | | |
| Nacelles | 0.5 | 0.4 | 0.5 | 0.5 | | 0.4 | | | | | 0.4 | | | |
| Stabilisers | | | | | | | | | | | | | | |
| Propellers | 0.8 | | 0.4 | 0.4 | | | | | | | | | | |
| Hydraulic system | | | | | | | | | | | | | | |
| Energy system | 0.8 | | 0.8 | 0.8 | | 0.8 | 0.1 | 0.1 | | | 0.1 | | | |
| Engine auxiliaries | 0.3 | 0.1 | | | 0.3 | 0.5 | | | | | | | | |
| Flight Control System | | | | | | | | | | | | | | |
| Avionics | | 0.2 | 0.1 | 0.1 | 0.1 | 0.4 | 0.2 | 0.2 | 0.4 | 0.4 | | | | |
| Auxiliary electrical | 0.3 | | 0.4 | 0.4 | 0.1 | 0.2 | 0.2 | 0.2 | 0.1 | 0.1 | 0.4 | | | |
| Cabling & Piping (C&P) | 0.5 | | 0.5 | 0.5 | 0.3 | | 0.2 | 0.2 | 0.1 | 0.1 | 0.1 | | | |
| Landing gear | | 0.5 | | 0.5 | | | | | | | | | | |
| Equipment and Furnishings (E&F) | | | | | 0.2 | | 0.1 | 0.1 | 0.4 | 0.4 | | | | |
| Fire protection systems | 0.2 | | 0.5 | 0.5 | | 0.5 | | | | | | | | |
| Environmental Control Systems (ECS) | | | | | | | | | | | | | | |
| Ice and rain protection | | | | | | | | | | | | | | |

| Mod Components to Mod Components Change Propagation | Electric motor(s) | Propeller(s) | Batteries - Firewall forward | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | Engineering management | Engineering design | Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Electric motor(s) | 0 | 0.2 | 0.4 | 0.4 | | 0.8 | | | | | | | | |
| Propeller(s) | 0.8 | 0 | | | | | | | | | | | | |
| Batteries - Firewall forward | 0.4 | 0.4 | 0 | 0.2 | | | | | 0.2 | 0.2 | 0.2 | | | |
| Batteries - Slipper power pod | 0.4 | 0.4 | 0.2 | 0 | | | | | 0.2 | 0.2 | 0.2 | | | |
| Electric controller - Throttle | | | | | 0 | 0.5 | | | | | | | | |
| Speed ctrl & motor cooling | 0.8 | 0.2 | 0.4 | 0.4 | 0.4 | 0 | | 0.2 | | | | | | |
| Circuit breakers | 0.1 | | 0.1 | 0.1 | | 0.1 | 0 | 0.1 | 0.1 | 0.1 | 0.1 | | | |
| Master switch | | | | | | | 0.1 | 0 | | | | | | |
| State of Charge (SOC) instrumentation | | | 0.5 | 0.5 | | | 0.1 | | 0 | 0.4 | | | | |
| State of Health (SOH) Instrumentation | | | 0.5 | 0.5 | | | 0.1 | | 0.4 | 0 | | | | |
| Recharge receptacle/exchange | | | 0.2 | 0.2 | | | | | | | 0 | | | |

Table 96 shows the second part of the modification design engineering DMM describing the relationship of the EP system modification components to requirements and design change parameters. This analysis evaluates the impact of the EP system modification components on requirements and design changes parameters to facilitate a functional view and physical view of the EP system modification. This aspect of the DMM provides a 'reverse' view of the functional analysis step undertaken in Systems Engineering as described earlier.

Table 96. EP System Mod design engineering DMM – Part 2

| **Mod Components to System Specification Reqts** | Electric motor(s) | Propeller(s) | Batteries - Firewall forward | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | Engineering management | Engineering design | Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| The modification shall provide a time of climb of less than 10 minutes to 14000ft altitude with the maximum useful load.. | | | | | | | | | | | | | | |
| TThe modification shall provide a useful load of least 500 kg to 14000 ft altitude with a time of climb of less than 10 minutes. | | | | | | | | | | | | | | |
| The modification lifecycle costs (LCC) shall be kept to a minimum | | | | | | | | | | | | | | |
| The modification shall comply with airworthiness standards and minimise impact to the TC basis. | | | | | | | | | | | | | | |
| The modification shall minimise emissions. | | | | | | | | | | | | | | |
| The modification shall be compatible with a range of single engine light aircraft types. | | | | | | | | | | | | | | |
| The modification shall not impact passenger, baggage or cargo space. | | | | | | | | | | | | | | |

