CRANFIELD UNIVERSITY

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Development of an Approach and Tool to Improve the Conceptual-Design Process of Wing Box Structure of Low-Subsonic Transport Aircraft

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Supervisor: Professor John P Fielding

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This thesis is submitted in partial fulfilment of the requirements for the degree of Doctor of Philosophy

ABSTRACT

To produce a better airframe design, it is imperative to investigate the problems of design and manufacturing integration early on at the conceptual design stage. A new design approach and support tool is required which will aid the designer in future product development. This is a particular necessity in the current context of increasing complexity and challenging economic situations.

The present work focuses on the development of a design approach and design aids for designing metallic wingbox structures of low-subsonic transport aircraft with small wing sweepback angles. Its aims are two-fold: to assist in producing alternative structural concepts, and to capture the effects of new materials and manufacturing processes on weight and cost. It will form the basis for selecting the structural concept at the early stage of the design process. The target users of this design approach and tools are relatively inexperienced structural designers and students.

The developed process and tools are quite general in their application as they use stand-alone modules which can be employed separately or jointly with existing techniques and tools used by industry, research centres and academia.

A comparison of the result from the developed analytical tools against a detailed study undertaken by an aircraft company on the original configuration was made. It showed stress analysis and sizing results that were within a 10% margin.

A case study was performed to investigate the reduction of Direct Operating Cost (DOC) of a turboprop transport aircraft by redesigning the wingbox structure. Weight reductions of wing box structure of 16% were achieved using new configurations and advanced metallic materials. The purchase price of the aircraft could also be reduced through use of cheaper labour and new manufacturing processes. These cost savings, if converted into DOC reductions, are only 0.36% of DOC due to fuel saving and 0.25% of DOC due to manufacturing cost reduction for the wingbox structure only.

It is obvious that the overall DOC reduction is the result of the total impact of relative DOC effects due to fuel cost saving, material prices, labour rates, and manufacturing process improvements. Within the range of the

calculated parameter values, the overall DOC reductions could be as much as 0.61% relative DOC. It appears that fuel prices, material cost and labour rates give greater impacts on DOC than high speed machining processes.

Due to the use of advanced aluminium, maintenance cost is also predicted to be less. It has better fatigue life and fracture toughness than the standard aluminium and therefore will increase the aircraft maintenance periods for inspection and repair due to slower crack damage growth. This cost saving will contribute in reducing the life cycle cost of the aircraft. In addition, the number of crack stoppers could be reduced, therefore minimising weight and manufacturing cost. These benefits however have not been analysed.

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LIST OF ABBREVIATIONS

ACDA	Airframe Conceptual-Design Aid
BEP	Break Even Point
CAA	Civil Aeronautics Administration
CAD	Computer Aided Design
CS	Certification Specifications
DBT	Design Build Team
DOC	Direct Operating Cost
EASA	European Aviation Safety Agency
EC	European Commission
ESDU	Engineering Science Data Unit
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FMEA	Failure mode and Effect Analysis
FEA	Finite Element Analysis
HSM	High Speed Machining
IPD	Integrated Product Development
IPPD	Integrated Product and Process development
KBE	Knowledge Based Engineering
KBS	Knowledge Based System
MSG	Maintenance Steering Group.
NDE	Non Destructive Evaluation
PIM	Product Introduction Management
QFD	Quality Function Deployment
R&M	Reliability and Maintainability
RPM	Revolution Per Minute
TEO	Technical Element Objective

LIST OF NOMENCLATURE

AR	Aspect Ratio
A_{sk}	Skin area
A_{str}	Stringer area
$a.c.$	Aerodynamic Centre
BM	Bending Moment
b	Wing span
c	Wing chord
$\bar{c}$	Chord length of the MAC
$c_{planform}$	Chord of planform section
c_{lb}	Basic lift coefficient (due twist)
c_{la1}	Additional lift coefficient at $C_L = 1.0$
c_l	Lift coefficient on the wing section
$c_{l\text{-}actual}$	Lift coefficient actual
$c_{m\text{-}ac}$	Aerodynamic moment coefficient
$c.g.$	Centre of gravity
C_{mp}	Relative cost associated with material-process suitability (workability or fabrication)
C_c	Relative cost associated with shape complexity
C_s	Relative cost associated with achieving minimum section thickness
C_{ft}	The higher the cost in achieving a specified surface finish, C_f or tolerance, C_t but not both
C_{fuel}	Price fuel

C_{mt}	Cost of material per unit volume
FH_{YR}	Flight Hour per year
$F_{density}$	Fuel density
f_b	Allowable bending stress
h_T	Total of effective depth of spar
I	Moment of inertia
K_c	Buckling coefficient due to compression
K_s	Buckling coefficient due to shear
K_g	Gust alleviation factor
K_{FF}	Fuel fraction factor
L/D	Lift to Drag ratio
L	Lift force
MS	Margin of Safety
m	Mass of the aircraft
m	The slope of lift coefficient versus α graph
$\bar{m}$	The average slope of section lift coefficient versus α graph
MAC	Mean Aerodynamic Chord
N	Annual production quantity for the part
N	Compressive force per unit length
n	Manoeuvring load factor
P_c	Basic processing cost for an ideal part
Q	Shear flow
R_c	Cost coefficient for the part design that takes into account shape complexity, material workability, section thickness, surface finish, and tolerances.
RF	Reserve Factor

S	Gross wing area
SF	Shear Force
SFC	Specific Fuel Consumption
T	Torsion
T	Cycle time in seconds to produce an ideal part
T_{fuel}	Total mass of fuel used during the aircraft life
TO_{GW}	Take Off Weight
TC_{fuel}	Total life cycle cost of fuel per aircraft
t	Maximum thickness of wing section
t_{eff}	Effective panel thickness
t_s	Skin thickness
t/c	Thickness/chord ratio
V_C	Design cruise speed
V_D	Design diving speed
V	Vertical shear force
V	Volume of material input to the process
W	Maximum take-off weight
x	Wing axis, coinciding with root chord
Y_S	Years of Service
y	position of the airfoil section at the lateral axis
y	Coordinate of section, measured from chordline
z	Axis perpendicular to XOY-plane
α	Angle of attack
α	Cost of setting up and operating a specific process
β	Total tooling cost for an ideal part
ε	Twist angle
Λ	Angle of wing sweep

λ	Taper ratio
Δn	Load factor increment
μ_g	Non-dimensional mass coefficient
ρ	Air density
σ_{comp}	Compression stress
$(\sigma_{comp})_{cr}$	Critical compression stress
σ_Y	Yield stress of material
σ_{max}	Maximum principal stress
$\sigma_{crushing}$	Crushing stress
τ_{xy}	Shear stress
$(\tau_{xy})_{cr}$	Critical shear stress
Γ	Dihedral angle
η	Plasticity reduction factor

IMPORTANT NOTE: the nomenclature listed must follow the original sources of the equations. In this case, the reader might find the same nomenclature may have different meanings and use throughout the thesis. However, the author has provided the explanation for each nomenclature immediately after the relevant equation.

Chapter 1 Introduction

1. INTRODUCTION

1.1 Airframe Structural Design and Manufacture Requirements

Designing an airframe is a complex activity; it has to meet a list of very stringent requirements to satisfy customer, airworthiness and company objectives by synergising important aspects that are very often conflicting with each other (Cutler and Liber, 2005; Howe, 2004; Swift and Booker, 2003; Fielding, 1999). The designer must consider the method of fabrication and tooling for each individual component and its quantity for each aircraft (Cutler and Liber, 2005; Swift and Booker, 2003; Moore et al., 2006; Curran et al., 2005a; Poli, 2001; Holmberg, 2000; Boothroyd et al. 2002). Designers should keep constant communication with maintenance department and airlines so that they can see the operating environment for the part that they design (Cutler and Liber, 2005; Holmberg, 2000; Thompson, 1999; Hearn et al., 1998; Thompson, 1999; Hearn, 1999; USAF, 1987; Friend, 1992; Smith, 1985). It is not sufficient to rely on published books of requirements as large proportions of these are out of date or do not reflect the actual problem in hand: the best requirement book can only give part of the picture.

The decisions made during the conceptual stage are of great importance in determining manufacturing cost, for example. They define the fundamental architecture of the product and how it will be manufactured, including its basic form and configuration, the material and processes to be used, technologies, key suppliers, and critical specification. In the later stages of the project, the decisions made then represent a sort of filling in of the details around the fundamental decisions made earlier. The final manufacturing cost is determined, to a large degree, very early in a project's life (Swift and Booker, 2003; Boothroyd et al., 2002; Jupp and Scott, 1998; Institution of Mechanical Engineers. Aerospace Industries Division, 1997; Magrab, 1997).

1.2 Structural Conceptual Design Stage Consideration

1.2.1 Assessment of Possible Innovations

There are three important stages in the conceptual design process: concept synthesis, concepts selection and parametric synthesis; in which the designer is required to explore the concepts by utilising both the information available and existing experience, as well as the possibility of new concepts utilising the latest technologies to improve competitiveness. The cooperation with suppliers of materials such as composite and advanced metallic materials will open up new possibilities and speed up the introduction of improved products. This was demonstrated with the A380 product development, when composite technology replaced metallic materials in certain areas and was more effectively and economically applied. At the same time though, metallic material suppliers were also developing enhanced materials and new processes specifically for the aircraft and so were able to maintain a significant share in the aircraft.

The main issue in commercial aircraft manufacturing is affordability, or cost effectiveness, of the product. Not only has the performance to be better than the competitor, but the aircraft must also have lower direct operating costs. This can be achieved during conceptual stage if decisions on concepts and associated technologies are effectively made.

The modern-day designer not only has the ability to design the structure but also assess the feasibility of new technology and together calculate its risk. In other words, the designer now has more responsibility than ever, but with less time and budget (Niu, 1999).

With the rapid development of composite and metallic technology, the information on the advantages and problems of specific designs could change during the product development timescale. For example, the use of an autoclave on metallic material could allow the use of integrated wing panels without exfoliation corrosion problems associated with integral machining. The better and cheaper composite materials and manufacturing now allow the use of composite materials for major structures with little cost penalty. It also gives the additional benefit of possible use of composite materials, such as the insertion of sensors into the wing structure that would allow continuous monitoring of the structure integrity and which would therefore change and ease the maintenance tasks. The problems associated with maintenance cost issues of composite material are now less. This total information could make a significant change on how the airframe is designed and manufactured.

All these possible advantages should be assessed for cost-effectiveness during the conceptual design stage. Lack of the assessment methods for this new technology during the conceptual stage, and trying to insert them during later stage, could risk company profits and market loss. As designers, the involvement of multidisciplinary teams from the company and suppliers could help in the analysis of new technology during the design stages. Therefore, the designer and team need to gather all the necessary information and to make design decisions based on that.

The use of structured requirements would help in defining the relationship between the requirements of economy and safety and product parameters. The designer would then be able to systematically assess the importance of the technology and the associated risks and maintain the relationship throughout the product development stages.

It could be safely said that technology is evolving, driven by customer requirements for safer, better and cheaper products. Designers have to design the airframe based on engineering experience; at times utilising a completely new concept. For each aspect of design, designers should know the advantages and also the limitations of the concepts. It would then allow them to incorporate or seek new technologies that could be used during the product development stage. The initial concept is used as the datum and then compared to the concept with new configuration, materials and processes.

The availability of technologies for new designs is not always obvious; hence, assessing new technology against product requirements would help identify the potential improvements and advanced technologies to employ. Again, by bringing the supplier and operator into the design team would help the designer in deciding the improvements to be made.

1.2.2 Conceptual Design Synthesis and Analysis Methods

Conceptual design requires a combination of designers and engineers' expertise in synthesis and analysis to produce an optimum design. Synthesis can be divided into concept synthesis and parameter synthesis. **Concept synthesis** is an exercise whereby the designer creates concepts that meet the requirements while remaining focused on the relevant technology. The result is the airframe configuration, material type and fabrication processes selection, and in-service maintenance requirements. The concepts are then down-selected to a smaller number, say one to three depending on the product complexity. For example, one concept could be chosen for a wing box, and several concepts for ribs and skin-stringer panels, for further detailed assessment. The selection process is a critical aspect, as too often the designer becomes focused on a single idea early in a design process and the evaluation exercise is undertaken to justify the preferences of the designer. After the concept synthesis is settled, the purpose of **parameter synthesis** decides the optimum level of individual concept parameters to avoid, for example, over-sensitivity of the concept to variation in fabrication tolerances and handling.

The purpose of analysis in conceptual design is to predict the characteristics of the given concepts more accurately, for example in the areas of static and dynamic loads, configuration, mechanisms of moving structures, reliability problems, and damage tolerance. The results of these analyses feedback to the designer to enable him to make the necessary refinements or modifications to his design.

1.3 Rationales and Research Targets

Based on the literature survey and the author's experience, there are several aspects which drive this research. Firstly, the competitive nature of aircraft industries these days demands new design methods. Secondly, is the need to achieve customer satisfaction for the entire aircraft product. In addition to these, the research is also affected by concerns about inefficient design environments for students and inexperienced design engineers.

1.3.1 Competition in the Aircraft Industry

These days, aircraft companies face the most challenging era, where competition between them is unusually intense. The mergers of several aircraft companies in the USA and Europe, such as Boeing Company and McDonnell Douglas, and the continuing projects between European aircraft industries on their Airbus products, illustrate this.

Any company's most basic goals are to increase revenues, decrease costs and produce new products rapidly. With regard to increasing revenues, companies must offer their customers more customised design configurations to suit their specific needs. For each of these configurations, multiple design iterations must be performed to produce optimised designs with improved and consistent quality. With regard to decreasing costs, the companies' development teams should plan projects to reduce the likelihood of midcourse corrections, the most devastating source of costs in most development projects. With regard to producing a new product rapidly, the companies should respond rapidly to changes in the market environment. A customer's buying decision is based not only on the product, but also its delivery date.

1.3.2 Customer Satisfaction as the Primary Objective

Whatever term one uses – distinctive difference, product value, added value, differentiation – the buyer's choice and the user's preference for one product rather than another are determined by the way design and the early stages of product development are managed. Of the modern, "borderless world", Kenichi Ohmae comments: "To develop economically, you must find ways to add value. To do that you must understand [your] customer well enough to figure out how best to differentiate your products." (Ohmae, 1990).

Robustness of good quality product is characterised by products that work well – close to ideal customer satisfaction – even when produced in real factories and used by real customers under real conditions of use. All products look good when they are precisely made in a model shop and are tested under carefully-controlled laboratory conditions. Only robust products provide consistent customer satisfaction. Robustness also greatly shortens the development time by eliminating much of the rework.

Aircraft products give further insight into robustness. The customers – airlines – do not want airplanes that require maintenance checks too often. They want a product that is robust against key operating conditions. For instance, the airlines would expect that in the first 5 years, their new airplanes would be free from heavy maintenance, such as fuel leak and corrosion. To reflect these issues, the production process has to be robust to produce less variation in product quality.

1.3.3 Work Experience and Design Environment for Students and Inexperienced Design Engineers

In addition to the previously highlighted issues, there is also a concern about the weakness of design environments encountered by students and inexperienced design engineers in the structural design process; a situation resulting mainly from these engineers' tendency to spend much of their time in front of computer screens, inputting data into computers and digesting the output and overlooking the importance of team work and communication with colleagues from other disciplines. Engineers are losing the opportunity to gain much needed experience from their senior, more experienced engineers. The end result is that industries gradually lose their most valuable resources – experience which will disappear along with the retiring engineers (Niu, 1999; Scott, 2000). This author believes the problem to be vital for the future competitiveness of the industry, and without remedy, both companies and customers will lose.

To overcome these issues, this thesis strongly asserts the need for the development of design methodology and tools to facilitate the structural design process.

1.4 The Approach and Research Contribution

1.4.1 Major Issues

Based on the above discussion, three major requirements can be identified as necessary to enhance the conceptual design process and to produce improved concepts. Firstly, a structured process is essential to allow the interaction between design and manufacture to be assessed effectively during conceptual design. Secondly, a design-support tool is required to reduce the time to produce the concepts. Finally, methods need to be developed to guide students and inexperienced engineers in the design process.

The approach is developed by assessing the current methods used and/or those developed by research institutions and industries. A generic approach is developed, based on the results of this assessment. Best practices from industries and research institutes were collected and used during the development of design tools.

For airframe analysis, there are many well-established and efficient tools to perform detailed calculations on the areas, such as static and dynamic analysis of airframes, reliability, damage-tolerance, mechanisms, tooling and fabrication simulation. Aircraft manufacturers have been extensively integrating the analysis softwares to create a seamless analysis tool to reduce the development cost and time to market (Jupp and Scott, 1998).

The tools to support synthesis activities are much less developed, compared to analysis tools. This is due to the complexity of the problem of modelling and the ways in which designers make judgements. The solution is normally context-sensitive and the problems are ill-structured; computers cannot therefore easily model and simulate them (Brezillon and Pomerol, 1997). To solve this problem is to use reasoning, decision-making on the basis of special domain knowledge, and experienced designers' insights.

Some designers have experience of manufacturing and assembly, and understand the manufacturing capability and limitations they must work within. However, there are many designers who do not have that experience and specifying the design that is too difficult and too costly to manufacture and assemble. In addition, due to economic pressures and reduced product demand of the last few years, many aircraft companies have made big changes in their organisational structures, and this has resulted in the downsizing of the workforce (Hayward and Royal United Services Institute for Defence Studies, 1994; Hayward, 2006; Weston, 2001; Weston 2000). In other words, work is completed by fewer people with possibly less experience. The chance of acquiring advice from company specialists is therefore often limited due to their small number and busy schedules, whilst the cost of inviting outside consultants might not be justified for a longer period of work.

The combination of the above puts additional burdens on an already difficult environment for the designer to explore more innovative concepts during the conceptual design stage. With relatively fewer design options uncovered, the analytical software might only analyse and eventually optimise suboptimal-concepts.

The previous discussion clearly establishes that there is a need for research to be driven by the need to develop a structured and intuitive approach for concept synthesis and easily accessible critical information to support creative thinking and the decision making process. The following section explains the approach and significant aspects of this thesis's research.

1.4.2 Approach Outline

The airframe conceptual design approach will be developed for synthesis activities. These will combine structured and ill-structured problem-solving methods and supported by existing design tools. The stages are shown in figure 1-1.

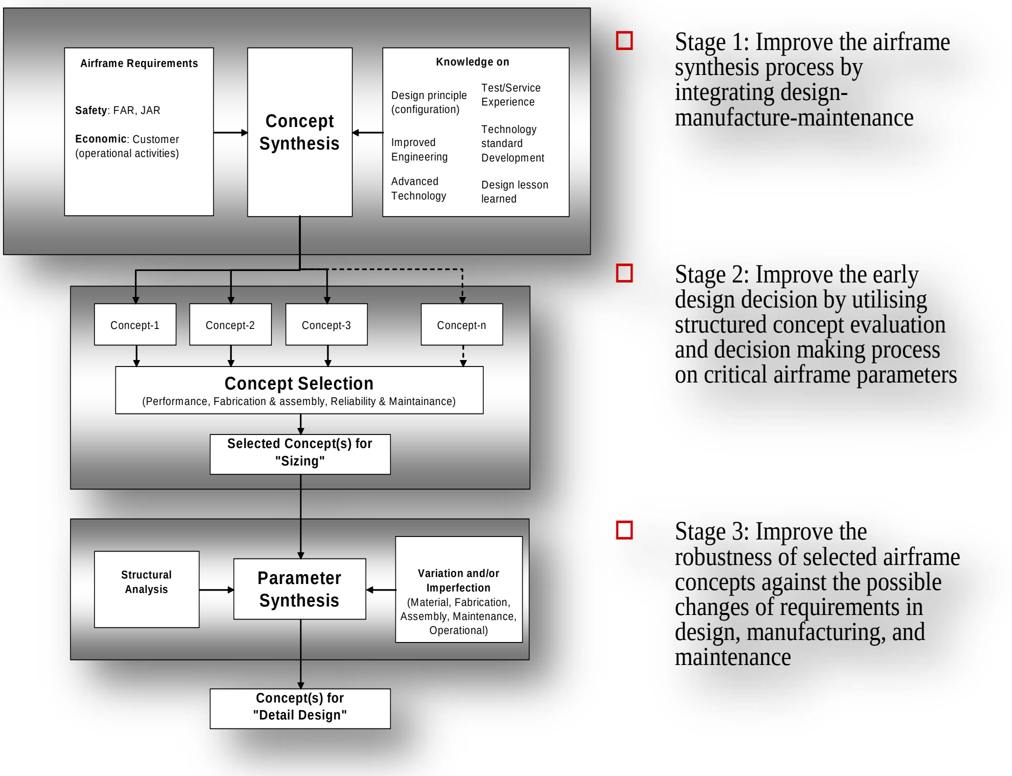


FIGURE 1-1 AIRFRAME DESIGN APPROACH

Stage 1 is where the synthesis activities and concept explorations are to be performed. It will consist of three major elements: requirements, synthesis, and product knowledge.

The requirement block in stage 1 should contain the procedure as to how the design specification is created, and consistently maintained throughout the product development process and product breakdown. The lessons learned from past projects are needed in this activity.

Critical information required to support the synthesis activities should be structured, based on major parameters such as configuration, fabrication and assembly, and in-service qualities, to ensure a comprehensive investigation of certain concepts. The developed Airframe Conceptual-Design Aid should help to reduce the time and effort normally spent to acquire this information. By having this structured information available (Emberey et al., 2007; Van Tooren et al., 2005; Booker et al., 2001; Nevins et al., 1989; Corbett et al., 1991) the designer can interactively question his/her design and make possible improvements during the 'synthesis' activity. This information will help the designers to more effectively explore their creative thinking in complex product modelling and decision making.

In stage 2, the airframe concepts are then to be assessed on product weight and manufacturing cost. These major parameters are the most technically and economically representative of the product that will satisfy customer requirements. Although the rating result will not necessarily represent the concept chosen, it will show the strong features and weaknesses of certain concepts. The objective is to allow the designer to make a decision to carry-on with the concept with the highest rating, or to improve other concepts based on certain strong features. The selection technique is to be flexible and meaningful, to support the designer in selecting concepts. More importantly, it should give more responsibility and control to the designer, to satisfy the main requirements.

In stage 3, the method should provide a quick and effective technique to perform parametric studies. Airframe analysis tools may be used as the basis of this method. Some modifications based on the lessons learned from designers and structural specialists should be presented.

A new supporting airframe design database needs to be developed. This will provide information on test and service experience, technology standard development, and implementation of design lessons in the following areas:

- critical issue of structural arrangements;
- characteristics of materials;
- methods of fabrication and production cost;
- in-support service requirements.

The information is to be collected from established literature, such as journals or working group papers, and combined with material from visits and discussion with experts in industry and academia.

The database is to be developed electronically and to be accessible on the internet/intranet so that any necessary information not normally available to the designers without extensive surveys, will be only a few clicks away. This database could potentially be useful to record and retain as much expert knowledge as possible. A screen shot of the proposed tool is shown in figure 1-2. The most challenging task during the tool development is to avoid overwhelming the user with information, but to be adaptable for different product development stages. The result should be an intuitive design tool that can be used to assist the designer in a variety of complex decision making problems during conceptual design.

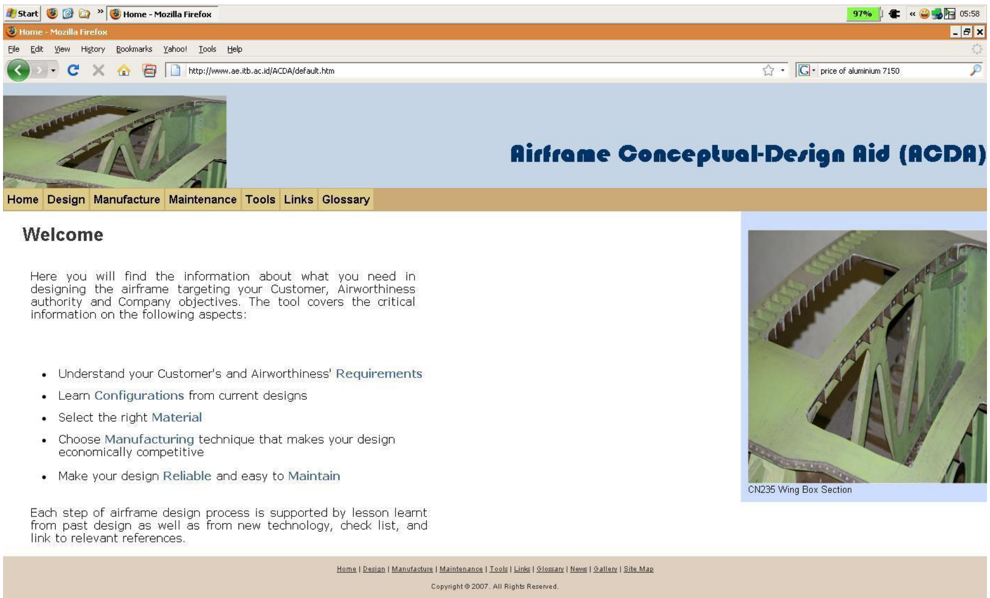


FIGURE 1-2 SCREEN-SHOTS OF THE AIRFRAME CONCEPTUAL DESIGN AID (ACDA)

Finally, programs will be developed to undertake analysis of the initial sizing of a wing box, weight calculations, and cost estimation. The programs will perform parametric studies on many critical parameters of wing boxes and will be used to create more than 3000 cases to achieve minimum weight and more than 2000 cases for minimum manufacturing cost. The process should be very straight forward and be able to run as many times as the designer wants. As an example, the above number of cases should be run in about 3 hours on a PC.

1.5 The Road Map of the Thesis

The remaining chapters of the thesis are structured as follows:

Chapter 2 - Airframe design process and techniques. It will discuss the critical issues of airframe design from different views: construction, fabrication, and maintenance. This chapter also evaluates the merits and problems of techniques to support designers in the conceptual stage. This provides a firm foundation for discussions in the following chapters.

Chapter 3 - Approach and design tool development is described, built upon the best practices of airframe design processes.

Chapter 4 – The case study contains the design process of a wing box structure utilising the new process and tool. A baseline wing box was redesigned to incorporate new technologies in advanced materials and processes. Weight savings due to the new materials and cost-savings resulting from improved manufacturing process are shown and compared.

Chapter 5 – Discussion of the case study and its contribution to knowledge.

Chapter 6 - Conclusions and recommendations for future work.

Appendices:

A. QFD – Translating customer requirements into engineering analysis
B. ACDA – Web based tool for supporting airframe design process
C. Wing Load Analysis – Detailed analysis of wing load distribution
D. Initial Sizing – Detailed analysis of initial sizing and failure modes
E. The support cost data
F. FEA – Finite Element Analysis
G. Example of use of the system in a tutorial context

Chapter 2 The Current Airframe Design Process

2. THE CURRENT AIRFRAME DESIGN PROCESS

This section describes the current process of airframe design with regard to requirements definition, structure layout synthesis, and manufacturing process. The interaction of these aspects to produce optimum design is shown. A list of items to be achieved during the development of the approach and tool to improve the process is then compiled.

2.1 Introduction

2.1.1 The importance of manufacturing assessment in the design process

The cost drivers on a new aircraft development have changed the design and manufacturing relationship. The design objective used to be performance, which is often translated to minimum weight. However, experience has shown that this is not the only main factor as the cost of high performance material, which is expensive and difficult to be fabricated and assembled, could be more important than weight saving. Experience also shows that the product simplicity, reduced part numbers, redesign fabrication and assembly sequences, and 'parts availability/off the shelf material' will be the main parameters in reducing the manufacturing cost (Swift and Booker, 2003; Poli, 2001; Boothroyd et al. 2002).

Figure 2-1 (Swift and Booker, 2003) shows the effect of several stages in committing the cost. The production stage has little effect on cost saving but on the other hand, any changes during the conceptual stage significantly affect cost. Addressing manufacturing issues early in the design to make the concept producible will bring the risks down and eliminate unnecessary rework during later stages.

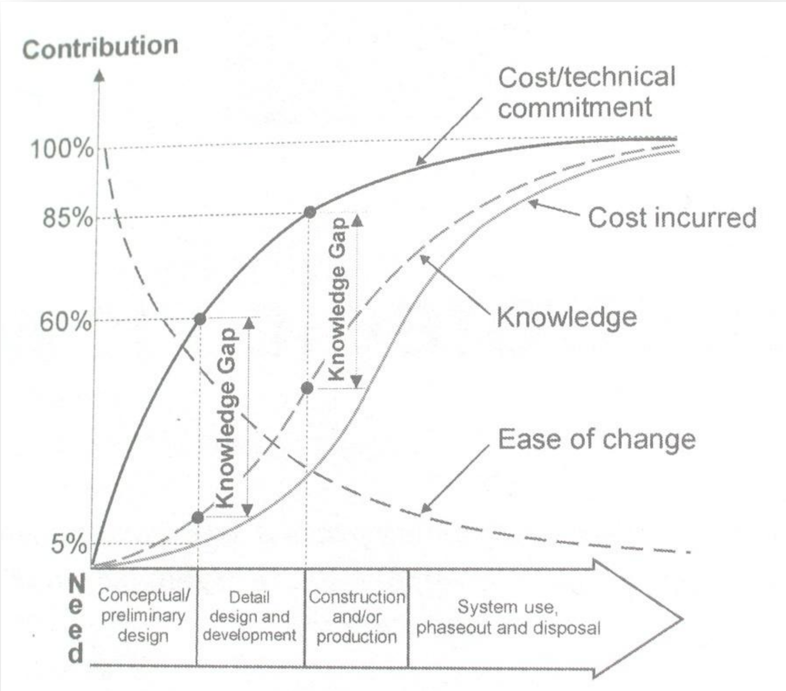


FIGURE 2-1 DESIGN COST COMMITTED AND 'KNOWLEDGE GAP' DURING PRODUCT DEVELOPMENT STAGES
(SWIFT AND BOOKER, 2003)

A good starting point for cost reduction is to provide possible alternatives at the design stage. It is often impossible to determine the best alternative without careful analysis of the probable manufacturing costs. Designing for function, interchangeability, quality, and economy requires a careful study of product quantity, production rate, tolerances, surfaces, finishes, processes, materials, and equipment (Swift and Booker, 2003; Boothroyd et al. 2002; Corbett et al., 1991; Bakerjian, 1992).

The manufacturing cost is also significantly affected by scrap waste, overhead work, rework, etc (Swift and Booker, 2003; Boothroyd et al. 2002; Corbett et al., 1991; Bakerjian, 1992). Boeing Company (Breuhaus et al., 1996) reported that the assembly problem, such as fitting tolerance, etc, was one of the major problems driving the increase in cost and delay delivery.

Manufacturing problems can be reduced or eliminated by considering the manufacturing and assembly aspects during the conceptual stage. The selection of materials for example, would dictate the type or manufacturing process, and tolerances related to it. The size and complexity of a product would dictate the manufacturing process and assembly activity required. If the above problems could be understood at the very early stage, and with the input from best practices in manufacturing and lessons learned, then the problems could be avoided at minimum cost.

The understanding of company capability and the available technology and suppliers would broaden the understanding and also reduce the risk of producing concept(s) that are difficult to manufacture or be supplied.

In addition to the above aspect, the product is also designed to be able to be modified cost-effectively to fulfil future requirements, such as a product family. The design team would need to consider whether a change in configuration to accommodate the change in requirements could be produced cost-effectively using the available tools and jig; then the product family could be maintained without the need for additional investment for maintenance.

2.1.2 Concurrent Development of Product and Processes

To stay competitive in a fierce market environment, the aircraft manufacturer has to produce high quality products or services at the right prices and at the right time. The manufacturer has to achieve maximum efficiency and effectiveness in their design, utilising their knowledge and process capability but also bringing the suppliers of new materials or manufacturing processes at the very early stage of design process to secure their expertise on specific technology to improve the final product. This current practise is quite different to the past, where the supplier was given the task of supplying the product after the concept was released from the design office. The product development design process is critical to this concurrent and partnership-based approval.

Traditionally, product development has been viewed as an organisational activity which begins with the design, followed by manufacturing. The process is shown in the following figure:

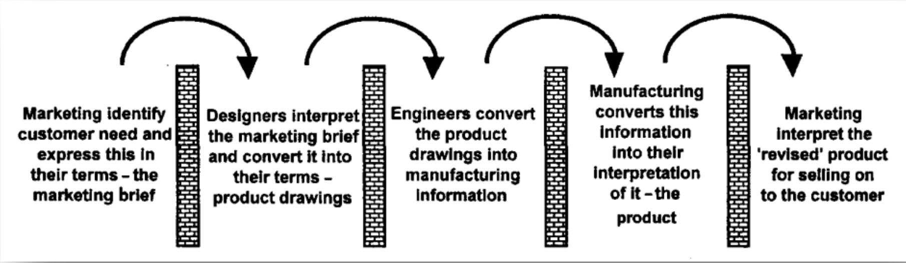


FIGURE 2-2 TRADITIONAL 'OVER THE WALL' DESIGN PROCESS (BOOKER ET AL., 2001)

The sequential process has shown the problems of long lead time and costly design changes due to the manufacturing problems found in much later stages. With the increase of competition in the market, the manufacturer has to overcome the above problems. The sequential design process is being replaced by concurrent process with the introduction of an integrated team of all the parties involved, from the conceptual stage to the manufacturing and assembly stage. Design and manufacturing communication should be more interactive, and at the same time design manufacturing interfaces should be established at more appropriate points. The manufacturing team should have greater involvement in the design and early commitment of the product production. Such a process is shown the following figure:

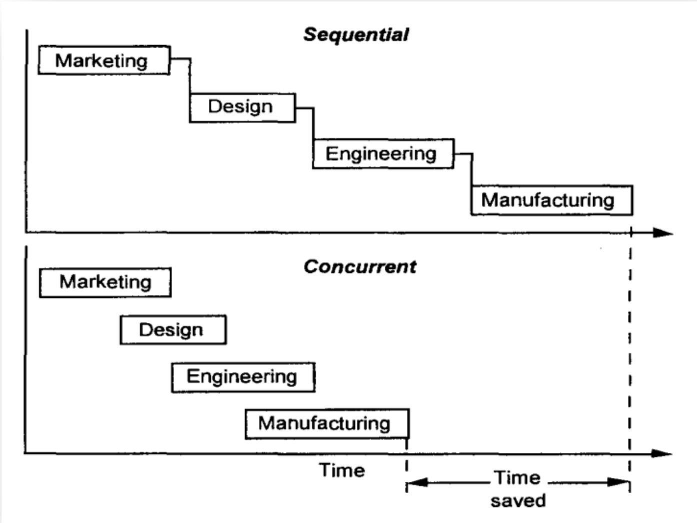


FIGURE 2-3 CONCURRENT DESIGN PROCESS (BOOKER ET AL., 2001)

One example of concurrent engineering in an aircraft manufacturer can be seen below:

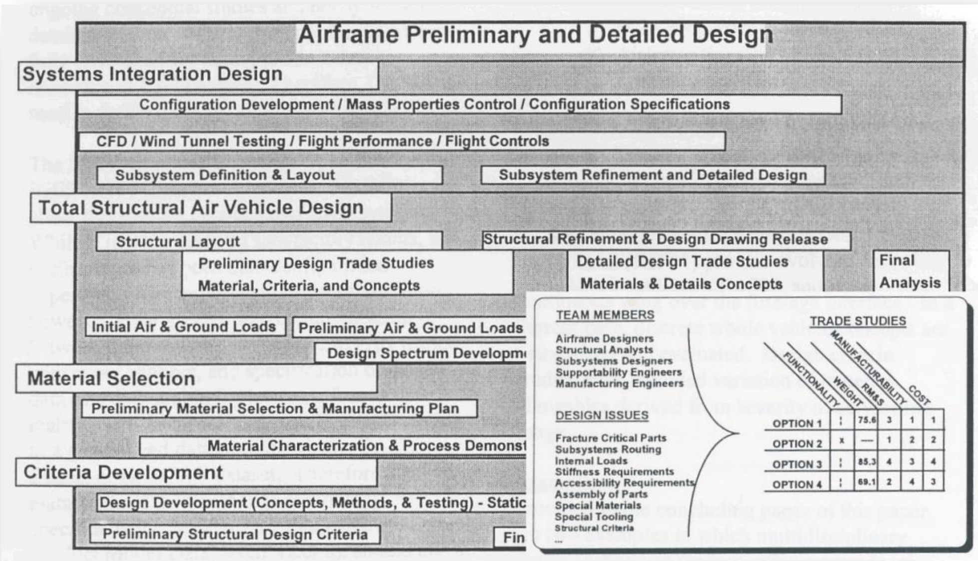


FIGURE 2-4 A CONCURRENT ENGINEERING APPLICATION IN AIRCRAFT DESIGN (LOVE, 1996)

In addition, the concurrent engineering process gives the project manager more confidence and control to review the progress of a design project and to allow decision gates to be inserted at appropriate points. Some of the benefits claimed by the manufacturer that has adopted the concurrent design process are:

- Reduced time to market
- Reduced process elapsed time
- Achieve more adherence to schedule time, quality and cost.

It is now a common approach in the aircraft industry to work on downstream design activities, such as tooling, fabrication and assembly, in-service support, etc, in parallel with the up-front design activities in order to minimise unnecessary iterations and midcourse corrections (Swift and Booker, 2003; Boothroyd et al. 2002; Jupp and Scott, 1998; Kessler, 1990). As discussed earlier, this experience shows that decisions made at the conceptual stage will give the greatest impact on the success of the product.

Manufacturers are making dramatic changes to the way in which products are brought to market, developing their own new product development processes which are employed and supported on site (Booker et al., 2001). Some of the industrial processes available in the literature are:

Boeing's Concurrent Product Definition Process using Design Build Team (DBT) (Breuhaus et al., 1996).

The process was adopted to address the schedule and organisation issues that drove change, error, and rework during the development of the B-777 aircraft. Teams were formed around the various airplane components and systems. These component DBTs ranged in size from about 15 to 60 collocated members and were co-led by Design and Manufacturing Engineering. The strategy of concurrent product definition was to recognise design integration in addition to build requirement when developing the design schedule.

Lucas Industry (now TRW) Product Introduction Management (PIM) process (Booker et al., 2001).

The objectives were to reduce the time to market, and reduce product and project costs. The generic process was characterised by five phases and nine reviews from opportunity evaluation phase to manufacturing support phase. Lucas found that PIM requires the collaborative use of team work, concurrent engineering, project management, and tools and techniques.

British Aerospace Systems' Integrated Product Development (IPD) Process (Jupp and Scott, 1998; Jupp, 1998).

IPD is defined as a combination of four principles: organisation, people, information technology, and process. The IPD process provides a structured and phased approach to product development which can be applied across the product lifecycle, from concept to decommission. In essence it provides a basic framework to apply decision gates, with clear input and output criteria to aid decision making. The objectives of adopting the process are to reduce process cost by 30%, to reduce process elapsed times by 50%, and to achieve 100% adherence to schedule, quality, and cost.

The project was built around teams of people from different disciplines, working together, either collocated or distributed, on the same part, component, or system of the aircraft. The team were required to have skills, knowledge, and culture to work in the new organisation.

The team had a total design process agreed to at the beginning of the project. For a consortium of several major manufacturers, each manufacturer was able to still use their own design process but they had to ensure phase gates where all the processes could be agreed upon at the input and output – at the beginning and at the end – of the gate.

The process was supported by tools and technologies in the data creation, storage, management, and product modelling and analysis. The tools and technologies supported the teams to work on downstream design activities, such as tooling, fabrication and assembly, in-service support, in parallel with the upfront design activities in order to minimise unnecessary iterations and midcourse corrections.

The similarities of the above processes are in the use of collaborative team, simultaneous (or concurrent) engineering during product development, clear project management, and the use of concurrent engineering tools to support the team activities.

2.1.3 Technology Innovation

In the current market environment, the relationships between improving product through the use of technological innovation and the relationship with suppliers is even more crucial. For example, the recent experience in the development of new aircraft suggested that the role of supplier in developing new material and related processes has been dominant. It could be understood that the aircraft manufacturer is becoming more of system integrator in which great amounts of product work-share are done by partners and suppliers. Economically this approach is very relevant as the manufacturer could reduce the development cost for new technology, and at the same time other companies' advanced technologies could be incorporated into the new aircraft. In addition, for some multinational projects, the involvement of other companies or suppliers in the project could increase the chance of the new aircraft being sold in greater quantities in the partner's country.

The following tasks should be considered in order to achieve some technology innovation in conceptual design (Dieter, 2000):

- To be well-informed with outside information

- To possess a coherent, though not necessarily detailed, picture of what needs to be done
- To challenge ideas
- To be able to work with minimum information and to communicate with people from different backgrounds
- To find ways to access new information and resources

In addition to these tasks, as a guideline, the following measures are taken to determine if the technology is mature enough to be used (Magrab, 1997).

- Can the new product be manufactured with known processes?
- Are the critical parameters that control new technologies functions identified?
- Are the safe operating ranges of the technology's parameters known?
- Have the failure modes been identified and evaluated?
- Has the technology's life cycle been evaluated and are its environmental affects known?

2.2 Airframe Requirements

Airframe requirements consist of two major elements: safety and economics. Safety is established by the Federal Aviation Administration (FAA) in Federal Aviation Regulation (FAR) or by the European Aviation Safety Agency (EASA) in Certification Specifications (CS). The manufacturer has to meet the requirements laid down by these institutions in order to be able sell their aircraft to market. The airworthiness requirements cover elements from the structural design process to the operation of the aircraft. They set the rules in the process of defining and analysing the flight envelope where the aircraft will operate, the loads acting on aircraft components, the fatigue and damage tolerance analysis of structure elements. Airworthiness also sets the requirements for the protection of aircraft material from problems due to external environments such corrosion and lightning. It ensures that the choice of materials and fabrication techniques are either based on a proven track record of applications or on a full test. The consistency of technique or method in designing is also required to be shown during certification process.

The economic requirement is defined from the market or customer requirements based on extensive market study of historical data and specific aircraft trend. It is a common practise to involve airlines in the process of designing aircraft from a very early stage. The technique used in requirement definition should offer a systematic approach in order to ensure the customer requirement is consistently maintained throughout the product design cycle.

The process of establishing a correct set of requirements is therefore placed highly in the design activity. Both safety and economic requirements should be assessed by the designer throughout the design cycle and maintained for different levels of aircraft product tree.

For the last 50 years of commercial aircraft history, the concept generated and selected is driven by different factors, including technology, cost, and weight, and each of these drivers attract a distinct characteristic to the concept.

Performance is one of the competing factors during the selection of the aircraft. The aircraft performance is measured by such factors as speed, fuel efficiency, mission capability and flexibility, operational safety, and readiness. To achieve the performance superiority of the aircraft, the designer should allow innovative concept and advanced technology to be incorporated into the new aircraft. However, in the current highly competitive market, the excellent performance should also be supported by low direct operating cost (DOC) of the aircraft.