| **Mod Components to Design Change Parameters** | Electric motor(s) | Propeller(s) | Batteries - Firewall forward | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | Engineering management | Engineering design | Engineering development support |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Decrease EP system installation drag | | | | | | | | | | | | | | |
| Decrease EP system installation weight | | | | | | | | | | | | | | |
| Reduce EP system related LCC | | | | | | | | | | | | | | |
| Maintain power output | | | | | | | | | | | | | | |
| Maintain aircraft payload volume and space capacity | | | | | | | | | | | | | | |

### 3.7.3 Cost estimation aspects

As stated above the means of cost estimation as shown in the right-hand columns of Table 95 and **Error! Reference source not found.** which are based on Cost B reakdown Structure (CBS) definitions as detailed by Fabrycki & Blanchard (1991). These are fully described in Chapter 4.

## 3.8 CERTIFICATION DOMAIN MAPPING MATRIX – STEP 7

### 3.8.1 Overview

This section describes the methods used to evaluate the impact of the EP system modification on airworthiness certification requirements (certification domain) along with the provision of associated certification cost estimates. As described earlier, this method which relies on using a DMM to provide rigor and coverage to the impact of change propagation on aircraft certification requirements and also provides a framework from which to estimate the associated costs noting the change severity risk elements.

### 3.8.2 Type certification basis

The Cessna 182P aircraft was originally type certificated in the US, and FAA Type Certificate 3A13 (2006) was issued to the aircraft and described as follows:

> *Part 3 of the Civil Air Regulations dated November 1, 1949, as amended by 3-1 through 3-12 and Paragraph 3.112 as amended October 1, 1959, for the Model 182E and on. In addition, effective S/N 18266591 through 18268586, FAR 23.1559 effective March 1,1978.*

### 3.8.3 Certification airworthiness standards

In this case study the EP system modification impacts the original type certification basis as described above affecting the airframe and engine. The original aircraft certification basis is Civil Airworthiness Regulations (CAR) Part 3 and Civil Airworthiness Regulations Part 13 for the engine. Both CAR Part 3 and CAR Part 13 are historical regulations forming the certification basis of the Cessna 182P and the Continental O-470 engine. CAR Part 3 does not include provision for electric propulsion system requirements under Sub-part E as it deals with powerplant installations for reciprocating engines. Furthermore, CAR Part 13 deals exclusively with reciprocating and turbine engines, and does not incorporate requirements for electric propulsion systems. For this reason, the impact of the EP system on the engine type certification basis will not be presented here as it does not provide any additional information not already provided in Appendix 1.

It should be noted that the intent here is not to establish an appropriate certification basis for this EP system modification. Given that the original certification basis does not account for electric powerplants then a more appropriate airworthiness standard may be the most recent amendment of FAA 14 CFR Part 23, which now incorporates performance-based airworthiness requirements. This FAR Part 23 Amendment 64 comprises a new framework, where an applicant demonstrates compliance with the FAA-accepted ASTM consensus standards. These ASTM consensus standards F2840-14 (2014) will be made available for the development of electric propulsion systems using batteries or hybrid power systems as fuel, and will provide a “means of compliance” to satisfy the new Part 23 fuel system performance-based standard. Nevertheless, the establishment of a suitable certification basis for a CAR Part 3 aircraft modified by an EP system installation is a separate issue, and as such it is not within the scope of this thesis. Therefore, this case study will proceed with an analysis based on the original type certification basis in order to demonstrate this methodology DMM structure.

Table 97 and Table 98 provides excerpts of the CAR Part 3 aircraft certification DMM which is based on design change propagation data using change likelihood and impact methods as described in Appendix 1. This DMM therefore presents the change severity/risk matrix for the EP system modification impact on the certification basis as shown. For each applicable EP system modification component, a likelihood and impact assessment is undertaken through reference to the particular requirement as defined in CAR Part 3. It should be noted that Table 97 and Table 98 lists the CAR Part 3 sub-paragraph requirements headings only. It is therefore necessary to refer to the actual content of each sub-paragraph as found in CAR Part 3 in order to make this assessment. The process follows the same change propagation analysis as described earlier in this section and is therefore best undertaken by the design team. In addition, reference should also be made to change propagation analysis results in the previous section (Table 88, Table 89 and Table 90) in order to characterise change severity/risks along with functional and performance impacts borne out from each particular sub-paragraph requirement.

Table 97 evaluates the impact of the EP system modification on the airworthiness certification domain using the change propagation type methods as discussed earlier. CAR Part 3 – Subpart A contains general requirements which are applicable to all EP system modification components. For this reason, the costings are applicable to certification management activities only as no compliance findings are required. The requirements in Sub-part B given by §3.83 to §3.86 are requirements pertaining to stall speed, takeoff, climb and landing performance which are impacted by the EP system modification. These severity/risk values indicate the potential impact of the new electric powerplant installation on stall speed, as well as the impact on takeoff, climb, and landing performance. As discussed earlier, the corresponding certification costs in this instance are determined by the highest severity/risk impact corresponding to the requirement. For example, the aircraft climb performance requirement (§3.85) impacted by the EP system will affect certification management, compliance findings, support equipment development and test operations costs.

Table 97 is the second excerpt of CAR Part 3 Sub-part E which deals specifically with reciprocating engine installations. The general requirements of Sub-part E given by sub-paragraphs §3.411 through §3.422 apply to modification components and general setup details, and provide the basis for later detailed requirements in this Sub-part. The remainder of these requirements in this Sub-part deal with reciprocating engine fuel systems, fuel pumps, oil systems, lines fittings and accessories etc. These requirements are clearly not applicable to EP systems, hence the adoption of an appropriate certification basis for this EP system modification is necessary to assure airworthiness of this EP system modification as discussed earlier.

The right-hand side of the DMM shown by Table 96 and Table 97 presents the certification costs resulting from EP system modification as described above. These costs are broken down into (1) certification management (2) airworthiness compliance findings and support, (3) equipment development and instrumentation support, and (4) test operations, again using a similar structure as described in Fabrycki & Blanchard (1991). The determination of these certification costs is not within scope of this thesis. However, the approach here is to use the results of the certification severity/risk matrix to inform the particular certification cost categories including allowances for severity/risk.

Table 97. CAR 3 aircraft certification DMM – Excerpt 1

| | | | | | | | | | | | | Certification costs | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Mod Components to CAR 3 Airworthiness Reqts - Airplanes** | Electric motor(s) | Propeller(s) | Batteries - Firewall forward - Cowl | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | Certification management | Airworthiness compliance & support | Equip development & inst support | Test operations |
| **CAR 3 - Subpart A - General** | | | | | | | | | | | | | | | |
| §3.0 Applicability of this part. | | | | | | | | | | | | | | | |
| §3.1 Definitions | | | | | | | | | | | | | | | |
| §3.10 Eligibility for type certificate | | | | | | | | | | | | | | | |
| §3.11 Designation of applicable regulations | | | | | | | | | | | | | | | |
| §3.12 Recording of applicable regulations | | | | | | | | | | | | | | | |
| §3.13 Type certificate | | | | | | | | | | | | | | | |
| §3.14 Data required | | | | | | | | | | | | | | | |
| §3.15 Inspections & tests | | | | | | | | | | | | | | | |
| §3.16 Flight tests | | | | | | | | | | | | | | | |
| §3.17 Airworthiness experimental and production certs | | | | | | | | | | | | | | | |
| §3.18 Approval of materials parts, processes and appliances | | | | | | | | | | | | | | | |
| §3.19 Changes in type design | | | | | | | | | | | | | | | |
| §3.20 Airplane categories | X | X | X | X | X | X | X | X | X | X | X | | | | |
| **Subpart B - Flight Requirements** | | | | | | | | | | | | | | | |
| **General** | | | | | | | | | | | | | | | |
| §3.61 Policy re proof of compliance. | | | | | | | | | | | | | | | |
| §3.62 Flight test pilot. | | | | | | | | | | | | | | | |
| §3.63 Noncompliance with test requirements | | | | | | | | | | | | | | | |
| §3.64 Emergency egress | | | | | | | | | | | | | | | |
| §3.65 Report | | | | | | | | | | | | | | | |
| §3.71 Weight & balance | | | | | | | | | | | | | | | |
| §3.72 Use of ballast | | | | | | | | | | | | | | | |
| §3.73 Empty weight | | | | | | | | | | | | | | | |
| §3.74 Maximum weight | | | | | | | | | | | | | | | |
| §3.75 Minimum weight | | | | | | | | | | | | | | | |
| §3.76 Centre of gravity location | | | | | | | | | | | | | | | |
| **Performance requirements** | | | | | | | | | | | | | | | |
| §3.81 Performance | | | | | | | | | | | | | | | |
| §3.82 Definition of stalling speeds | | | | | | | | | | | | | | | |
| §3.83 Stalling speed. | 0.3 | 0.3 | | | | | | | | | | | | | |
| §3.84 Takeoff. | 0.3 | 0.3 | 0.3 | 0.3 | | 0.3 | | | | | | | | | |
| §3.85 Climb. | 0.3 | 0.3 | 0.3 | 0.3 | | 0.3 | | | | | | | | | |
| §3.86 Landing. | 0.3 | 0.3 | 0.3 | 0.3 | | 0.3 | | | | | | | | | |