Minimum Weight is probably the most common driver in airframe design process. Weight is essential for the success of an aircraft and this is not just achieved by optimising the main load carrying structure but also by careful attention to detail. The following note shows clearly what the above paragraph means (Niu, 1999):

> *"Let's consider an aluminium alloy skin and skin stringer panel with a compression loading of 21000 lb/in and shear loading 3400 lb/in. The compression optimisation analysis yielded the following dimension:*
>
> *stringer spacing b = 2.7 in, skin thickness t_s = 0.15 in, and effective thickness t_e = 0.376 in. The combined margin of safety was a negative 10%.*

In a subsequent optimisation analysis, the stringer pitch was doubled b = 5.4 in. The resulting skin thickness was 0.22 and the effective thickness obtained was 0.396 in, representing a 5% weight increase. The margin of safety was 1%.

The increase in stringer pitch has the additional beneficial effect on the so called non-optimum weight; for this case it permits the elimination of 40000 fasteners and 1000 stringer-to-rib clips, and a reduction in the amount of material for tank sealing. The combined effect of the increase in stringer pitch amounted to a 400 lb weight reduction and a cost reduction. The increase in skin thickness will also yield increases in torsional stiffness and fatigue life due to reduced stress levels."

The use of light material and associated manufacturing processes are required to achieve the weight target. The development of advanced aluminium alloy, composite, and titanium alloy with super plastic diffusion bonding are some of the results of this driver.

Easy to Manufacture and Assembly is increasingly important as airframe configuration is becoming more complex and the labour cost to manufacture and assembly is rising higher. The utilisation of the best manufacturing and assembly technique is vital to produce a high quality product at competitive prices in the least possible time.

Minimum maintenance is critical in order to minimise DOC of the aircraft. Aircraft shall be designed for rapid and easy maintenance and for despatch with in-operative equipment or with configuration deviations, so as to achieve low maintenance cost, high utilisation, and despatch reliability. When applying new materials, such as, but not limited to, composites, the economics of minimum maintenance should be considered. Before these new materials are used, acceptable methods for inspection and repair by the operator in case of local damage shall have been developed and demonstrated.

There are two major considerations regarding the process of requirement assessment:

- to get the correct information on requirements
- to maintain the requirement throughout the design stages and product level.

It is very easy to be over-ambitious with regard to the product or to indulge technological advances without maintaining the cost and time to market. Well-documented in aircraft history are the projects cancelled due to lack of thoroughness or wrong prediction or political involvement that resulted in millions of pounds going to waste, not including the loss of markets to competitors from other countries (Fielding, 1999; Wood, 1975).

It is also relatively easy to deviate from the initial requirement due to the difficulties of maintaining it in different design stages. For example, if a design is too difficult to manufacture, it would require a costly iteration or modifying design to ease the fabrication, but with the penalty of not meeting the target requirement. Additionally, the complexity of the design team structure and the various company standards could cause the requirements not to be communicated properly. Consequently, the challenges or changes at any level which are supposed to improve the final product could be ignored or forcibly altered for the sake of achieving the target schedule at minimum cost.

2.2.1 Relating the Customer Requirements into Airframe Design Process

The manufacturers have to listen to their customers to deliver a product of high value to the customers. The current economic health of the airline industry and future market analysis will dictate the criteria for new aircraft.

It should be borne in mind that the airlines will have further or different requirements when selecting aircraft for their fleets. To facilitate this, the manufacturer should present the trade-off analysis whenever departure from the basic requirements is necessary, despite the potential advantages in weight, performance, or DOC. Therefore by taking into consideration specific characteristics of the design, such as the size of aircraft, number of engines, engine power, growth potential, range, and family concept, the process of selecting the layout of wing box or fuselage structure should consider these extended requirements.

One method for capturing customer requirements is Quality Function Deployment (QFD). QFD is a structured method to capture customer requirements, prioritising the needs, and identifying solutions to meet those needs. The method has been successfully implemented in the automotive industry, and has a great potential to be applied in aircraft business.

The aircraft is normally broken down into a smaller sub-system, a sub-sub system and components. Each work is performed by teams who should work concurrently to ensure the coherence and integration of the overall product. The consistency of main requirement is maintained by dispersing the requirement to all levels of work, i.e. from overall aircraft, wing, and spar to a much smaller component such as stringer.

Therefore, to generate airframe concepts, the design team breaks the function requirement down and translates it into a physical design parameter. Each physical design parameter then forms the product breakdown tree in which, for each component in the tree, the above aspects should be incorporated to achieve the target.

2.2.2 Meeting the Airworthiness Requirements

Safety, of both the airplane's passengers and the people living under the aircraft's flight path, is the priority of airworthiness requirements and the first requirement during the product development process. Airlines, with their increasing passenger numbers and fleet size, obviously continually aspire to significantly reducing accident rates. New aircraft entering the market might incorporate new structural concepts, materials, and manufacturing processes to achieve lighter and more economical structures, but safety dictates the use of proven technology or the feasible application of new technology within the product development time frame. Similarly, the application of new material and manufacturing processes are dictated by the maturity of the material and the process to allow the product to withstand the operational environment without unexpected failure.

2.2.3 Functional Requirements

Airframe construction is driven by the functional requirements given to the structural designer. The following factors describe a list of common requirements imposed upon the design of wing structure. For unconventional aircraft or aircraft types not in this study, the functional requirement will follow accordingly:

Aerodynamic: The wing structure must possess stiffness to maintain an aerodynamic profile during flight and in which aerodynamic characteristic has not changed otherwise load redistribution analysis must be done.

Fuel: Integral tanks are common in commercial aircraft, and this dictates a wing leak-proof fuel tank. The designer should provide an access panel on the wingbox to enable inspection and resealing of the tank. The lower access panel is located in a primarily tension skin area; the designer must also note, therefore, that the panel will introduce stress concentration in an area where crack propagation is a major consideration.

Control surfaces: These include flaps, aileron, and spoiler. The control surfaces and support structure govern the front and rear spar positions and also the rib position in conjunction with the optimum rib pitch for cover design.

Leading edge: Bird strike and de-icing requirement will dictate the type of structure required for the forward part of the wing and its associated moving nose or slat.

Landing Gear: Wing mounted landing gear will give inertia load but will transmit ground load to the wing. It may also interfere with the wing structure.

Engine: The engine will give inertia relief during flight but give ground load on the ground. Engine placement on the wing will transmit load to the wing structure.

2.2.4 Structural Requirements

Strength and stiffness: Before attempting any optimisation of the structure for strength, the stiffness requirements should be investigated. These are likely to determine the initial skin thickness. The most important stiffness requirement is probably that concerned with wing torsional stiffness, i.e. to account for flutter, aileron reversal, and structural divergence.

Safe life: The safe life design principle requires that the airframe can support the repeated loads during its life without any detectable cracks. An appropriate safe life scatter factor must be applied in the analysis or test to take into account the appropriate scatter factor in the fatigue behaviour of metallic materials (EASA, 2003).

Damage tolerance (fail-safe): This requirement requires that the airframe must be able to withstand load at various modes of damage on the structure due to fatigue, accidental damage, and corrosion at probable locations (EASA, 2003). The damage tolerance design principle comprises two categories: 'single load path' and 'multiple load path' structure. Acceptable residual strength of the damage structure is in the order of limit load as defined in CS 25.571.

Minimum weight: This requirement is an especially important parameter in the airframe design process. One hundred kilograms of weight saving can have a significant effect on the operating costs of the aircraft (as shown in the case study).

Minimum cost: It comprises development cost and operational cost of the aircraft, in which the aircraft manufacturers have to translate it into many derived requirements, such as weight, fuel consumption, manufacturing cost, and maintenance cost.

2.3 Airframe Configuration

Airframe conceptual design is an iterative process in which the designers generate and work on different concepts based on input from other factors, including configuration, loads, weight and balance. Analysis process is quite straightforward, but synthesis requires a setting down of initial assumption to create a structure with a given space and mass target.

Often these initial thoughts are a direct result of the previous experience supported by a selection of statistical data on similar aircrafts. The more skilled the designer, the quicker will this iterative process converge. Thus past experience or knowledge of similar types plays a very large part in the conceptual design phase.

The structural designer and analyst therefore have the responsibility and capacity to significantly affect the efficiency of airframe structure. Structure is the most easily identified weight of an airplane. For Boeing aircraft, the airframe is about 55 to 60 percent of the empty weight (Brehauss, 1996).

Some of the practical aspects that should be considered during airframe design process are:

Firstly, to ensure the design conditions are correct and logical. The designer should then try to eliminate or reduce those loads that predominate.

Secondly, to select efficient materials and type of construction. The designer should employ structural indices and trade studies to be sure of the efficiency of the selected design. Airworthiness regulation and the airline's economic consideration would dictate the use of safe-life or damage tolerant philosophy for different parts of the structure. Rather than be satisfied with a previous successful implementation on an aircraft, the designer should always strive to stimulate creative ideas towards improving structural concepts..

Finally, and most importantly, to use common sense; in other words, to support engineering facts and diplomatic persuasion, and where necessary, to change someone else's structure to ensure simple, straight, short load paths. The construction has to transmit and resist the external and internal load in the most efficient way. The load path is thus made as short as possible to minimise weight and to increase simplicity.

For almost any type of product, the design process is normally divided into three design stages: conceptual, preliminary and detail.

FIGURE 2-5 AIRFRAME DESIGN STAGES

The requirement to produce a good design at the early conceptual stage is to ensure an optimised final design. It is a common habit for young engineers to give very little attention to the conceptual design and instead perform the optimisation at a later stage of detail design, which, by then, is mainly the territory of the senior experienced engineer.

Since the experience required for the work in the conceptual design stage derives from an accumulation of a wealth of experience from previous projects, it is very difficult for young engineer in the industrial environment, to reach the same level of expertise as his senior to perform the design task. There is a danger therefore of young engineers falling into *refining an un-optimised design.*

2.3.1 Structural Layout

Wing load distribution will affect the structural layout of the wing box. Prior to the process of designing a wingbox, it is important to understand the role and layout of these structural components and its effects on the load path from external load into internal stresses in wing box components.

For the type of aircraft in which the load intensity is moderate to high, it becomes practical to use the upper and lower skin-stringer panel between the spars to provide the main reaction to the spanwise bending. Thus the skins are constructed to carry the end load by supporting their area with spanwise stringers. Upper and lower skin-stringer panel design is governed by the load type, i.e. compressive buckling load on the upper, and fatigue tensile dominant in the lower panel. However, some access panels will be required on the skin panel for maintenance purposes.

To improve the damage tolerance of the structure, the skin panel construction is divided into a number of spanwise planks joined by cracking stopping butt straps. The front and rear spar are fitted with the titanium crack stopper and integral crack retarder. The spar web joint behind the rear spar supports the main wing jacking point. Holes in the spars are provided for the slat track and the spoiler and airbrake actuator pass. The spars holes have integral reinforcing augmented by separate titanium reinforcing plates. These plates are joined by the titanium crack stopper.

FAR and CS require the lower surface of the wing panel to be fail-safe. Each spanwise splice between panels is a tear stopper, which is designed to stop the failed panel continuously cracking to the adjacent panels. The rivet patterns and shear strength shall be designed such that it is strong enough to transfer a failed panel load (fail-safe load) into the adjacent panel.

The upper panel on a wing structure is also designed to be fail-safe, but since the only structural separation that can occur is during ground operation where the tension load is small, the FAR / CS requirements could be easily justified. The skin panel is designed as wide as possible to minimise the weight and the expense of spanwise joints. Since positive flight load factor is always higher than negative flight, the wing's upper surface is usually critical for compression load. When a large weight such as the fuel tank is concentrated at the wing tip, the upper surface near the tip may be critical in tension for a positive flight condition.

From the structural standpoint, appreciable weight saving is possible through the integral-section design; a design which has developed high resistance to buckling loads as well as a reduction in the number of basic assembly attachments to give a smooth exterior skin surface. In aircraft application, the most significant advantages of integrally stiffened structure over comparable riveted panels have been (Niu, 1999):

- Reduction of amount of sealing material for pressurised fuel tank structure

- Increase in allowable stiffener compression loads by elimination of attachment flanges
- Increased joint efficiencies under tension loads through the use of integral doublers, etc.
- Improved performance through smoother exterior surfaces by reduction in number of attachments and non-buckling characteristics of light weight skin structure.

Niu (1999) has reported studies that an integrally stiffened section (blade) can attain an exceptionally high degree of structural efficiency. A weight reduction of approximately 10-15% was realised by the use of an integrally stiffened structure. A study (Niu, 1999) conducted on typical transport aircraft upper surfaces of integrally stiffened and built up skin stiffener types of construction with rib spacing of 26 inches has been made under the simplifying assumption that an optimum design is attainable for all stations along the wing span. In this analysis, all non-optimum factors (such as joints, cut-outs, etc.) are ignored. The integrally stiffened skin and the stringers are manufactured from 7075-T6 aluminium alloy extrusion, and the skin is 7075-T6 bare plate. In both cases a 10% margin for shear bending interaction was maintained. The resulting weight of the integrally stiffened upper surface is 6000 lbs and that of the skin-stringer surface is 6600 lbs. It is indicating that the built up configuration to be approximately 10% heavier than the integral construction.

In order to obtain true weight difference, all non-optimum factors must be taken into account. The integrally stiffened design will have a relatively low weight for the so called non-optimum features. This is attributed to the machined local padding and reinforcing material and permitted by integral cover construction. In contrast, the built-up type of design generally requires a relatively large non-optimum weight because of the many chordwise splices for ease of tank sealing and fabrication, discrete doublers and so on.

2.3.2 Existing Aircraft Structure Assessment

The different structural requirements of the aircraft component lead to a variety of constructions. Several concepts of aircraft structure incorporate the features associated with the latest requirements in airworthiness regulation and current technology, both of which are fully utilised in order to meet the customer target.

The design scenario for concept generation is firstly to gather information on existing similar aircraft and to break it down into each component related in the product breakdown structure. This is used as the concept baseline. Secondly, the product is analysed to seek some possible improvements through the introduction of new concept material, process, assembly, and maintenance technology. The information on the lesson learned (best practise) in industries on similar aircraft configuration could be useful during this analysis as it comprises the assessment of function, manufacturing and maintenance aspects.

To further improve the assessment on existing aircrafts, the following design information could be gathered from similar aircraft. In this situation, the concept of wing box configuration is broken down into smaller components for the following assessment:

- Function of configuration
- Materials and manufacturing process
- Joint method and assembly process
- Structure protection and maintenance access

For example, table 2-1 and table 2-2 display information on several configurations applied to different aircrafts that could be used during the design process. The assessment could begin by comparing these various configurations. Such a comparison provides a greater understanding about the critical parameters of the wing box structure, including the configuration, the material and the sizing than just following the common tradition of doing things similarly for the sake of minimising the risk.

TABLE 2-1 AIRFRAME CONFIGURATIONS OF EXISTING WING AIRCRAFTS (NIU, 1999)

Aircraft name	Wing panel	Construction	Material Skin/stringer	Panel shape	Spar & Ribs
DC-9	Upper Lower	Skin-str Skin-str	7075-T6/ – 2024-T4		2 spar; rib-web
DC-10	Upper Lower	Skin-str Skin-str	7075-T651/7075-T6 2024-T351/7075-T6		2 spars; rib-web; 18-34 in
B747	Upper Lower	Skin-str Skin-str	7075-T6/-7075-T6 2024 / 2024		3 spar; rib-web; 25 in
B737	Upper Lower	Skin-str Skin-str	7178-T651/- 2024-T351/-		2 spars; rib-web
B757 and B767	Upper Lower	Skin-str Skin-str	7150-T6/7150-T6 2324-T3/2224-T3		2 spars; rib-web
L-1011	Upper Lower	Skin-str Skin-str	7075-T7651/-7075- T7651		2 spars; rib-web; 21 in
A300	Upper Lower	Skin-str Skin-str	7075-T6/- 2024-T3/-		3 spars; rib-web
A310	Upper Lower	Skin-str Skin-str	7075-T651/7075- T651 2024-T351/2024- T351		2 spars; rib-web
A330/A340	Upper Lower	Skin-str Skin-str			2 spars; rib-web

The above table imparts useful information on different types of configuration. The designer could see these as the initial point for starting the wingbox configuration. For example, the following table is created based on the data of similar aircrafts, as in the case study, describing some of the critical parameters on different components of wingbox structure.

TABLE 2-2 THE SUMMARY OF WING BOX STRUCTURE OF SIMILAR EXISTING AIRCRAFT (NIU, 1999)

Design criteria	Wing structural components					
	Upper Wing panel	Lower wing panel	Front Spar web	Rear spar web	Heavy Wing ribs	Light wing ribs
Structural principle	Built-up construction.	Built-up construction Skin comprise several panels to improve damage tolerant	Integral or differential construction Titanium crack - stopper and integral crack-retarder	Integral or differential construction Titanium crack - stopper and integral crack-retarder	Integral construction	Integral construction
Material	7075-T6 7150 7055	2024-T3 2324-T39	7075-T6 7150 7055	7075-T6 7150 7055	7075-T6 7150 7055	7075-T6 7150 7055
Fabrication	Rolled stretched plate and stretched extrusion. Taper machined and pocketed to save weight. Shot peening to return the shape after machining process.	Rolled stretched plate and stretched extrusion. Taper machined and pocketed to save weight. Shot-peening is used for lower surface double curvature	Rolled stretched plate and stretched extrusion. Taper machined and pocketed to save weight Shot peening	Rolled stretched plate and stretched extrusion. Taper machined and pocketed to save weight Shot peening	Machined element and pocketed to save weight Shot peening	Machined element and pocketed to save weight Shot peening
Assembly and joints	For fatigue critical areas, interference bolts with high performance used	For fatigue critical areas, interference bolts with high performance used	For fatigue critical areas, interference bolts with high performance used	For fatigue critical areas, interference bolts with high performance used	For fatigue critical areas, interference bolts with high performance used	For fatigue critical areas, interference bolts with high performance used
Protection and Access	chromic acid anodised and sealed	chromic acid anodised and sealed	chromic acid anodised and sealed	chromic acid anodised and sealed	chromic acid anodised and sealed	chromic acid anodised and sealed

2.3.3 Material Assessment

New material that is lighter, stronger, easier to maintain, more damage tolerance and cheaper to produce is continuously developed for different types of applications on the aircraft. The current trend has shown how the application of specific material has benefited the aircraft to fly faster and cheaper. The material can be divided into two major categories: metallic and composite.

The metallic application is still the major part of the current commercial aircraft. It is favoured by the customer due to its predicted properties, ease of inspection, repair and established tools for maintenance. The material availability and handling are also significant factors for the use of metal. The following figure highlights the industry requirements of a material supplier:

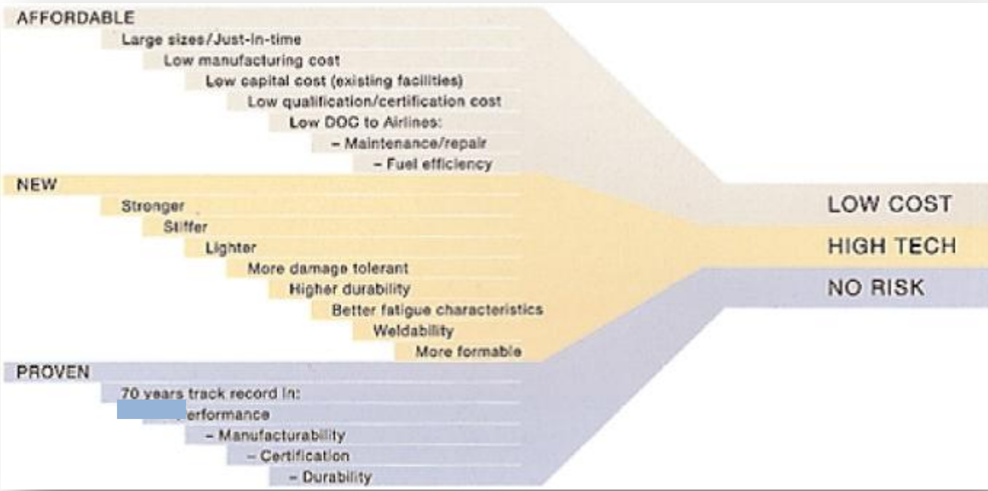


FIGURE 2-6 NEW MATERIAL CONSTRAINTS (ALCOA, 2008)

Nevertheless, the requirement for composite application has been growing rapidly for future aircraft, namely the A350 and B787. These new aircraft will use more than 50% composite materials. The main challenges normally associated with use of composite structure, such as the cost increase in material, production , inspection, and repair, could outweigh the cost saving due to the weight reduction, have been minimised with a greater understanding on the mechanics of composite material, the manufacturing process and the maintenance procedure. Inspection cost can be lower due to increased inspection intervals (even though individual inspection can be more complex). Boeing Aircraft Company has put this forward as a reason for the composite B737 fuselage.

However, it should be remembered by the designer that the use of new material is an evolutionary process and involves a large commitment of time and resources from both the manufacturer and supplier. It is therefore preferable to prioritise the material selection from the established material using a novel construction design than the other way round.

Figures 2-7 and 2-8 illustrate the practise used by aircraft manufacturers in applying new material and processes. Before new material and processes can be certified, several stages of test, beginning with a coupon test and ending with a complete aircraft test, need to be undergone.