Table 98. CAR 3 aircraft certification DMM – Excerpt 2

| | | | | | | | | | | | | Certification costs | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **Mod Components to CAR 3 Airworthiness Reqts - Airplanes** | Electric motor(s) | Propeller(s) | Batteries - Firewall forward - Cowl | Batteries - Slipper power pod | Electric controller - Throttle | Speed ctrl & motor cooling | Circuit breakers | Master switch | State of Charge (SOC) instrumentation | State of Health (SOH) Instrumentation | Recharge receptacle/exchange | Certification management | Airworthiness compliance & support | Equip development & inst support | Test operations |
| **Subpart E—Powerplant installations reciprocating engines** | | | | | | | | | | | | | | | |
| **General** | | | | | | | | | | | | | | | |
| §3.411 Components. | 0.3 | 0.3 | 0.3 | 0.3 | 0.3 | 0.3 | | | | | | | | | |
| §3.415 Engines. | 0.5 | | | | | | | | | | | | | | |
| §3.416 Propellers. | | 0.5 | | | | | | | | | | | | | |
| §3.417 Propeller vibration. | | 0.5 | | | | | | | | | | | | | |
| §3.418 Propeller pitch and speed limitations. | | 0.5 | | | | | | | | | | | | | |
| §3.419 Speed limitations for fixed pitch propellers, ground adjustable pitch propellers and automatically varying pitch propellers. | | 0.5 | | | 0.5 | 0.5 | | | | | | | | | |
| §3.420 Pitch and speed limitations for controllable pitch propellers without constant speed controls. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.421 Variable pitch propellers with constant speed controls. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.422 Propeller clearance. | | 0.3 | | | | | | | | | | | | | |
| **Fuel System** | | | | | | | | | | | | | | | |
| §3.429 General. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.430 Fuel system arrangement. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.431 Multiengine fuel system arrangement. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.432 Pressure crossfeed arrangements. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.433 Fuel flowrate. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.434 Fuel flowrate for gravity feed systems. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.435 Fuel flowrate for pump systems. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.436 Fuel flowrate for aux fuel systems and fuel transfer systems. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.437 Determination of unusable fuel. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.438 Fuel system hot weather operation. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.439 Flow between interconnected tanks. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.440 Fuel tanks: General. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.441 Fuel tank tests. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.442 Fuel tank installation. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.443 Fuel tank expansion space. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.444 Fuel tank sump. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.445 Fuel tank filler connection. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.446 Fuel tank vents and carburetor vapor vents. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.447 Fuel tank vents. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.448 Fuel tank outlet. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| **Fuel Pumps** | | | | | | | | | | | | | | | |
| §3.449 Fuel pump and pump installation. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.550 Fuel system lines, fittings and accessories. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.551 Fuel valves. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.552 Fuel strainer. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.553 Fuel system drains. | X | X | X | X | X | X | X | X | X | X | X | | | | |
| §3.554 Fuel system instruments. | | | | | | | | | 0.3 | 0.3 | | | | | |

## 3.9 CONCEPT ANALYSIS – STEP 8

### 3.9.1 Overview

As described earlier in this Appendix, the main metrics associated with this aircraft EP system modification are those associated with aircraft climb performance, costs and sustainability as described in the QFD matrix. In addition, similar metrics can be attributed to the ground charging station infrastructure where charging performance and cost and sustainability impact are important attributes underpinning the overall systems solution. Given that this conceptual design methodology is an early step in the design process, these metrics can be further modelled using the morphological matrix quantisation as a starting point to provide initial estimates of system attributes for later consideration. The output of this activity is a Design Specification, where a summary of this conceptual design process is documented for further refinement in later design phases.

### 3.9.2 Useful load-climb performance

This case study has evaluated useful load-climb performance as part of the initial sizing of the EP system modification, which also supported the quantisation of the aircraft morphological matrix.

#### *3.9.2.1 Climb performance*

A climb performance model was developed which accounted for a range of electric motor types, battery specific energy densities, propeller types and battery locations. This climb performance model determined time of climb to two altitudes (10000 ft and 14000 ft), energy of climb, reserve energy and recuperative energy. Table 99 provides an estimate of the time of climb performance of the Cessna 182P aircraft modified with the candidate Siemens AG SP260D motor installation. This climb performance data was estimated at an aircraft maximum takeoff weight of 2950 lbs (1338 kg).

Table 99. Climb performance – Cessna 182P aircraft modified with Siemens AG SP260D motor/Slipper power pod – Config 10

| Altitude | Time of climb at MTOW (minutes) |
|---|---|
| 10,000 ft (3050 m) | 6.57 |
| 14,000 ft (4300 m) | 9.45 |

#### *3.9.2.2 Comparable climb performance results*

As stated in Table 99, the estimated time of climb performance for the modified Cessna 182P aircraft is 6.57 minutes to 10000 ft, at a maximum takeoff weight of 2950 lbs (1338 kg). An Extra 330LE prototype aircraft fitted with the **same** Siemens AG SP260D electric motor recently set an *actual* time to climb record as reported in Siemens AG (2016). The maximum takeoff weight of the Extra 330LE aircraft was approximately 1000 kg (2205 lbs) which was 338 kg (745 lbs) less than the case study Cessna 182P aircraft. The record set by this Extra 330LE aircraft was 4 minutes and 22 seconds to 10000 ft. This Extra 330LE time of climb result is comparable to the heavier Cessna 182P case study aircraft setup, and therefore is a point of validation supporting this analysis.

#### *3.9.2.3 Useful load*

Useful load was determined by traditional weight accounting methods based on reciprocating engine powerplant items removed, and electric propulsion system items added as part of the modification. Given that this was a simple process the relationships were represented in the quantisation where these weights determined the useful load which was then normalised as a metric. A simple calculation was carried out to ensure that the weight removed forward of the firewall was balanced with the weight added (i.e. battery weight) to maintain the Centre of Gravity (CG) of the aircraft. In most concepts the battery weight mounted forward of the firewall exceeded the weight removed. Therefore, these concepts were eliminated from further consideration as a satisfactory balance solution cannot be achieved. The Slipper power pod mounted beneath the fuselage allows the residual battery weight to be mounted in this location, and therefore solves this CG issue. This Slipper power pod results in increased aircraft drag which was accounted for in the climb performance model where applicable.

The useful load and time of climb performance metrics were normalised and were used in a *FoM* to determine the best solutions for this skydiving mission. Although some concepts provided adequate climb rate performance the requirements/change options ranked useful load as being more important than time of climb. Useful load is weighted higher as this load equates to the number of skydivers that can be carried to the jump altitude. The number of skydivers carried can be viewed as potential revenue and hence this is the value proposition for this mission. The *FoM* incorporates this propulsion system cost metric as well as climb performance with the highest scoring concept being Configuration 10.

### 3.9.3 Sensitivity analysis

A sensitivity analysis was undertaken to investigate the impact of battery specific energy density, motor peak specific power and propeller type on useful load and total battery weight. This analysis has focused on useful load and battery weight as these are parameters that determine the viability of the EP system modification as described earlier.

Table 100 provides useful load and total battery weight as a function of battery specific energy density as determined from Table 81 for Configuration 14. These parameters are determined for the skydiving mission involving a climb to 14000 ft altitude at maximum aircraft weight. Also shown is the relationship of firewall forward battery weight, which is constant to maintain aircraft CG, and the residual Slipper power pod battery weights required as function of battery specific energy density. As expected, increasing battery specific energy density reduces total battery weight and increases useful load. It can be seen that a battery specific energy density of about 240 Wh/kg the useful load reduces to about 500 kg, which is threshold requirement as stated in Table 57. Therefore, on his basis the minimum battery specific energy density would be about 240 Wh/kg to ensure that this EP system modification complies with requirements.