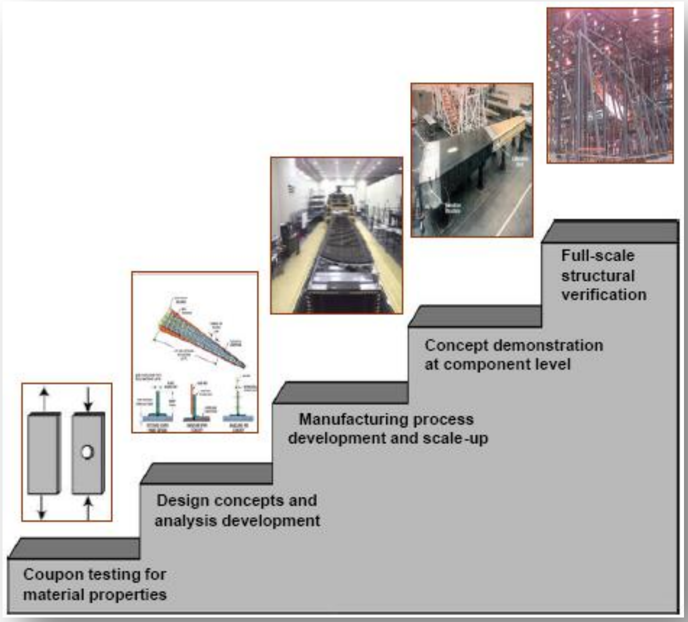


FIGURE 2-7 BUILDING BLOCK APPROACH (HARRIS ET AL., 2001)

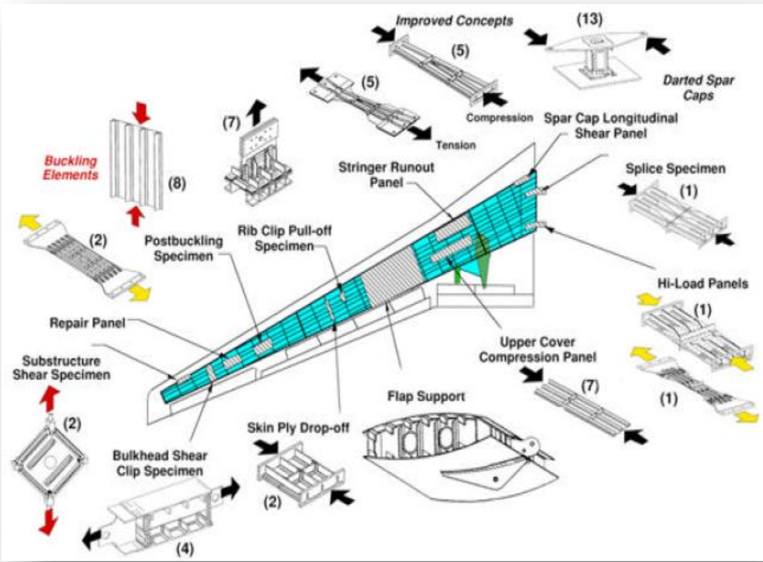


FIGURE 2-8 DESIGN DEVELOPMENT TEST IN THE BUILDING BLOCK APPROACH (HARRIS ET AL., 2001)

The following considerations are given during the selection process on the use of advanced technology and new material (Harris et al., 2001):

- Acquisition, manufacturing, certification and lifecycle cost; incomplete understanding of failure mechanism and their interactions; technological risk; and the state of material supplier base.
- Increasingly, airframe manufacturer are using an integrated product development approach that considers such factors as producibility, cost, non-destructive evaluation (NDE) methods and criteria, and repair and maintenance issues; and involves airlines, and supplier from the outset of development program.
- Commercial aircraft are built and operated on a global basis with international teaming of manufacturer and supplier. The competitive pressures (cost) will continue to influence the selection criteria for the application of new material and the processing of technology.
- With the increased emphasis on affordability, it is likely that less new material will be developed. On the other hand, robust and cost-effective processing methods as well as compliance with environmental regulation will become paramount issues to provide lower cost.

The following figure shows the trend of new material applications on transport aircrafts in Boeing’s product line (Airliners, 1998). During design synthesis, this information could be a useful starting point for designers to compare the trend of new material application in other companies, and to begin their search for the latest and best material developed by suppliers. It is important to note here that the material data used in the structural analysis is based on established sources, i.e. from the manufacturer's test data result, such as described in FAR/CS 25.613; if the test result data is unavailable, it is based on material handbooks, such as in Mil Handbooks 5J (DoD, 2003) and subsequently superseded by FAA in document MMPDS-01, or from the material supplier's data.

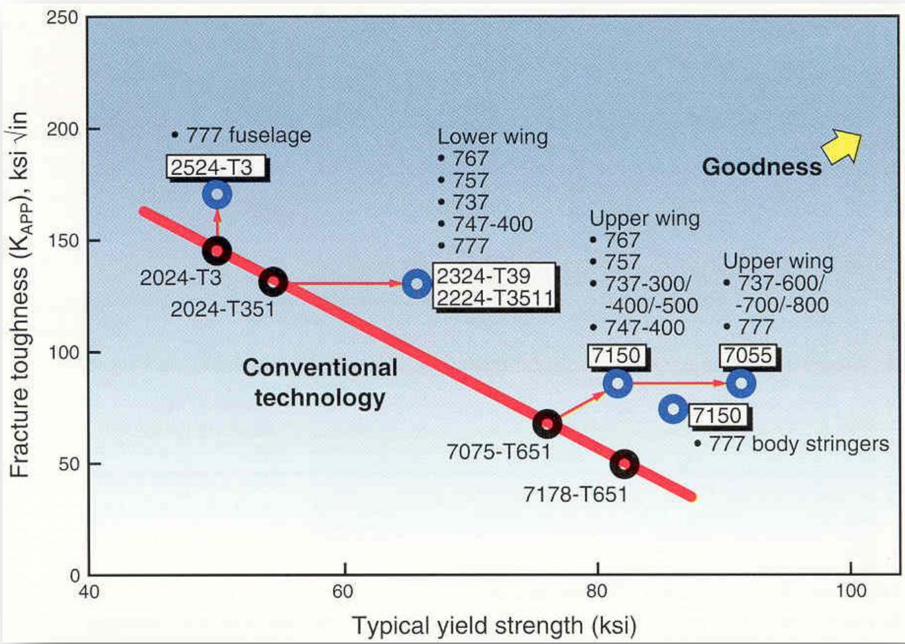


FIGURE 2-9 METALLIC MATERIAL DISTRIBUTIONS ON BOEING'S COMMERCIAL AIRCRAFTS (AIRLINERS, 1998)

A similar graph is produced by a major material supplier company specialising in metallic material on the application of the latest material and its trend:

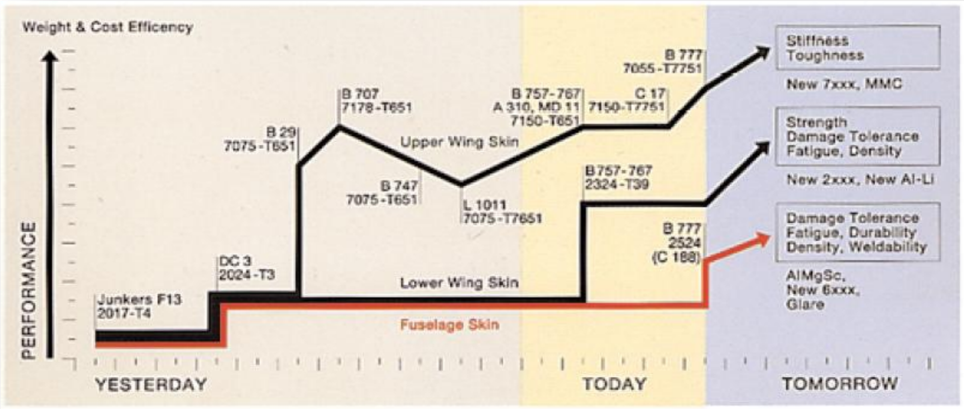


FIGURE 2-10 ALUMINUM ALLOY DEVELOPMENT AND APPLICATIONS (ALCOA, 2008)

2.3.4 Aircraft Joints

Structural joints are mainly provided by slug automatic riveting, but for fatigue critical areas and high stress areas, interference bolts, such as hilok bull nose are used. The following figure shows an example of top mounted wing structure configuration:

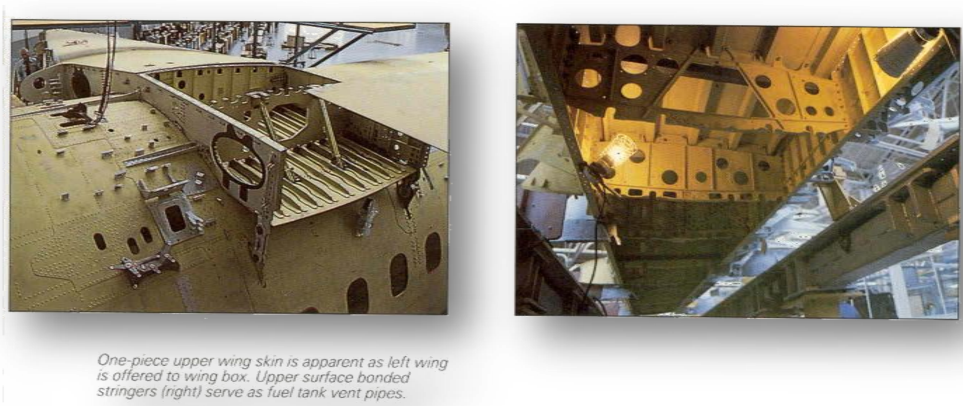


FIGURE 2-11 CENTRE TO OUTER WING JOINT ON RJ146 TOP MOUNTED WING AIRCRAFT (BAE, 1998)

As a comparison, the next figures are for wing joint attachment of lower mounted wing:

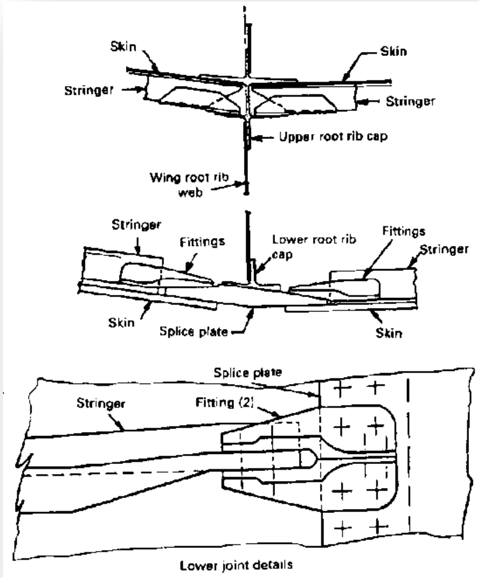


FIGURE 2-12 WING ROOT SKIN-PANEL JOINTS B727 (NIU, 1999)

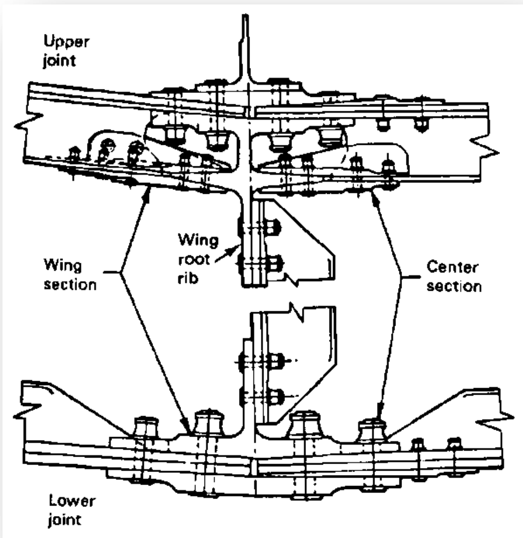


FIGURE 2-13 WING ROOT SKIN-PANEL JOINTS L1011 (NIU, 1999)

2.4 Design for Maintenance

The current generation of civil aircraft were designed to last at least 20 to 25 years and up to 90,000 flights. However, these designed service goals are exceeded by many operators of jet and turboprop. Future types of aircraft are designed for at least the same goals, but their structure, designed for higher fatigue life, higher damage tolerance capability, and higher corrosion resistance, is required to minimise the maintenance cost and to comply with the requirements of the operator and the enhanced airworthiness regulations (Schmidt et al., 2000).

Structural maintenance is taken into consideration during the conceptual stage by assessing the design concepts against the potential problems. It is an exhaustive process, but it will minimise later-stage large or costly modification to achieve the desired reliability and maintainability (R&M) target. The qualitative assessment is enough to allow development of a generation of design concepts that are reliable and maintenance friendly.

One of the simplest techniques to assess the R&M qualitatively at the early stage of structural design process is firstly to look at the modes of damages in the aircraft structure, i.e. fatigue, environmental (corrosion) and accidental damages. The designer could then identify the significant items of the structural maintenance that could greatly deteriorate the R&M of the aircraft. These significant items are subsequently rated against each mode of damage and used to predict the frequency of the exposure to and the location of damages. Finally, a potential solution to the R&M issues, such as structural protection, inspection requirements and repair, are discussed and rated. The concept selection, therefore, shall be based on the total assessment and rating from these previous steps.

The maintenance cost of the airframe is largely dependent on the direct labour cost, which is 45% of the total cost, and the material cost and subcontract work, which is 30% of the total cost. These figures are quite different to engine maintenance, whose costs constitute just 9% and 52% of the total cost respectively. The effect of component modularity and ease of inspection on the engine design and the high cost of superior performance material contributes to this distribution. Nevertheless, for the airframe maintenance, the objective of this approach is to reduce the labour cost whilst keeping the cost for material and subcontracts constant or less.

As an illustration, based on an International Air Transport survey (Green, 1996), it was estimated that 36-40% of damage to aircraft is from ramp and maintenance damage, sometimes called friendly foreign object damage. To safeguard the R&M of the aircraft against accidental damage on the ground, special attention should be given to the interfaces areas on aircraft structure, servicing and other equipment especially prone to damage. Consideration of aircraft maintenance and repair will be an important part of designing the airframe construction and selecting the material. Previous service experience with the same typical aircraft will also be valuable in supporting design for maintainability.

Aluminium material can be treated with Alodine and other similar product (note: Alodine is a specific Henkel product) or Chromic acid. Therefore, as another example, specific consideration needs to be given to the areas of high contamination and high condensation, where anodic corrosion between different materials could occur. Areas which are subject to contamination by aggressive fluids are primed and painted with fluid-resistant primer and top coat. To avoid water accumulation, drain holes are also provided in the critical areas.

The wing access holes are provided at the skin panel and must be large enough for a man to pass through to inspect and reseal the inside if necessary. On a shallow wing section, the access must be in the lower surface to allow maintenance people acceptable work access, although they cannot climb in completely. Apart from the sealing problems associated with the lower access panel, it is primarily a tension skin and hence introduces stress concentration in the area where crack propagation is a major consideration. To overcome this problem, man-hole doors are machined elements and non load carrying, except for a few load carrying doors in the outer wing. A non load carrying door consists of an inner sealed door and outer door shaped to the wing profile.

If every consideration is given to the above factors during the design process, the design will not suffer from severe reliability problems. However, the challenge remains of how to qualitatively and quantitatively measure the reliability characteristics of certain concepts comparative to others.

Some of the key points that should be considered during reliability and maintainability design at the conceptual stage include:

- Understanding the aspect of design decision, such as on configuration, material fabrication/assembly on the probability of failure and its consequences. The designer then has to predict what the causes of failure are and how to inspect and repair them.
- Maintenance Steering Group (MSG)-3 guidelines combined with airlines and manufacturer information on current and past aircrafts are very valuable and should be referred to as much as possible. This information represents the complete picture of the overall interaction factor on the reliability and maintainability (R&M). MSG-3 is maintenance process oriented. The process is the means for classifying the way in which a particular component is maintained. The intent of the process is to ensure the inherent design reliability of the component is maintained.
- For design purposes, the above information can be systematically arranged to guide the conceptual design team in decision making process on potential R&M concern and potential design solution

The team needs only to be well equipped with enough experience and knowledge based tool on reliability to improve the quality of design work; in-depth knowledge on Failure Mode and Effect Analysis (FMEA) is not a necessity. Therefore, for the designer it is the principle of risk assessment that is significant rather than the knowledge of a complex analysis method.

2.5 Design for Manufacture

2.5.1 Cost Driven Structure

The cost driven structure has changed the design and manufacturing relationship in aircraft design process. The main design objective used to be structural performance which was translated into the minimum weight requirement. But experience has shown this to be not the only main factor: the cost of high performance material, which can be expensive and difficult to manufacture and assemble, outweighs the weight saving. Experience also shows that product simplicity, reduced part numbers, redesign, fabrication and assembly sequences, and 'parts availability/off the shelf material' will be the primary ways of cost reduction (Barrow, 1997). The understanding of the designer of the manufacturing process, and its capability and limitation will greatly improve the manufacturability of the design.

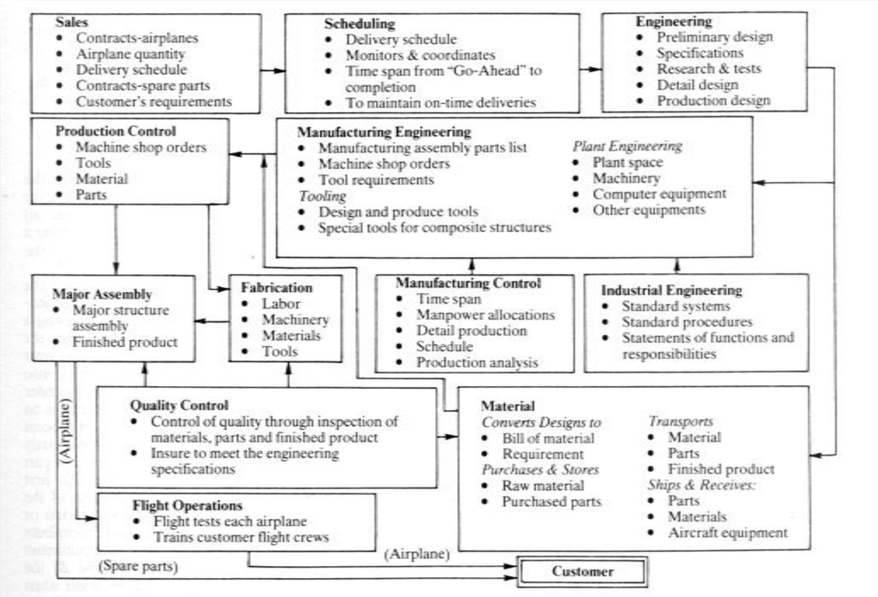


FIGURE 2-14 TYPICAL DESIGN FOR MANUFACTURE PROCESS (NIU, 1999)

Figure 2-14 above identifies the much earlier process prior to an aircraft being manufactured. It clearly shows the specific structural design dictating many stages of the manufacturing process. For example, the type of skin stringer panel, i.e. built-up or integral, will determine the form of material to purchase and the type of machine to process the raw material into the final shape. Hence, if manufacturing is considered much earlier in the design stage, any difficulties with the ordering of raw material and the machine capabilities could be solved much earlier. Difficulties of fabricating the parts could be anticipated much better on the drawing board if the production cost is known.

Figure 2-1 showed the cost-commitment at different product development phases. The production stage has little effect on cost saving, but on the other hand, any changes during this stage significantly affect cost. In essence, making the concept producible reduces the risk and eliminates unnecessary rework during later stages.

A good starting point for cost reduction is to make possible alternatives available when making a design. It is often impossible to determine the best alternative without careful analysis of the probable manufacturing cost. Designing for function, interchange ability, quality, and economy requires a careful study of product quantity, production rate, tolerances, surfaces, finishes, processes, materials, and equipment (Bakerjian, 1992).

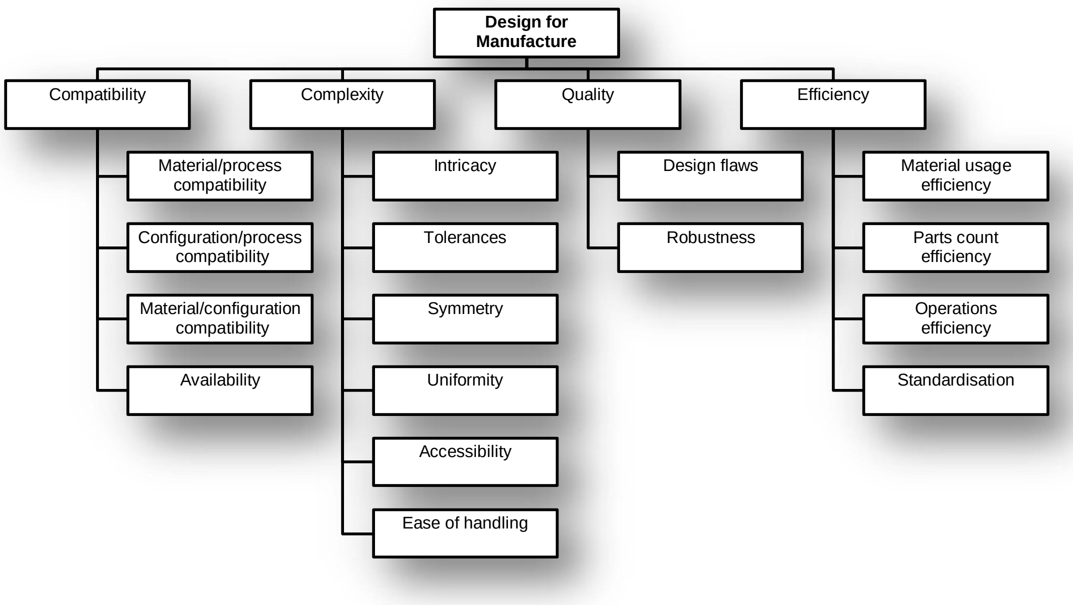


FIGURE 2-15 DESIGN FOR MANUFACTURE AND ASSEMBLY (ALI-KHAN AND FIELDING, 1997)

The above figure shows that manufacturing problems can be reduced or eliminated by considering the manufacturing and assembly aspects during the conceptual stage. The selection of material for example, dictates the type or manufacturing process, and tolerances related to it. The size and complexity of a product dictates the manufacturing process and assembly activities. If the above factors could be understood at the very early stage, and with input from best-practices in manufacturing and lessons learned, then problems could be avoided at minimum cost.

An awareness of company capability and technology and supplier availability would broaden understanding and reduce the risk of producing concept(s) that are difficult to manufacture or to be supplied by vendors.

In addition to the above aspect, the products are also designed to allow for later cost-effective modification to fulfil future requirements, such as for a product family. Hence, the design team needs to consider whether the change in configuration to accommodate the change in requirement could be produced cost-effectively using the available tools and jigs and then the product family could be maintained with minimum need for additional investment for maintenance.

2.5.2 Manufacturing cost

Manufacturing cost estimation leads to the following objectives during the development of an approach for structure conceptual design process:

- The need to design a product that can be manufactured more cost effectively and be robust to variation.
- The need to design a product that can be tailored to future requirement or different market without costly modification.
- The need to incorporate the technique on evaluation of manufacturability and producibility metrics into the design approach.
- The need to recognise economic consideration as the main driver for concept selection.

Swift and Booker (2003) proposed a strategy for implementation during the manufacturing selection:

- obtain an estimate of the annual production quantity
- choose a material type to satisfy the product design specification
- select candidate manufacturing process
- consider each candidate against the engineering and economic requirements; these include:
    - understand the process and its variation
    - consider the material compatibility
    - assess conformance of component concept with design rules
    - compare tolerance and surface finish requirements with process capability data
- consider the economic positioning of the process and obtain component cost estimates for alternatives
- review the selected manufacturing process against business requirements

The importance of estimating the manufacturing cost is one of the most important aspects of the selection of amanufacturing process. The cost to produce a part can be estimated by considering the material and processes to produce a part with a specific form, complexity and tolerance.

High Speed Machining Process:

The increase in the removal speed using different cutting tools will save the manufacturing cost quite significantly. It is therefore suggested to study the effect of using the high speed machining (HSM) process on the recurring and non-recurring cost. Conventional machining of aluminium is achieved with cutter rotations of roughly 3000 revolutions per minute (RPM); high-speed machines have rotations of 10,000 to 40,000 RPM with considerably higher metal removal rates than conventional machining. One advantage of this technique is simply faster part fabrication and hence a reduction in machine operator hours per pound of part. A more fundamental advantage is that with multi-axis cutters running at high speeds, HSM can produce more complex unitized parts than can conventional machining; and, as noted previously, unitized parts save weight and assembly time (Younossi et al., 2001).

HSM is also characterized by a significant reduction in machining forces and heat absorption by the part. It dramatically shifts the heat energy distribution from the cutter/work-piece to the chips. Because of the reduced heat build-up and force required for the cutter, the webs and flanges of the part can be thinner, thus saving weight.

Autoclave Process:

To minimize the exfoliation corrosion or inter-granular corrosion on extruded and heavily worked aluminium alloy, an autoclave process is used et al., 2001). It works by putting the machined material into an autoclave at an elevated temperature and pressure to achieve the final shape, which is then heat treated to increase the strength properties and alleviate the exfoliation.

Advanced Alloy Material:

The use of advanced alloy material should be considered seriously for the next generation aircraft. Some of the proposed materials which are being suggested are listed in table 2-3 and figure 2-16 below:

TABLE 2-3 FUTURE ALLOY FOR AIRCRAFT STRUCTURE (LEQUEU ET AL., 2001)

Aircraft part		Principle design driver(s)	Reference alloy	Proposed alloy
Wing	Upper panel	Compression	7150-T6/T77 7050-T74	7449-T6/T79/T76
	Upper stringer			
	Lower panel	Damage tolerance	2024-T3/2324-T39	IS262-T3/IS249-T3
	Lower stringer	Tension and DT	2024-T3	IS249-T3
	Spar and ribs	Static and K_{IC}	7010-T76/7050-T74	7040-T76/7449-T76

Fuselage	Upper panel	Compression and DT formability	2024 clad T3	2024A and HF clad
	Lower panel	Tension and DT		6056 clad T6 / bare T78
	Stiffeners	Tension / compression	7175-T73	7349-T6/T76
	Main frames	All kind / complex	7010 & 7050-T74	7040-T74
	Seat tracks	Tension	7175-T73/T79	7349-T6/T76
Other structural parts Engine fittings		All kind	7010/7050/7075 Plate/Forging	7040-T74 Plates

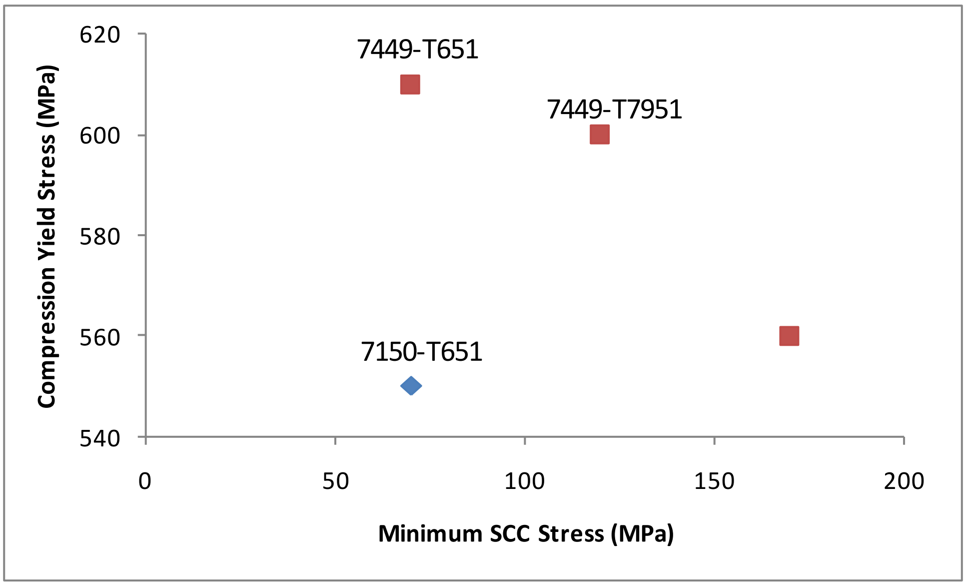


FIGURE 2-16 TYPICAL COMPRESSION PROPERTIES AND MINIMUM SCC STRESS FOR 7449 ALLOY IN VARIOUS TEMPERS (LEQUEU ET AL., 2001)

Overall Process:

By looking through the entire process from raw material to finished aircraft parts, there are sequences of processes, all of which are candidates for cost reduction initiatives. The following table gives a simplified list of the processes together with some of the cost saving initiatives (Lequeu, et al., 2001):

TABLE 2-4 COST REDUCTION INITIATIVES (LEQUEU ET AL., 2001)

	Processes involved	Cost reduction initiatives
Aluminium manufacturer	Processing of semi finished products (plates, sheets, extrusion, tubes)	Run internal continuous improvement program
	Product characterisation (NDT, release & periodic test)	Reduce number of tests through QA control and capability analysis
Aircraft manufacturer	Machining	More near-to-final shape products Low residual stress (LRS)material Machining sequences for reduced distortion Heavy gauge LRS plates as an alternatives to forgings
	Forming: Stretch forming Age forming Joggling	High formability qualities Alloy and dedicated ageing practices Stringer alloy adapted to severe joggling
	Heat treatment Solution	Avoid heat treatment because of high formability sheets

	treatment/quenching Ageing	Adapt treatments to the customer capabilities (shorter equivalent ageing practices)
	Assembly	Reduce joints: Weldable solution Integral machining of heavy gauge plates Cast doors Improved alloys for increase stringer pitch Design alternatives

2.6 Airframe Conceptual Design Tools

Throughout the design process stages, the effectiveness of the tool types can be described as in figure 2-17. Different methods and tools give varying impacts on the design decision depending on the characteristic of the design stage. Design guidelines supported by modelling and simulation give the greatest impact during the conceptual stage. The use of design tools and Computer Aided Design (CAD) is more effective when used during the detailed design stage when most of the critical configuration has already been selected (Bakerjian, 1992).

Conceptual design stages are unique as at play are infinite variations for the designer, regarding construction, material and manufacturing process to create the product. The support tools represent the design guidelines based on current and past experience, modelling, simulation, and informal reviews. The following sections review the tools provided for the tasks in the conceptual design stage.

2.6.1 Requirement capture and deployment

Farrel (1993) suggested that tools should have the capability to trace the requirements vertically throughout the product breakdown structure (product decomposition), and horizontally throughout the design life cycle, from design to manufacture to in-service.

The Quality Function Deployment (QFD) process revolves around understanding what the customer really expects and focuses efforts on meeting those needs through extensive trade-offs. QFD also provides a way of tracking and tracing trade-offs through various levels from requirements through design decisions to production and support processes.

A technique like QFD can be useful throughout the product development process, particularly for complex processes such as capturing and deploying customer requirement. The capturing process could entail a period of intensive discussion, including some preliminary work, before the requirement becomes firm. An example of this is the Lockheed Tristar, whose original concept was based on an American Airlines specification two-engine medium range aircraft. As the design developed and more potential customers became involved, the range requirement was increased with a corresponding increase of size and the addition of a third engine.

Boeing employed the QFD technique in their Boeing Phantom Works - Future Technology Aircraft Enhancement (Gill, et.al., 1997), in which a list of 44 technologies, jointly developed by Wright Laboratory (WL/FI) and Boeing-Phantom Works, were evaluated to assess the impact of each technology on each Technical Element Objective (TEO). QFD provided the capability to assess the impact of each technology on every subsequent target. These results were then used to establish three prioritized sets of technologies. The first set identified those technologies that demonstrated the best promise of achieving both the affordability and performance goals. Those technologies that illustrated the most promise of achieving the affordability goals and performance goals independently were identified as the second and third sets respectively.

A study of QFD and costing (Crow, 2002) detailed a case study regarding the application of QFD in a company under contractual obligation to deliver a quick release top nozzle (QRTN) to several customers. This is a complex subassembly costing approximately $1,700 each for a major piece of capital equipment. The current product, a removable top nozzle (RTN), is needed in case a product has to be repaired because of a failure. While this repair is a low probability occurrence, the cost of downtime is very high as significant operational costs occur if the product isn't repaired within a reasonable period. Through the process of capturing the voice of the customer, planning the product, selecting the concept, and targeting the cost, QFD helps the team bring the product into the final development and testing phases. Estimated costs at this point are approximately 15% above the RTN cost and 5% above the cost target. However, this is significantly less than the previous QRTN development where costs were estimated at 80% over the RTN costs. As a result, this project is considered a major success and a successful demonstration of QFD.

Sullivan (1986) reports that QFD system has been used by Toyota since 1977, following four years of training and preparation. The result is that between January 1977 and April 1984, Toyota Autobody introduced four new van-type vehicles. Using 1977 as a base, Toyota reported a 20% reduction in start up costs on the launch of the new van in October 1979; a 38% reduction in November 1982; and a cumulative 61% reduction by April 1984. During this period, the product development cycle (time to market) was reduced by one third with a corresponding improvement in quality due to a set of reductions in the number of engineering changes.

2.6.2 Concept generation

Any tool developed has to allow for manufacturing, assembly and maintenance considerations in the design synthesis process. Tools to support the synthesis and creativity process are normally in the form of brainstorming, synthetic, enlarging the search space, and counter planning

(Magrab, 1997). The above methods have been found effective in encouraging the designer or design team to think creatively about the design. However, the minimum information available during this stage has limited the use of detailed analysis tools; hence, to support the synthesis activity, the designer or design team have to make judgments based on their experience and limited external information data available.

Aside advances in computer systems, the computerised structural design programs were being developed to help designers explore structural configurations to achieve the optimum design. The programs were used to assist the designer and relieve repetitive design tasks. Early in the computer era, priority was given to developing Computer Aided Design (CAD) as the design tool for visualising the design concepts. The computer program is also developed for structural analysis, including finite element analysis packages for simulating the stress condition of a certain design under its load conditions and the behaviour of the structural components, and also packages for integrating the computerised design tools in one system. As computer resources are becoming more powerful, the objective became to explore other detailed aspects of structural design early in the process.

The European Commission (EC) sponsored project ‘Multidisciplinary Design and Optimization of Blended Wing-Bodies’ (MOB) is the development and application of a fully integrated Computer Design Engine (CDE) (Morris, 2002). TU Delft contributed to the project with the development of a Blended Wing-Body Multi-Model Generator (La Rocca et al., 2002), which is able to supply geometries and data to the analysis software used by the project team's various disciplinary groups including aerodynamics, structures, stability and control. A full parametric definition of the aircraft has been implemented in the ICAD environment. The ICAD Model Generator holds the 'knowledge' of the Blended Wing Body aircraft, such that consistent models can be generated, at different levels of fidelity, suitable for the various disciplines involved in the CDE. The main issues in this integrated environment are to allow different tools to communicate and to structure the knowledge of each discipline in the model generator so that the process can be integrated and work seamlessly.

TU Delft (Van Tooren et al., 2005) described an approach and tool using Knowledge Engineering and optimisation techniques. The work was developed based on several finding from MOB. Regarding design process, they attempted to allow variation of the design parameters rather than the design variables, using Knowledge Based Engineering (KBE) tools such as ICAD and the MATLAB optimisation toolbox. The implementation of the approach on composite aircraft tail, focusing on the structural analysis of the stiffened blade-type panel, shows that the optimisation process could be supported and accelerated through the automation of non-creative and repetitive design activities.

Epistemics, TU Delft and Stork Fokker AESP (Emberey et al., 2007) developed a KBE application within CATIA environment for the automation of design process fibre metal laminate (FML) fuselage skin panels for the A380 aircraft. This research has previously reported the benefit of KBE in automating the repetitive process. The current process of the prepreg plies accounts for approximately 20% of the total production preparation time. The informal model of the design process is used to produce a KBE application. With respect to the traditional process, a lead-time reduction of 75% can be gained using this KBE application. The development time for the KBE system is equal to approximately six design cycles of the traditional process. However the results of the KBE application in relation to the design made by the design engineer are highly accurate. The prepreg cutting waste can be reduced by 50% for the basic laminate of the panels.

Similar findings in KBE applications using ICAD design environments have been previously reported by MSc students from Cranfield University under the supervision of H. Locket (Hafiz et al., 2001; Martins Pires et al., 2002; Ramirez Quintana et al., 2001; Reveillere et al., 2001) for various components in aircraft design projects. These researches claim significant time reductions during the preliminary design stage.

R. Curran, et.al (Curran et al., 2007; Curran et al., 2006; Curran et al., 2005; Curran et al., 2004) from Queen's University reported their attempts on incorporating a manufacturing aspect into the design stage. Several approaches in knowledge based modelling for manufacturing and structural analysis were investigated. Curran attempted to manage the integration of the process rather than the establishment of new laws of integration. The management process is developed in CATIA's product lifecycle management (PLM) for manufacturing tool.

However, there were very little published reports from the two biggest aircraft manufacturers in the world, Airbus and Boeing Company, on the application of KBE and the integration of design and manufacturing analysis tools during conceptual airframe design stage.

One published report describes the early development of Knowledge-based Concurrent Engineering (KBCE) of aircraft structural components in the Boeing Company (Breuhaus et al., 1996). The tool automates the concurrent engineering process for aircraft structural components, which trying to assess Knowledge Based Engineering (KBE) in the design and manufacturing process. This KBCE work is developed using the existing commercial tools ICAD and CATIA, and structural analysis packages are used as an external input to the KBE system.

Similar development was also being developed in British Aerospace - Airbus Industry. The paper (Rondeau and Soumilas, 1999) details British Aerospace's (BAe) progress in the development of a tool to produce MSC/NASTRAN data decks of commercial transport aircraft wings that is integrated into British Aerospace Airbus' Generic Transport Aircraft (GTA) knowledge-based design tool, created using the ICAD Design Language. The GTA knowledge-based design tool enables a project team to design, analyse and optimise the primary structure of civil aircraft wings before creation and submission of MSC/NASTRAN decks. The company claimed that the tool rapidly produces consistent, high quality designs enabling several concepts to be considered during preliminary design. Recent developments have enabled the production of loads loop finite element

(FE) models for a number of projects in a fraction of the time previously required.

2.7 Airframe Design Process Summary

The above discussion illustrates the underlying problems during airframe conceptual design stage and is summarised in the following list:

- Aircraft manufacturers are even more constrained by less budget, less time, and fewer experienced designers than in the past.
- The effect of early decisions on subsequent design stages require early integration of design-manufacture in the airframe design process.
- There is a clear need for a structured, comprehensive airframe conceptual-design tool to operate in research institutions and academia.

The solutions to the above problems have been proposed and developed by research institutions and industries, and consist primarily of an integrated product development process. The attempts described in their publications display a trend of developing the electronic design environment where the multidisciplinary teams, either collocated or distributed, could use their own tools but still be able to communicate seamlessly. The environments being developed are almost literally bringing the later detail analysis tools (not just the people) much earlier into the conceptual design stage. At the same time, the KBE tools try to capture most of the repetitive process to reduce the time.

Based on the state of the art of the current processes and tools development, the author identifies several issues yet to be satisfactorily addressed. Difficulties still exist in providing an effective and easy-to-use approach and tool for the designers to apply in early stage conceptual design stage; in collecting manufacturing cost information for analysis at the conceptual stage; and in the limited availability of publications regarding works developed in industry as well as those of non-confidential design tools.

Therefore this research is directed toward the following tasks:

- To provide information on airframe design that is necessary but difficult to obtain in an electronic database, so that it can be accessed through the internet
- To structure the information, based on interviews from designers, text books or industrial practices, so that these can be utilised to support the airframe conceptual design process
- To develop an easy-to-use tool for the designer to employ in the airframe conceptual stage to integrate design and manufacturing assessment quickly in order to reach more effective decision making on the concepts developed.

Chapter 3 Wing Box Structural Design Methodology

A DEVELOPED APPROACH AND TOOL

3. DEVELOPMENT OF WING BOX STRUCTURAL DESIGN: A DEVELOPED APPROACH AND TOOL

The previous chapter identified the airframe design processes including best practices and problems associated with methods and tools. The new method is developed to tackle some of the problems of acquiring the relevant information on configuration manufacturing assembly and maintenance. The first objective of the strategy is to gather the best practices and lessons learned from available literature and to take into account the industrial processes. The second objective is to provide the designer with the method to generate the concept, size it, and perform the parametric study. The developed tool will provide the information on design aspect for different material, components, and manufacturing issues at different stages. The third objective of the method is to utilise the decision making techniques for airframe concepts selection.

3.1 The Proposed Approach

This chapter discusses the approach on how the airframe designer could be supported to speed up the design process and at the same time produce a robust concept that can progress to the next stage with few or no iterations. The use of decision making techniques would help the designer assess many concepts and give fair assessment.

The major tasks and also the project’s contribution to society relate to the following areas:

- The new process: to systematically integrate airframe design and manufacturing.
- The supporting tools: to speed up the process of gathering and structuring the relevant information during the airframe conceptual design process. The information should comprise design and manufacturing aspects.
- The sizing tools: to size and analyse the critical parameters to achieve the target.
- Process improvement on concept selection and decision making.

The proposed approach is therefore built around these three issues, as shown in figure 3-1.

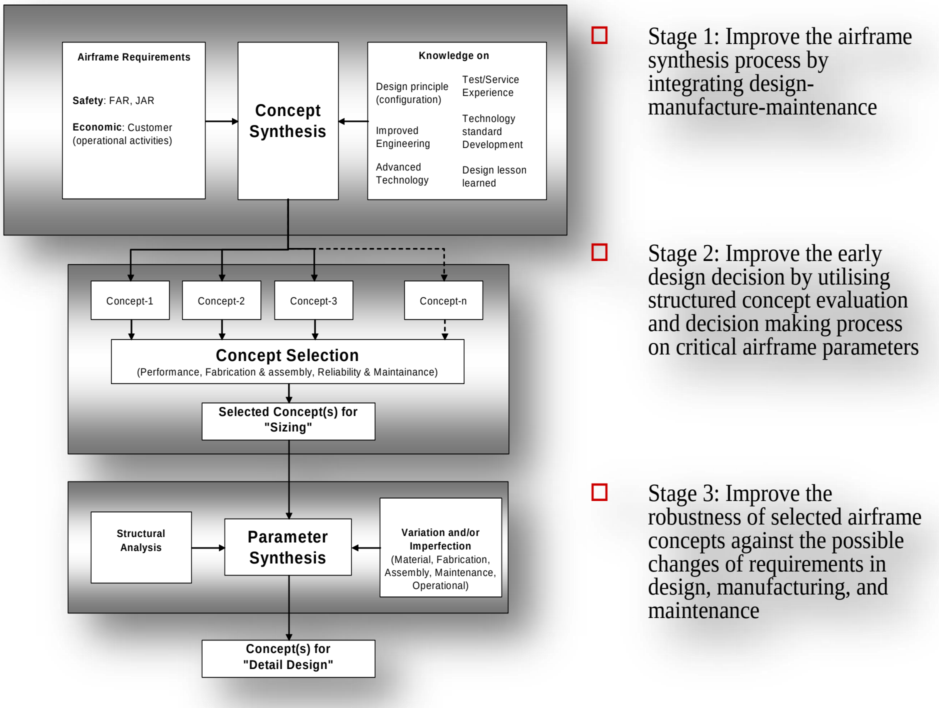


FIGURE 3-1 FRAMEWORK FOR AIRFRAME CONCEPTUAL DESIGN

3.2 Generating the Concepts - Synthesis Process

The process of generating the concepts as shown in stage 1 of figure 3-1 can be detailed as follows:

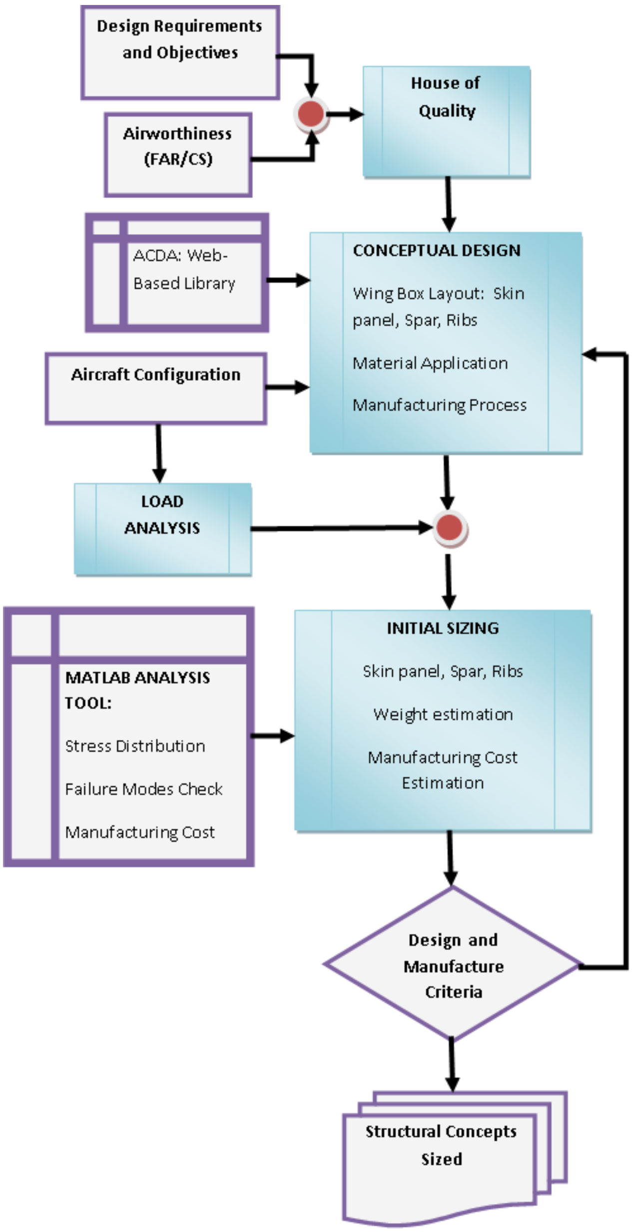


FIGURE 3-2 CONCEPT GENERATION FRAMEWORK

3.2.1 Defining and Prioritising Requirements

Defining and prioritising requirements in the House of Quality matrix relies on knowing the information that will support the design process. The more knowledgeable the designer regarding the airframe and its related issues, the better and the quicker will be the target solution. The less experienced designers will have to build their design based on information concerning new material or technological development that is scattered in text books and articles and from suppliers. It is vital that information is gathered from valid sources.