Table 100. Battery specific energy – Config 10 – Climb to 14000 ft

| Battery specific energy density (Wh/kg) | Useful load (kg) | Total battery weight (kg) | Firewall forward battery weight (kg) | Slipper pod battery weight (kg) |
|---|---|---|---|---|
| 160 | 369.6 | 378.8 | 170.4 | 207.8 |
| 180 | 411.6 | 336.1 | 170.4 | 165.8 |
| 200 | 445.3 | 302.5 | 170.4 | 132.2 |
| 220 | 472.8 | 275.0 | 170.4 | 104.7 |
| 250 | 505.8 | 242.0 | 170.4 | 71.1 |

Table 101 provides useful load and total battery weight as a function of the motor peak specific power as determined for a range of motors shown in Table 81. The useful load and total battery weight are determined for a battery specific energy of 250 Wh/kg. As expected, increasing motor peak specific power increases useful load. This useful load range is of the order of 64 kg for the motors shown. A threshold useful load of about 500 kg achieved with a motor peak specific power of about 5 kW/kg. This table also shows the relationship of total battery weight with motor peak specific power where the variation is proportional to motor peak power. However, the exception is the YASA 750 axial flux motor, where the contra-rotating propeller installation possesses a small increase in propeller efficiency. This results in a small reduction in total battery weight albeit with a penalty of a higher propeller installation weight. This reduction in total battery weight with increasing motor peak power is obviously a result of the higher climb rates achieved (with reduced time of climb) reducing the energy requirements. However, there are physical constraints that limit the installation of high-power electric motors on aircraft that were originally designed for engines rated at a lower power.

Table 101. Motor peak specific power – Climb to 14000 ft

| Motor description | Continuous power (kW) | Useful load (kg) | Total battery weight (kg) |
|---|---|---|---|
| YASA 750 axial flux | 150 | 442.3 | 283.4 |
| Emrax 348 | 150 | 460.7 | 293.1 |
| Siemens AG260D | 230 | 505.8 | 242.0 |

Table 102 illustrates effect of propeller energy recuperation on total battery weight for a Siemens AG260D motor installation flown on a skydiving mission as described above. As described earlier, this recuperative propeller arrangement operates as a turbine that regenerates power on the descent segment of the skydiving mission. This energy recovery acts to partially recharge the batteries which results in a reduction in total battery weight for the mission. This in turn results in an increase in useful load as shown in Table 102, without a significant penalty in propeller installation weight.

Table 102. Impact of propeller type – Siemens AG260D - Slipper power pod – Config 4 and Config 10

| Propeller type | Useful load (kg) | Total battery weight (kg) |
|---|---|---|
| Conventional | 478.3 | 269.5 |
| Recuperative | 505.8 | 242.0 |

This case study has identified some general trends in electric aircraft engaged in a skydiving mission as described above. These general trends are summarised in point form as follows:

- Propeller efficiency and recuperative power performance act to decrease total battery weight and hence increase useful load.

- Electric motors of high specific power have the potential to reduce climb energy requirements. However, for existing aircraft designs the ability to integrate high power motors is limited by physical constraints such as propeller ground clearance requirements.
- Estimated skydiving mission battery weight based on a specific energy density of 250 Wh/kg exceeds the firewall forward weight removed. Batteries need to be added elsewhere to complete the mission and maintain aircraft CG (i.e. Power slipper pod installation).
- Contrarotating propellers provide a small performance benefit but adds propeller weight to an already weight constrained installation.

### 3.9.4 Battery recharge and exchange

This case study has also evaluated various options for ground charging station infrastructure supporting the skydiving mission described above. This evaluation focused on cost metrics and battery recharge and exchange metrics affecting the flight duty cycle (the number of skydiving flights per day). This evaluation has shown that DC fast charge ground stations are necessary to achieve an acceptable number of flights per day. Whereas slower charging stations, at lower cost, can only provide three flights per day depending on the number of available exchange battery sets. The exchange of these battery sets weighing more than 250 kg on each flight could be problematic in terms of manual battery handling and aircraft turnaround time. Therefore, development of ground handling equipment to exchange batteries as a complete set may be required. A discussion with Glassock (2018, pers. comm., 8 October) has indicated that a battery exchange system is viable and may be an essential solution.