The information required depends on the amount of experience that a designer has regarding existing airframes, load paths, optimum design, weight, and operational data on the application of similar concepts on structural integrity, as well as other operational aspects of the product.

As discussed in the previous chapter, QFD is finding growing acceptance in aerospace industries during the requirement definition process. Together with the created database tool, Airframe Conceptual-Design Aid (ACDA), these will be useful for the less-experienced designers to develop their design and also to seek possible improvement based on the experience learned from current or past aircraft. It has to be mentioned that the information in the database is used as a guideline only. The designers should still seek the latest information from the appropriate vendor or supplier. A detailed explanation on how to use the QFD technique is discussed in Appendix A.

ACDA is providing the essential information in a web-based format to improve the selection of the structural layout and materials, covering the following issues:

- Design principle
- Improved engineering
- Advanced new technology
- Test/service experience
- Technical standard development
- Design lesson learned

3.2.2 Load Path and Design Check List

The majority of the aircraft structures experience a combination of the following basic load. It is very rare for a structure to have only one of the following basic loads, i.e.:

- Shear
- Torsion
- Bending
- Axial (Tension and Compression)

For design purposes, optimum structural design can be estimated quickly using the structural index. The structural index is useful in design as it contains information on the intensity of the loads and dimensions which limit the size of the structure (Niu, 1999). Therefore it can be used for:

a. Determining what type of construction fits to the particular loading

b. Sizing structures quickly

c. Selecting the most efficient material

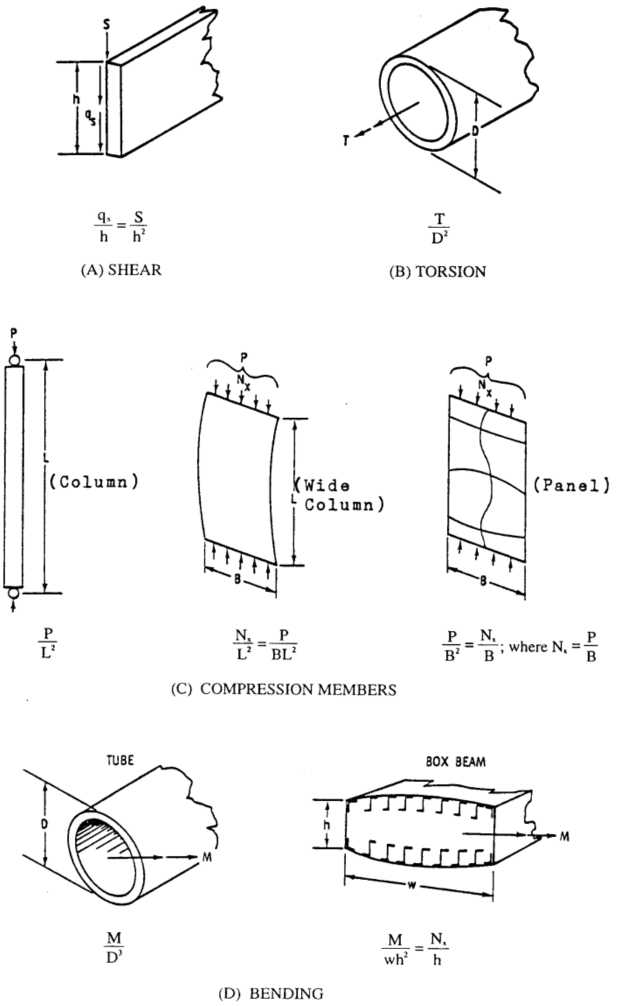


FIGURE 3-3 STRUCTURAL INDEX FOR DIFFERENT LOAD TYPE (NIU, 1999)

The above figure shows the critical configuration parameters for different loading types, i.e. shear, torsion, compression, and bending. Definition of each variable is self-explained in the figure. At the early stage of design, for any given space and applied load provided to the structural designer, he or she could choose the appropriate configuration and dimension to achieve the optimum structure.

It is also shown from the previous discussion that knowledge about airframe technology: available technology (current aircraft), including benefits and disadvantages; as well as best practices, either within the company or a new technique, will further speed up the design process.

Farrel (1993) suggested the following tasks during concept synthesis:

- Decompose functions and allocate to define alternative architecture of product processes
- Allocate parametric requirements to product and processes
- Define functional interfaces
- Define physical interfaces

At the end of concept generation stage, the following tasks are completed:

- Designing various structural concepts, selecting material, fabrication and maintenance
- Structural sizing based on strength and buckling
- Target mass statement
- Cost estimation for various configurations

Before beginning the initial sizing process, it is important to check the load path of concepts and to anticipate issues that may arise. The following check list is taken from Howe (2004):

TABLE 3-1 LOAD PATH DESIGN CHECK LISTS

Load Path Design check list: During the definition of Airframe LAYOUT
Keep load paths straight wherever possible
Keep load paths as short as possible, the load will always take the stiffest route, and this is often the shortest
Where load paths intersect maintain orthogonality if possible. This minimises the possibility of offset moment effects
Where load paths intersect at a point defined by the intersection of the lines of centroids of area of the members
Avoid offsets, but where an offset is inevitable arrange a structural member to react the offset moment if possible, or provide sufficient additional material.
Identify the most highly loaded path at an intersection and break the other

Load Path Design check list: REACTION of applied loads
Identify the most severe loading situation in terms of geometric configuration of the structure
Avoid reacting loads by bending structure when an alternative, e.g. shear, is available
Avoid bending due to pressure in a non circular shell by using circular arc cross section with ties across the kinks
Where tensile loading is inevitable plan for redundant load paths and/or crack stopping members
Ensure that there is adequate load and overall support to avoid premature buckling of compression members.
When a box beam is used it is desirable to make it as deep as possible, but the width depends upon the compromise between reducing stress levels and avoiding buckling. In the case of a wing,

make the width as great as possible for reasons of maximum fuel capacity

- React torques with a closed section wherever possible, providing it is not very shallow compared with the width. Relative large cut-outs in one side of a torque box can be tolerated.
- Taper heavily loaded members toward their ends when there is no connection to a comparable member
- Ensure that there is adequate backup structure to react locally applied heavy loads.

Load Path Design check list: JOINTS and CUT-OUTS
➢ Joints always cause a problems, avoid them wherever possible
➢ Avoid cut-outs in primary load carrying structure. When a cut-out is inevitable use the maximum possible corner radii. If there is a load carrying filling panel it is easier to make it work in compression and shear rather than in tension.
➢ Cut-outs are easier to handle in shear members than directly loaded ones.

3.3 Design Loads

Design air load is defined as the critical air load acting on the structure and therefore used in the structural design process. It consists of shear force, bending moment, and torsional moment distribution along the wing span. Before obtaining it, we need to calculate wing aerodynamic load, inertia relieve load due to fuel, and the landing gear and engine, if they are placed on the wing. Once the airframe mass distribution is known, then the total wing load is revised to include the airframe inertia relieve load. The process of defining design wing load is shown in the following figure:

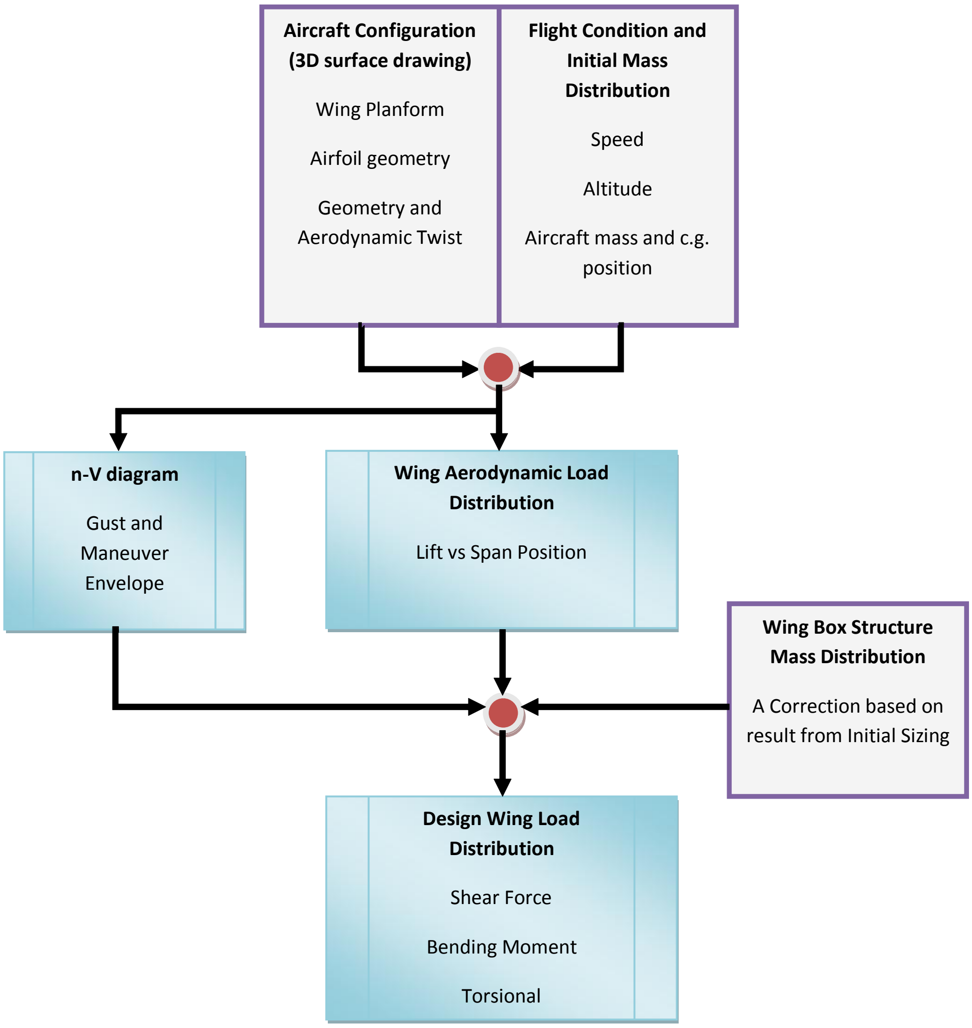


FIGURE 3-3 WING LOAD DEFINITION PROCESS

A computer program is developed to calculate total load distribution taking into account, aerodynamic load, fuel, engine, and initial mass distribution.

Based on a regulation in CS-25, a safety of factor of 1.5 is multiplied by the limit load in order to get the ultimate design load. The definition of limit load and ultimate load is explained in more detail in section 3.4.1.

3.3.1 Manoeuvre and Gust Load Factor

Load factor for vertical acceleration is selected from whichever is greater between critical flight cases of manoeuvre and gust envelope.

CS 25.337 Limit manoeuvring load factors

(a) Except where limited by maximum (static) lift coefficients, the aeroplane is assumed to be subjected to symmetrical manoeuvres resulting in the limit manoeuvring load factors prescribed in this paragraph. Pitching velocities appropriate to the corresponding pull-up and steady turn manoeuvres must be taken into account.

(b) The positive limit manoeuvring load factor 'n' for any speed up to V_D may not be less than

$$2.1+\left(\frac{24000}{W+10000}\right) \tag{3-1}$$

except that 'n' may not be less than 2·5 and need not be greater than 3·8 — where 'W' is the design maximum take-off weight (lb).

(c) The negative limit manoeuvring load factor —

(1) May not be less than -1·0 at speeds up to V_C; and

(2) Must vary linearly with speed from the value at V_C to zero at V_D.

(d) Manoeuvring load factors lower than those specified in this paragraph may be used if the aeroplane has design features that make it impossible to exceed these values in flight.

The value of n^{+}_{limit} in the *n-V* diagram is calculated according to CS 25:

TABLE 3-2 AIRWORTHINESS REQUIREMENTS FOR DETERIMING THE MAXIMUM LOAD FACTOR

MTOW	n+ limit	n- limit
W ≥ 50000lbs	2.5	-1.0
4100lbs≤ W < 50000lbs	$2.1+\frac{24000lbs}{W+10000lbs}$	-1.0
W < 4100lbs	3.8	-1.0

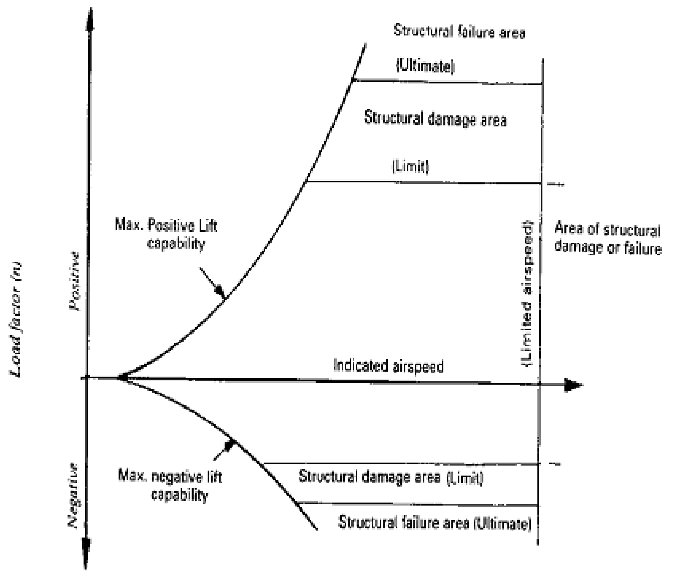


FIGURE 3-4 STRUCTURE LIMITATION ACCORDING TO THE FLIGHT ENVELOPE (NIU, 2002)

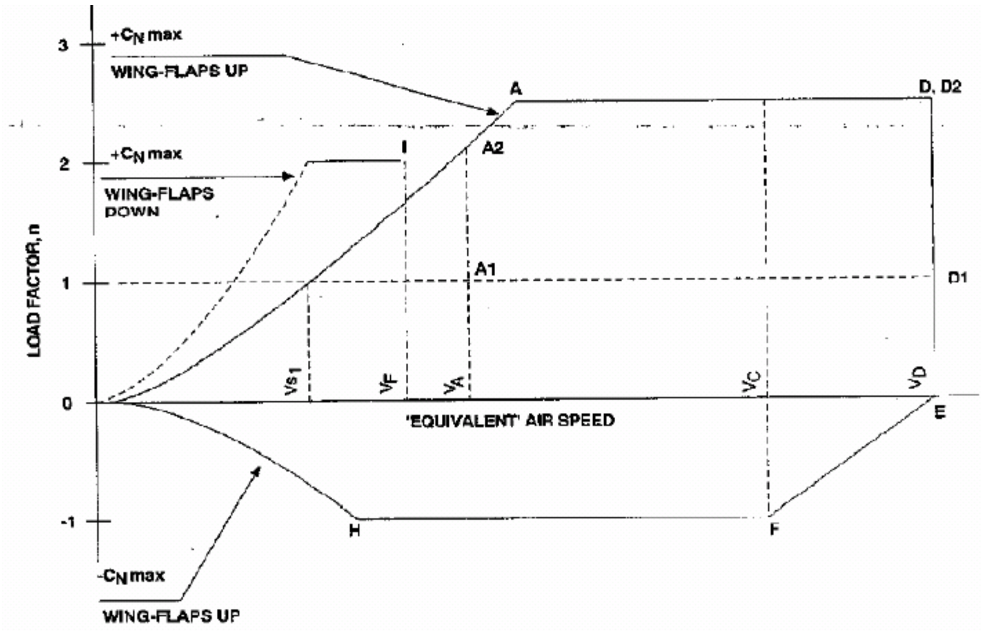


FIGURE 3-5 MANOUVRE ENVELOPES (EASA, 2003)

Additional load factors due to gust for various speeds are calculated using the equation below:

$$\Delta n = \frac{\Delta L}{mg} = \frac{\frac{1}{2}\rho_0 \frac{dC_L}{d\alpha}}{mg/S} V w_g K_g \qquad 3\text{-}2$$

Where:

$$K_g = \frac{0.88\mu_g}{5.3 + \mu_g} \quad \text{and} \quad \mu_g = \frac{2m/S}{\rho \frac{dC_L}{d\alpha} \overline{c}} \qquad 3\text{-}3$$

where:

K_g = gust alleviation factor

μ_g = non-dimensional mass coefficient

W_g = vertical gust speed

By superposition, the manoeuvre and gust load factors, the maximum and minimum vertical load factor for the aircraft structure, are determined.

3.3.2 Spanwise Wing Load Distribution with Aerodynamic Twist

The wing load tool is developed to calculate spanwise aerodynamic load distribution for a wing of low subsonic transport aircraft with aerodynamic twist. For this type of aircraft there are several methods available for example: Diederich's (Diederich, 1952) and Schrenk's (Schrenk, 1940; Peery, 1950) methods. The methods are relatively simple and yield satisfactory results for the conceptual design purposes. This research uses the latter method for the developed tool. Schrenk's method may be considered as a simplified vortex line theory and has been accepted by the Civil Aeronautics Administration (CAA) as satisfactory for this type of aircraft (Schrenk, 1940; Peery, 1950).

Most wings are designed to have better stalling characteristics by using different airfoil sections near the tip i.e. with a more negative zero lift angle than near the root, or by making the zero lift chords non-parallel along the span (twist angle). The aerodynamic load distribution for wings with aerodynamic twist is obtained in two parts. The first part, called the basic lift distribution, is obtained for the angle of attack at which the entire wing has no lift. The second part, called the additional lift, is obtained by assuming the wing has lift but no aerodynamic twist. Therefore, ***the total lift coefficient distribution is***:

$$c_l = c_{lb} + c_{la} \quad \text{3-4}$$

The method of calculating the ***additional lift coefficient*** consists simply of averaging the lift forces obtained from an elliptical lift distribution with those obtained from a planform lift distribution

$$c \cdot c_{la1} = \frac{1}{2}\left(\frac{m_o}{\overline{m_o}} c + \frac{4S}{\pi b}\sqrt{1-\left(\frac{2y}{b}\right)^2}\right) \quad \text{3-5}$$

Where:

c_{la1} = additional lift coefficient on sections corresponding CL=1.0

c = chord of section

S = wing area

b = wing span

y = position of the airfoil section at the lateral axis

m_o = the slope of lift coefficient

$\overline{m_o}$ = is the average slope of section lift coefficients and calculated from the following equation:

$$\overline{m_o} = \frac{\int_0^{b/2} m_o c dy}{\frac{S}{2}} \quad \text{3-6}$$

It is important to note that c_{la1} in the above equation is relative lift coefficient where reference maximum lift coefficient is 1.0. The actual value of $c_{l\text{-}actual}$ is calculated by multiplying c_{la1} with lift coefficient of aircraft C_L that is calculated from equilibrium of flight condition, i.e.:

$$C_L = \frac{W}{\frac{1}{2}\rho V^2 S} \qquad 3\text{-}7$$

Therefore the actual additional lift coefficient is:

$$c_{la} = c_{la1} \cdot C_L \qquad 3\text{-}8$$

The ***basic lift coefficient*** distribution is obtained from the following equation:

$$c \cdot c_{lb} = \tfrac{1}{2} \cdot c \cdot m_o \cdot \alpha_a \qquad 3\text{-}9$$

Whereas:

c_{lb} = the basic lift coefficient at any point on the span

α_a = the angle of attack in radians measured from the zero-lift plane of the entire wing to zero-lift chord line for the section

And to calculate the wing angle of attack for zero lift is obtained from the following equation:

$$\alpha_{w0} = \frac{\int_0^{b/2} m_o \alpha_{aR} c\,dy}{\int_0^{b/2} m_o c\,dy} \qquad 3\text{-}10$$

Where an arbitrary reference plane is assumed and α_{aR} is measured from this plane to the zero-lift chord of each section, α_{w0} is the angle from this reference plane to the plane of zero lift for the wing.

3.3.3 Internal Stress Distribution

Shear Force, Bending Moment, and Torsional Moment distributions are calculated from the integration of small elements of forces acting on the wing box structure. For a typical aircraft configuration, the load distribution is shown in the following figure:

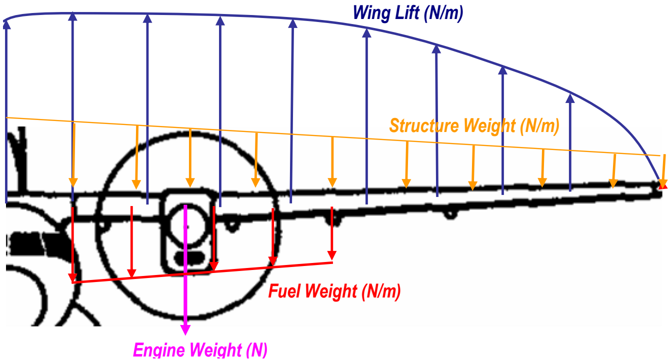


FIGURE 3-6 TYPICAL WING LOAD DISTRIBUTION

The wing is assumed to have a fixed-end condition at the wing root position, with the tip being a free condition. Again, it is important to note that wing structure mass and fuel load distribution are likely not linear; therefore, the above lines are only for illustration.

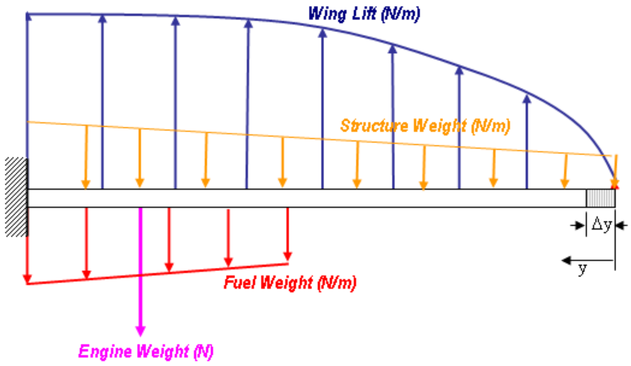


FIGURE 3-7 MODELING OF FORCES DISTRIBUTION ACTING ON AIRCRAFT WING

Please note that, the y-position during the modelling in figure 3-7 is assumed to be started from the wing tip.

When considering small elements from wing tip to the root and using equilibrium equation for each element, then:

$$SF_{i+1} = SF_i + \Delta L_{i+1} - W_{fuel}\Delta y - W_{engine} - W_{Structure}\Delta y \qquad 3\text{-}11$$

And bending moment distribution can be calculated using the following equation:

$$BM_{i+1} = BM_i + \Delta BM_i \qquad 3\text{-}12$$

and $$\Delta BM_i = (SF_{i+1} + SF_i)\frac{\Delta y}{2}$$

Torsional Moment at the shear centre of each section is calculated as follows:

$$T_{i+1} = T_i + \Delta T_i \qquad 3\text{-}13$$

and $$\Delta T_i = \Delta M_{ac} + \Delta L_i\left(\frac{(c_{FS} + c_{RS})}{2} - 0.25c\right)$$

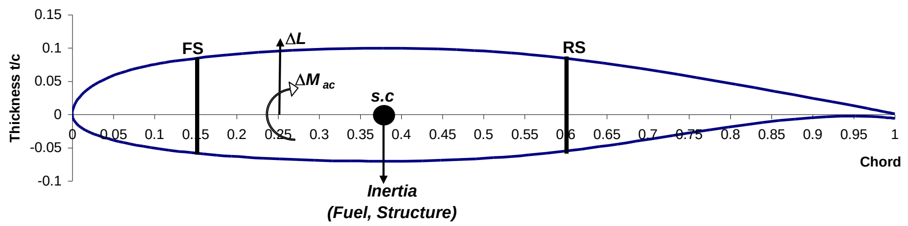


FIGURE 3-8 WING BOX CROSS SECTION

Where:

BM_i, T_i,and SF_i represent bending moment, torsion, and shear force at any point in the spanwise section. L and M_{ac} are the lift and the aerodynamic moment around the aerodynamic centre.

$$\Delta L = \tfrac{1}{2}\rho V^2 \Delta S c_l \qquad \text{AND} \qquad \Delta M_{ac} = \tfrac{1}{2}\rho V^2 \Delta S c_{MAC} c \qquad 3\text{-}14$$

c_{FS} and c_{RS} are the position of front and rear spars from the leading edge.

3.4 Wing Box Initial Sizing and Analysis Procedure

The method used for initial sizing is based on three principles:

- Using the method developed by Howe (2004) combined with the sizing technique and failure modes analysis used within industry.
- Allowing several structural parameters to be optimised for minimum weight by minimising the margin of safety for different structure failure modes (maintaining Reserve Factor equal or just above 1.0).
- Maintaining minimum thickness requirements due to machining limitation and airworthiness lightning requirements on the fuel tanks area.

The procedure of initial sizing is shown in the following figure:

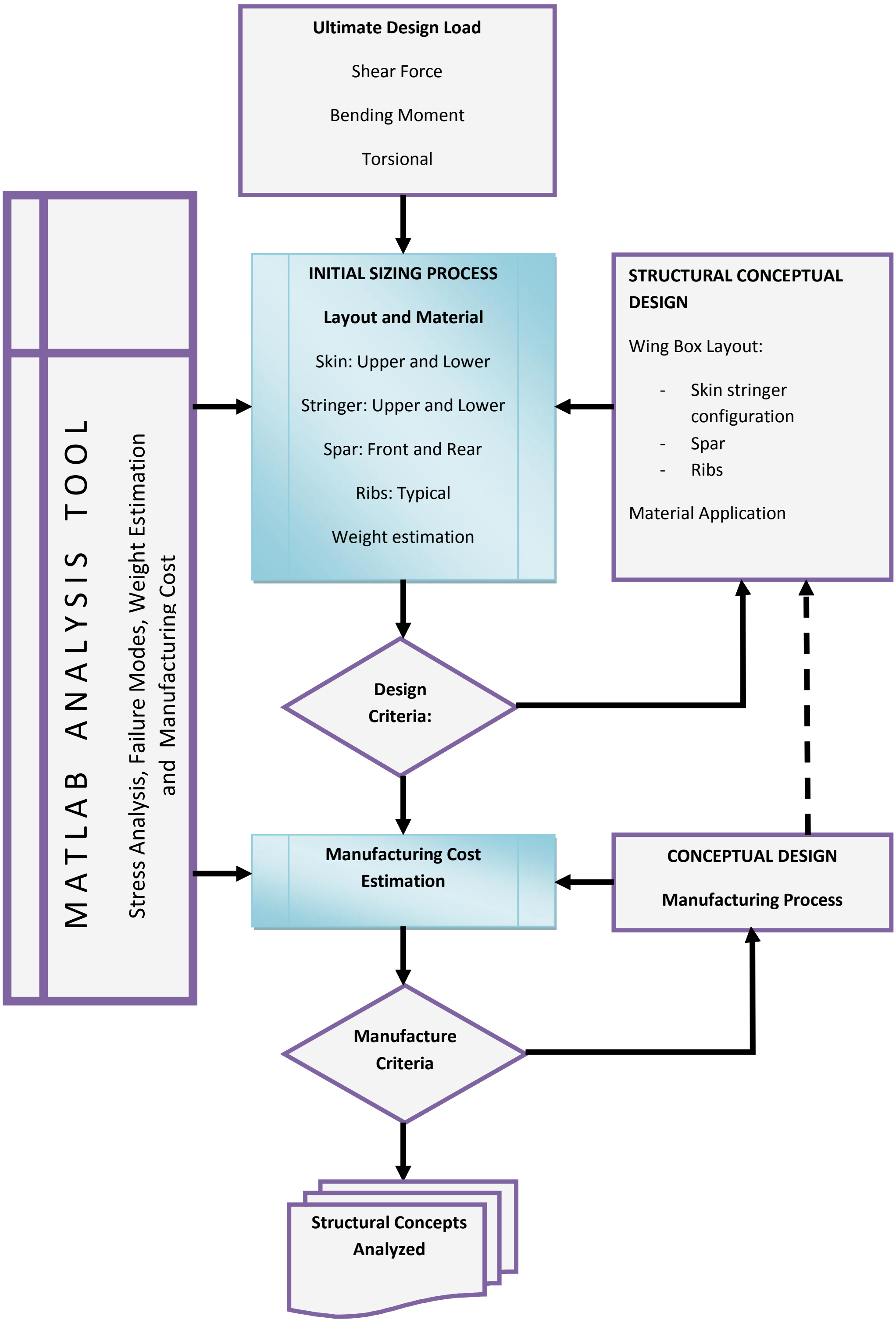


FIGURE 3-9 CONCEPTS GENERATING PROCESS AND INITIAL SIZING PROCEDURE

The main parts of wing box structure which are covered are:

- upper skin-stringer panel
- lower skin-stringer panel
- spars : front spars and rear spar
- ribs: typical

The objective of this procedure is to help the structural designer in defining the optimum initial layout and sizing of the wing box structure in the earliest stage based on the predicted loads, design configuration, design criteria, and design limitation, so that the further refinement process to obtain the optimum wing box structure will be faster.

3.4.1 Structural Design Criteria: Static Strength Criteria

The following structural design criteria are commonly used within the aircraft industry and therefore adopted during development of the approach and tool:

- ***Ultimate Loads***
  The stresses imposed by ultimate loads should be just below the failure (collapse) stresses of the structure. Such failure could be the result of material rupture, or the buckling instability of the structure.

- ***Limit Loads***
  The stresses imposed by the limit loads should not exceed the 0.2 percent offset yield stress of the material. This criterion limits the

permanent strains in the structure to 0.002 (in some cases, permanent deformation is acceptable but must not be detrimental).

- ***Limit Loads***

  The deflections at the limit loads shall not interfere with the mission of the aircraft, e.g. those which prevent the free motion of moving parts, changing the distribution of external or internal loads.

From an instability point of view, local buckling is not an important factor to the ultimate strength of the structure. However, for some external structure components, such as the upper and lower panels of wing-box, local buckling of the skin is not allowed due to aerodynamic problems. For other structure components such as spars, local buckling of the skin is also not allowed due to leakage problems of the fuel and the functionality of the systems (control systems, hydraulic systems, etc) attached to the spar. Nevertheless, stress redistribution takes place over the entire structure when buckling of the skin panels occurs. In this case, we assume no local buckling is allowed for all of the evaluated structures.

In addition to this, according to Advisory Circular No. 20-53 (FAA, 1985), in the integral fuel tank area, the skin thickness should not be less than 0.08 in. (2 mm).

3.4.1.1 Wing Box Structural Components

The following structural components are designed as the following:

- Stiffened panel, i.e. wing skin in tension, compression combined with shear load;
- Shear webs, i.e. spar webs and heavy ribs, which may be either transversely stiffened or unstiffened plate webs;
- Standard wing ribs, either stiffened plate webs or unstiffened plate webs;
- End load carrying member, spar flanges, and local reinforcement.

The allowable stresses are based on the following considerations:

- Material allowable stresses
- Initial buckling criteria
- Flexural buckling

3.4.1.2 Upper skin-stringer panel

The buckling criteria for the combination stresses can be calculated by the following equation (Niu, 1999):

$$\frac{\sigma_{comp}}{(\sigma_{comp})_{CR}} + \left(\frac{\tau_{xy}}{(\tau_{xy})_{CR}}\right)^2 \leq 1 \qquad 3\text{-}15$$

where :

$(\sigma_{comp})_{CR}$ is the critical buckling stress in compression, which is the smallest value of:

- Allowable yield stress of material
- Compression local skin buckling
- Crippling stress

$(\tau_{xy})_{CR}$ is the critical buckling stress in shear, which is the smallest value of:

- Allowable shear stress of material
- Shear local skin buckling

3.4.1.3 Lower skin-stringer panel

Lower skin panel is sized based on maximum-distortion-energy yield (failure) criterion under tension and shear stresses (Craig, 2000)

$$\frac{\sigma_Y}{\sigma_{\max}} \geq 1.0 \qquad \text{3-16}$$

Where:

σ_Y is the yield strength of material

σ_{max} is the max applied stress under combined tension and shear

$$\sigma_{max} = \sqrt{(\sigma_x)^2 + 3(\tau_{xy})^2}$$

3.4.1.4 Front and Rear Spars Web

Spar webs are sized using shear criterion (Craig, 2000):

$$\frac{\tau_{cr}}{\tau_{xy}} \geq 1.0 \qquad \text{3-17}$$

τ_{cr} is the critical stress in shear, which is the smallest value of:

- Allowable shear stress of material
- Shear local buckling

3.4.1.5 Rib Web – typical

Typical rib web is sized due to crushing loads. The buckling criteria for the compression stress can be calculated by the following equation (Niu, 1999):

$$\frac{\sigma_{cr}}{\sigma_{crushing}} \geq 1.0 \qquad \text{3-18}$$

σ_{cr} is the critical buckling stress in compression, which is the smallest value of:

- Allowable yield stress of material
- Compression local buckling

3.4.2 Wing Box Structure Modelling and Simplification

External Loads

The design loads used for wing box initial sizing are in the form of sectional loads determined using the procedure of calculating aerodynamic load. For this initial sizing purpose, only three significant sectional loads will be taken into account; they are:

- bending moment, M
- torsional moment, T
- vertical shear load, V

These three sectional loads are assumed to be applied in the shear centre of wing box cross section.

Internal Loads – on Load Carrying Members

Wing sectional loads M, T, V are assumed to be supported by the wingbox only; leading edge and trailing edge do not take part in carrying these loads.

Bending moment, M, is assumed to be supported by the upper skin-stringer panel, lower skin-stringer panel and spar caps only. Crushing Load on skin due to bending moment, M, is assumed to be supported by the rib. The direction of the ribs is not taken into consideration.

As suggested by the method (Howe, 2004) for initial sizing stage, the actual wingbox is simplified as a symmetrical rectangular box. Therefore the compression stress on the wing skin-stringer panel due to bending moment can be calculated by flexural stress equation, as in the following:

$$\sigma_{comp} = \frac{M.z}{I_{skin+caps}} \qquad 3\text{-}19$$

where :

σ_{comp} is the compression stress

z is the distance from the neutral axis

$I_{skin+caps}$ is the moment of inertia of the wing skin-stringer panel to the neutral axis

Buckling at skin panel between 2 ribs is shown in the following figure:

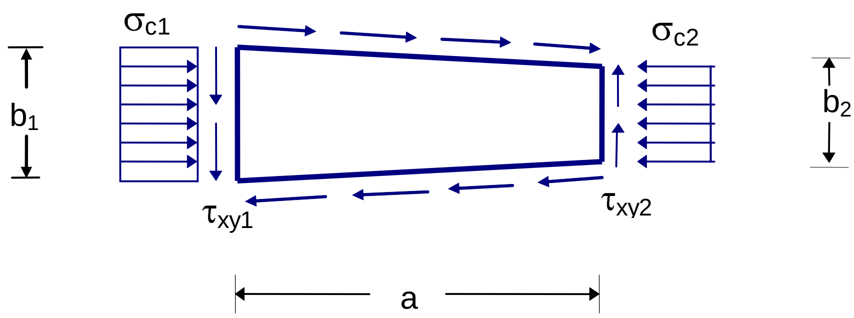


FIGURE 3-10 A TYPICAL BUCKLING PROBLEM ON THE WING SKIN

Since the actual skin plate is tapered and the magnitude of the stress is different between two edges, the simplifications have been taken for the calculation as follows:

The width of the panel: $b = \dfrac{b_1 + b_2}{2}$

The skin thickness: $t = \dfrac{t_1 + t_2}{2}$

The compression stress: $\sigma_{comp} = \dfrac{\sigma_{c1} + \sigma_{c2}}{2}$

The shear stress: $\tau_{xy} = \dfrac{\tau_{xy1} + \tau_{xy2}}{2}$

The simplified model now can be treated as the model shown in figure 3-11:

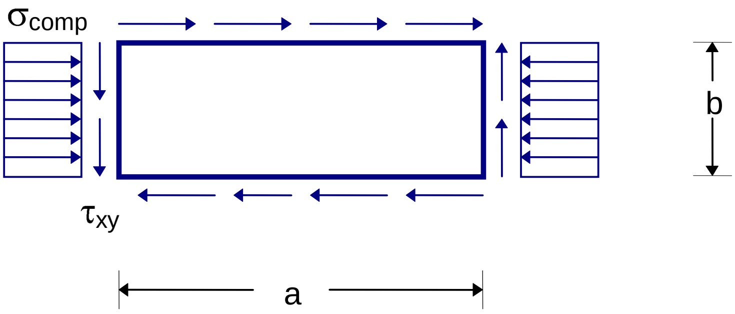
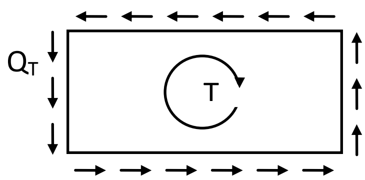


FIGURE 3-11 THE SIMPLIFIED MODEL ON THE WING SKIN

Torsional moment, T, is assumed to be supported by the skin of the upper panel, the skin of the lower panel, and the web of spars.

Using the Bredt-Batho theory, the shear force due to torque can be calculated as follow :

$$Q_T = \frac{T}{2A} \qquad (3\text{-}20)$$



Where :

T is the applied torque around the s.c. at that section (Nm)

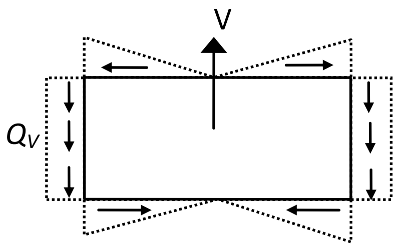
Q_T is the shear flow in the skin due to Torque (N/m)

A is the enclosed area of wing box at that section (m^2)

Vertical shear force, V, is assumed to be supported by the spar web only. The skin on the upper panel and the lower panel are not taken into account.

And, the shear flow in the spar web due to shear force, Q_V was calculated by:

$$Q_V = \frac{V}{h_T} \quad (3\text{-}21)$$



where

Q_T is the shear flow in the skin due to Torque (N/m)

V is the applied vertical shear force and (N)

h_T is the total effective depth of all the spars. (m)

3.4.3 Sizing Procedure

3.4.3.1 Initial Sizing: Upper Skin Panel

Upper skin panel is sized to meet the strength requirement of material yield or local buckling, whichever is smaller.

$$\frac{\sigma_{comp}}{(\sigma_{comp})_{CR}} + \left(\frac{\tau_{xy}}{(\tau_{xy})_{CR}}\right)^2 \leq 1 \qquad 3\text{-}22$$

By substituting the following equations into the above criterion,

Applied compression stress at upper panel: $\sigma_{comp} = \dfrac{M}{hwt_e}$

Critical local buckling strength: $(\sigma_{comp})_{CR} = \eta K_C \mathrm{E}\left(\frac{t_e}{b}\right)^2$

Applied shear stress on the upper skin: $\tau_{xy} = \frac{Q_T}{t_s}$

Critical shear local buckling strength: $(\tau_{xy})_{CR} = \eta K_S \mathrm{E}\left(\frac{t_s}{b}\right)^2$

Skin to stringer area ratio, $\frac{A_{sk}}{A_{st}} = c$ then $t_s = t_e\left(\frac{c}{1+c}\right)$

$$\frac{\eta^2 {K_c}^2 {K_s}^2 E^2}{b^4}({t_s}^3)^2 - \frac{\eta K_c {K_s}^2 EMc}{b^2 hw(1+c)}({t_s}^3) - {K_c}^2 {Q_T}^2 = 0 \qquad 3\text{-}22$$

And therefore, the upper skin thickness can be estimated from the solution to the above question:

$$(t_s{}^3) = \frac{-B \pm \sqrt{B^2 - 4AC}}{2A} \qquad 3\text{-}24$$

Where: $A = \frac{\eta^2 K_c{}^2 K_s{}^2 E^2}{b^4}$; $B = -\frac{\eta K_c K_s{}^2 EMc}{b^2 hw(1+c)}$; $C = -K_c{}^2 Q_T{}^2$

K_C is buckling coefficient under compression load

K_S is buckling coefficient under shear load

η is plasticity reduction factor

E is Young's modulus of material

t_e is effective thickness of skin-stringer panel

t_s is skin thickness

The above solution for initial sizing of skin thickness allows the designer to complete a parametric study on several parameters at once whilst always keeping the Reserve Factor is equal 1.0.