### 3.9.5 Emissions

Although not discussed directly in this case study, an electric aircraft possesses zero emissions at the point of operation. Compared to hydrocarbon fuels there are obviously no HC, CO, $CO_2$ or $NO_x$ emissions. More importantly, there are no lead emissions compared to AVGAS fuel.

### 3.9.6 Operating costs

The operating costs associated skydiving using an aircraft modified with an electric propulsion system is dependent on several parameters related to energy requirements of the flight, usage profile and the conditions associated with electricity supply. Given that there are currently no electric aircraft used for skydiving at this time, these operating costs are difficult to determine accurately. Like other analysis undertaken in this Appendix certain parallels can be drawn with Electric Vehicles, noting that flight duty cycle and usage rates will be different for electric aircraft. On this basis operating costs can be determined on the basis of simplifying assumptions related to electricity charges and battery useful life which must be amortised over each flight. The latter will not be considered in this study as the life of batteries is dependent on many variables, the specifics of which are not available at this early stage of the design lifecycle. It is important to note however that these battery amortisation cost will comprise a significant part of the operating cost. Nevertheless, the focus here are the costs associated with electric charges and the energy requirements of the skydiving mission.

#### *3.9.6.1 Electricity Consumption Charges*

Smith & Castellano (2015) state that operating costs includes the cost of electricity to charge the vehicles, with the annual electricity consumption cost for an EV owner determined by the electricity rate measured in dollars per kilowatt-hour ($/kWh) and the amount of electricity consumed.

In 2015, Smith & Castellano (2015) state that commercial electricity rates typically range from $US0.08-$0.15 per kWh, while industrial EV fleets could have lower rates. The consumption of electricity will vary based on the number of vehicles using the EV Supply Equipment (EVSE), power output of the EVSE, vehicle power acceptance rate, climate, and amount of time the vehicles charge. In this case it is assumed that the typical skydiving school/club operates only a single aircraft. Therefore, these EV cost savings do not apply.

#### *3.9.6.2 Electricity Demand Charges*

The report by Smith & Castellano (2015) notes that in addition to electricity costs based on energy consumption, many commercial and industrial facilities may be subject to power demand charges from the utility. The use of Level 2 and DCFC stations located at an airfield may result in higher electricity costs by increasing the facility's peak electricity demand. These demand charges can cause a monthly utility bill to increase by as much as four times, and therefore this may also need to factored into operational costs.

#### *3.9.6.3 Estimated flight costs*

Given commercial electricity rates as stated above, and an estimate of the climb energy requirements, an estimate of costs to complete a typical skydiving mission can be estimated.

- As above the commercial electricity rates in 2015 is assumed to be \$US 0.10 per kWh.
- The estimated climb and reserve flight energy requirements are determined for the preferred Configuration 14 as 60.4 kWh.

Therefore, the estimate cost per flight = 0.10 × 60.4 = **\$US 6.04 per flight**.

Note that this cost does not include battery amortisation costs or demand charges as discussed above. These battery amortisation costs will add significantly to each flight depending on battery life and battery charging rate.

#### *3.9.6.4 Other costs*

It should also be noted that the costs associated with this EP modification is highly dependent on other factors such as the number of flight hours flown annually, the EP modification cost, the insured value of the modified aircraft, and the loan repayment schedule. Accordingly, these other factors could adversely affect Life Cycle Costs presented in this thesis.

### 3.9.7 Other observations

This case study has highlighted that the EP system modification is a highly integrated system with change impacts and dependencies having a significant impact on the engineering design and certification effort. For example, a change in battery specific energy density will impact performance and weights as well as having impacts on motor, propeller and electric controller selection. Therefore, robust design controls are required to ensure that changes in the subsystem or components are managed accordingly.

Certification of the EP system modification also plays a major role in relation to establishing an acceptable solution. In all likelihood, some or all of the EP subsystems and components may require specific certification programs as well as airworthiness certification as outlined in this Appendix. In this instance the CPP output of this design methodology is the main means of establishing an acceptable certification basis and achieving certification.