3.4.3.2 Initial Sizing: Lower Skin Panel

Lower skin panel is sized to meet the strength requirement of material yield based on maximum-distortion-energy yield (failure) criterion under tension and shear stresses (Craig, 2000)

$$\frac{\sigma_Y}{\sigma_{max}} \geq 1.0 \qquad 3\text{-}23$$

When substituting the following equation into the above

$$\sigma_{max} = \sqrt{(\sigma_x)^2 + 3(\tau_{xy})^2} \qquad 3\text{-}24$$

and $\sigma_x = \frac{M}{hwt_e}$ and $\tau_{xy} = \frac{Q_T}{t_s}$

Then the effective lower skin panel is:

$$t_{eff-lower} = \sqrt{\frac{1}{\sigma_Y{}^2}\left[\left(\frac{M}{hw}\right)^2 + 3\left(\frac{(1+c)Q_T}{c}\right)^2\right]} \qquad 3\text{-}25$$

And $$t_s = \frac{c}{1+c}(t_{eff-lower}) \qquad 3\text{-}26$$

3.4.3.3 Initial Sizing: Front Spar Web

Front spar web is sized using shear criterion (Craig, 2000):

$$\frac{(\tau_{xy})_{cr}}{\tau_{xy}} \geq 1.0 \qquad 3\text{-}27$$

τ_{cr} is the critical stress in shear, which is the smallest value of:

- Allowable shear stress of material
- Shear local buckling

As shear local buckling is normally the lesser of the two, by substituting the following equations into the shear criterion:

$$(\tau_{xy})_{CR} = \eta K_S \mathrm{E}\left(\frac{t_s}{b}\right)^2 \qquad 3\text{-}29$$

And $\tau_{xy} = \frac{Q_{FS}}{t_s}$ whereas: $Q_{FS} = Q_V - Q_T$

Then the front spar web thickness is:

$$t_{FS} = \left(\frac{Q_{FS} b^2}{\eta K_s E}\right)^{\frac{1}{3}} \qquad 3\text{-}28$$

3.4.3.5 Initial Sizing: Rear Spar Web

Similar to the sizing for front spar, the web thickness can be estimated using the following equation:

$$t_{RS} = \left(\frac{Q_{RS} b^2}{\eta K_s E}\right)^{\frac{1}{3}} \qquad 3\text{-}29$$

Where: $Q_{RS} = Q_V - Q_T$

3.4.3.6 Initial Sizing: Rib Web

Typical rib web is sized to support crushing loads. The buckling criteria for the compression stress can be calculated by the following equation (Niu, 1997):

$$\frac{\sigma_{cr}}{\sigma_{crushing}} \geq 1.0 \qquad 3\text{-}30$$

Whereas by substituting the following question:

$$\sigma_{crushing} = \frac{2\sigma^2 t_e L}{Eht_{rib}} \qquad 3\text{-}31$$

And : $$\left(\sigma_{comp}\right)_{CR} = \eta K_C \mathrm{E}\left(\frac{t_e}{b}\right)^2 \qquad 3\text{-}34$$

Therefore:

$$t_{rib} = \left(\frac{2\sigma^2 t_e L b^2}{\eta K_c E^2 h}\right)^{\frac{1}{3}} \qquad 3\text{-}32$$

3.5 Weight Variation on Fuel Cost

Reflecting the different concepts above, the effect of weight difference on fuel cost can be estimated through the following equations:

$$\Delta Fuel = T_{fuel.2} - T_{fuel.1} \qquad 3\text{-}33$$

Based on the Breguet formula, fuel consumption of the aircraft can be obtained from the following equation (Burns, 1994):

$$T_{fuel} = 1.1 * \frac{K_{FF} * TO_{GW}}{L/D} * SFC * FH_{YR} * Y_S \qquad 3\text{-}34$$

And, therefore:

$$\frac{\Delta Fuel}{T_{fuel.1}} = \frac{TO_{GW.2}}{TO_{GW.1}} - 1 \qquad 3\text{-}35$$

Hence, by substituting each element in the Breguet equation, or by comparing the structure's weight difference and fuel consumption from initial aircraft, fuel saving due to weight difference can be calculated.

In which the total fuel cost of the aircraft is:

$$TC_{fuel} = \frac{T_{fuel}}{F_{density}} * C_{fuel} \qquad \text{3-36}$$

Where:

T_{fuel} = Total mass of fuel used during the aircraft life

K_{FF} = Fuel fraction factor

L/D = Lift to Drag ratio

TO_{GW} = Take Off Weight

SFC = Specific Fuel Consumption

FH_{YR} = Flight Hour per year

Y_S = Years of Service

TC_{fuel} = Total life cycle cost of fuel per aircraft

$F_{density}$ = Fuel density

C_{fuel} = Price fuel

3.6 Manufacturing Cost Estimation

Manufacturing cost estimation procedure is developed based on a method developed by Swift (Swift & Booker, 2003). During the case study, it was found that the accuracy of the method is reliant on company data or experience of the designer in inputting the actual cost of similar manufacturing processes.

Swift and Booker proposed a method to estimate manufacturing cost based on material cost and processing cost. Manufacturing cost, M, can be estimated from the following equations (Swift and Booker, 2003):

$$M = VC_{mt} + P_c R_c \tag{3-37}$$

where:

C_{mt} = Cost of material per unit volume

V = Volume of material input to the process

P_c = Basic processing cost for an ideal part

R_c = Cost coefficient for the part design that takes into account shape complexity, material workability, section thickness, surface finish, and tolerances.

The basic processing cost of an ideal design for a particular process is affected by the following parameters:

- Equipment cost including installation
- Operating costs (labour, overheads, etc)
- Processing times
- Tooling costs
- Component demand

The above parameters are formulated in the calculation of basic processing cost by the following equation:

$$P_c = \alpha T + \beta / N \qquad 3\text{-}38$$

where:

α = cost of setting up and operating a specific process

T = cycle time in seconds to produce an ideal part

β = Total tooling cost for an ideal part

N = annual production quantity for the part

Values for α and β are based on expertise from companies specializing in producing components in specific technological areas. Whilst this method provides the data P_c against annual production quantity, N, it is suggested that the user should use their own data for their chosen process.

The values of P_c represent the minimum likely costs associated with a particular manufacturing process at a given annual production quantity.

The design dependant factors are included in the R_c term and represent how much more expensive it will be to produce a component with more demanding features than the ideal design. It is included in the following equation:

$$R_c = C_{mp} C_c C_s C_{ft} \qquad 3\text{-}39$$

where:

C_{mp} = relative cost associated with material-process suitability (workability or fabrication)

C_c = relative cost associated with shape complexity

C_s = relative cost associated with achieving minimum section thickness

C_{ft} = the higher the cost in achieving a specified surface finish, C_f or tolerance, C_t but not both

3.7 The Impact of Weight Reduction and Manufacturing Cost on Direct Operation Cost (DOC)

Direct operating costs (DOC) vary with the aircraft type and trip length (Fielding, 1999). Generally, DOC consists of the cost of ownership, fuel, oil and taxes, crew, and maintenance. In the current market, the introduction of new technology such as new configurations, advanced material, and manufacturing processes must be cost-effective in reducing DOC (Kinder, 1995).

$$\Delta DOC = \frac{\sum_{i=1}^{n} \Delta Cost_i}{DOC_{initial}} \times 100\%$$

The above requirement is used to justify both the fuel saving, due to new designs using advanced material and configuration, and manufacturing cost variation, due to labour cost, price of new material, and improved manufacturing process, toward the percentage of reduction in DOC.

The result of fuel saving analysis (eq. 3-40) and manufacturing cost assessment (eq. 3-41) are compared to the initial DOC of the aircraft and therefore:

$$\Delta DOC = \frac{\Delta fuel\cos t + \Delta manuf\cos t}{DOC_{initial}} \times 100\% \qquad \text{3-40}$$

Kinder (1995) from Douglas Aircraft Company (now part of Boeing Company) describes that a new aircraft should provide roughly a 10% improvement in operating cost to provide sales potential necessary for production commitment.

Chapter 4 Case Study

IMPLEMENTATION AND VALIDATION PROCESS

4. CASE STUDY: IMPLEMENTATION AND VALIDATION PROCESS

4.1 Overview

The following case study is performed to test the approach and tool proposed in chapter 3 to advance the process of structural design of a commercial aircraft. The discussion will be focused on the following aspects:

- Requirement assessment
- Synthesis of structural concepts layout
- Initial sizing and structural analysis
- Manufacturing cost estimation
- Concepts selection
- Structural optimisation
- Validation of the developed software for the initial sizing

The discussion in design process is arranged as the following:

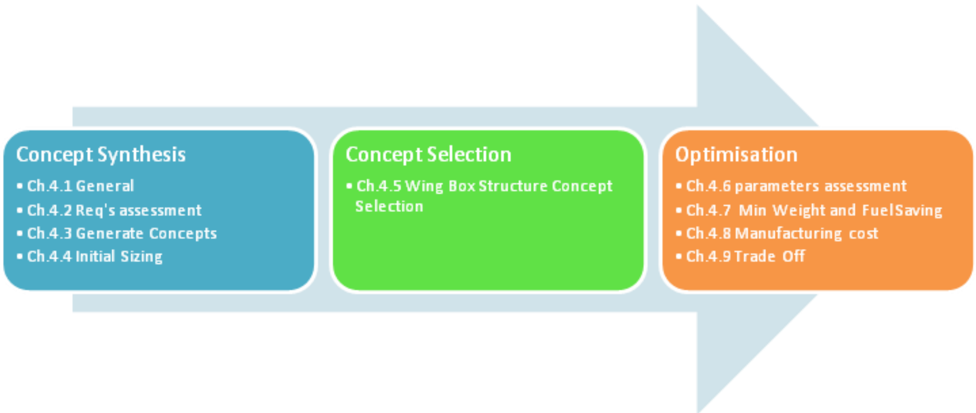


FIGURE 4-1 FLOW CHART OF CASE STUDY

During concept synthesis, the work is focused on structural layout, and material and manufacturing processes. An initial sizing procedure is employed to obtain preliminary estimation of concept characteristics to be used for analysis in the concept selection stage. During this stage, among the concepts generated, the most promising based on weight and manufacturing design criteria is selected. Additional criteria such as maintainability is also included during concept selection and analysed qualitatively. In the optimisation stage, the parametric study based on the design parameter of the selected concept is performed to obtain a set of structural parameters that meet the design target of minimum weight and cost. Validation of method and tool is performed in two stages: on stress load distribution using commercial FEA software, and on the actual mass of the existing aircraft.

4.2 Aircraft Baseline Data

The following tables and figure present aircraft baseline data for a 64/68 economy seat passenger twin engine turboprop aircraft:

TABLE 4-1 N250 AIRCRAFT BASELINE DATA (JACKSON, 1998)

Overall length	28.115 m
Overall height	8.780 m
Wing span	28.000 m
Wing area	65.000 m^2
MTOW	25,000 kg

Max Fuel weight (both)	4,000 kg
Maximum Payload	6,200 kg
OWE	15,700 kg
Maximum Cruise Speed at 6100m (20,000ft), TAS	330 Kt (611km/h; 380mph; 0.54M)
Economic Cruising Speed at 6100m (20,000ft), TAS	300 Kt (556 km/h; 345 mph; 0.49M)

TABLE 4-2 WING GEOMETRY (JACKSON, 1998)

Wing area	65.000 m^2
Wing span	28.000 m
Aspect ratio	12.1
Root chord	2.800 m
Kink chord	2.800 m
Tip chord	1.450 m
Root wing setting Incidence angle	2 deg
Dihedral angle	3 deg
Twist	3 deg

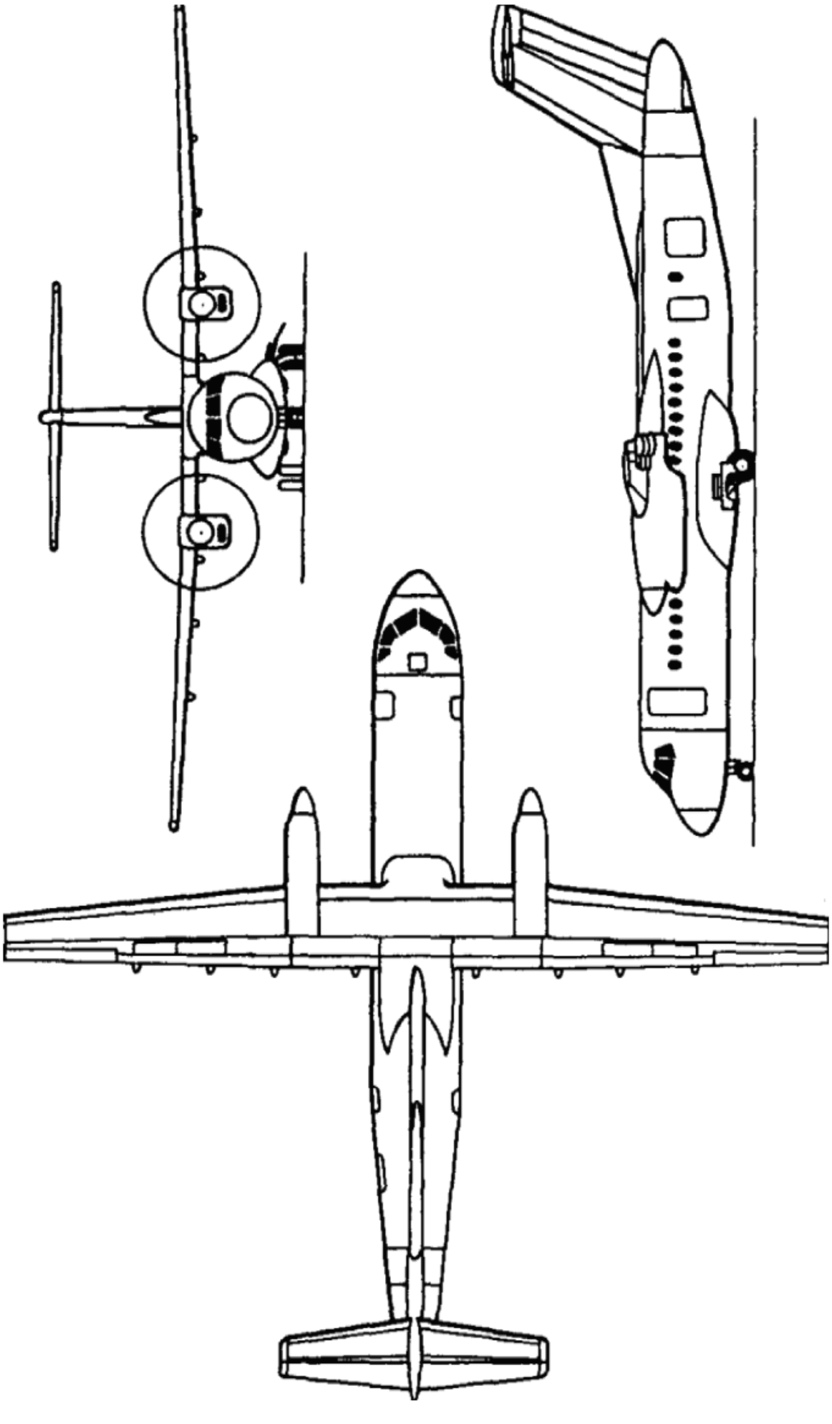


FIGURE 4-2 N250 AIRCRAFT CONFIGURATION (JACKSON, 1998)

4.2.1 Airworthiness Requirements

The airworthiness requirement for this type of commercial aircraft refers to FAR or CS chapter 25. The following section is relevant to the airframe structure and included during the case study; it includes:

- Function of structural configuration
- Materials and manufacturing process
- Joint method and assembly process
- Structure protection and maintenance access

ACDA provided information online regarding some airworthiness regulations for the above airframe requirements:

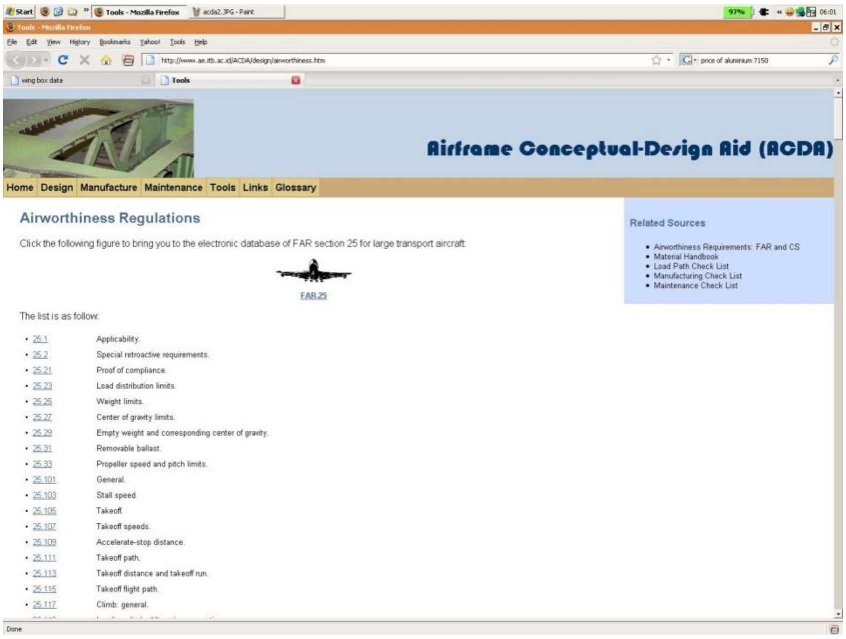


FIGURE 4-3 SCREEN SHOT ACDA: AIRWORTHINESS

By clicking the relevant section, it is summarised as in the following table:

TABLE 4-3 STRUCTURE REQUIREMENTS (EASA, 2003)

CS 25	Remark
25.601	**Structural principle**
25.603	**Material**
25.607	**Fasteners**
25.609	**Protection of structure**
25.611	**Accessibility provision**

It is therefore during the design process that each concept will be assessed against the analysis or based on in-service experience of similar aircrafts.

4.2.2 Customer requirements

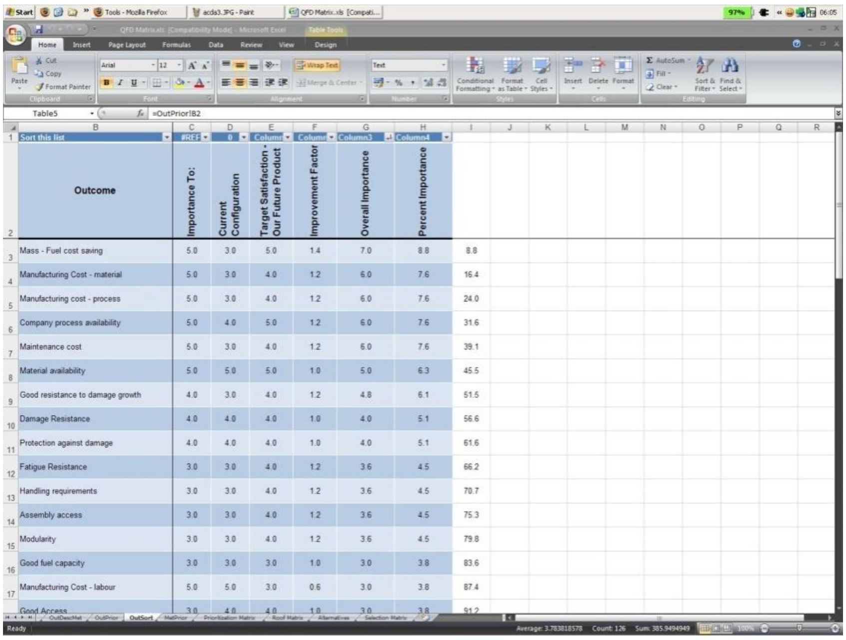
The process of capturing and translating the customer requirement into structure components and related aspects is very critical at the conceptual stage. It includes performance, manufacturing and maintenance aspects. This helped the design team to ensure the top level requirements from the customer as well as airworthiness are met throughout the design process.

TABLE 4-4 CAPTURING THE CUSTOMER REQUIREMENTS

Demanded quality	Description	Performance measure
Configuration	Advanced wing design (greater t/c than previous design)	**More efficient structure, more fuel space, cheaper to manufacture and assembly.**
Performance and mission	High technology FBW System	**Reduced critical load, lighter structure**
Weight	Extensive use of advanced production process and new and improved material to save weight	**Lower weight, cheaper to produce, more reliable, less maintenance cost**
Low operational cost	Reliable, centralised maintenance system and high part commonality within product family	**Long Fatigue life, good damage tolerance, corrosion resistance, cheaper to produce and less maintenance cost**

4.2.3 Translating Customer requirements into Design Requirements

Based on the customer requirements above, using QFD technique, these are correlated into critical design requirements. Interrelated parameters could be seen more easily as shown in the following matrix:



Outcome	Importance To:	Current Configuration	Target Satisfaction - Our Future Product	Improvement Factor	Overall Importance	Percent Importance	
Mass - Fuel cost saving	5.0	3.0	5.0	1.4	7.0	8.8	8.8
Manufacturing Cost - material	5.0	3.0	4.0	1.2	6.0	7.6	16.4
Manufacturing cost - process	5.0	3.0	4.0	1.2	6.0	7.6	24.0
Company process availability	5.0	4.0	5.0	1.2	6.0	7.6	31.6
Maintenance cost	5.0	3.0	4.0	1.2	6.0	7.6	39.1
Material availability	5.0	5.0	5.0	1.0	5.0	6.3	45.5
Good resistance to damage growth	4.0	3.0	4.0	1.2	4.8	6.1	51.5
Damage Resistance	4.0	4.0	4.0	1.0	4.0	5.1	56.6
Protection against damage	4.0	4.0	4.0	1.0	4.0	5.1	61.6
Fatigue Resistance	3.0	3.0	4.0	1.2	3.6	4.5	66.2
Handling requirements	3.0	3.0	4.0	1.2	3.6	4.5	70.7
Assembly access	3.0	3.0	4.0	1.2	3.6	4.5	75.3
Modularity	3.0	3.0	4.0	1.2	3.6	4.5	79.8
Good fuel capacity	3.0	3.0	3.0	1.0	3.0	3.8	83.6
Manufacturing Cost - labour	5.0	5.0	3.0	0.6	3.0	3.8	87.4
Good Access	3.0	4.0	4.0	1.0	3.0	3.8	91.2

FIGURE 4-4 SCREEN SHOT ACDA: THE DESIGN REQUIREMENT BASED ON THE CUSTOMER REQUIREMENT

Based on the QFD tool available in ACDA it can be seen that the configuration of advanced wing design allows for improved efficiency due to the structure’s greater thickness compared to current aircraft. The weight requirements drive the use of advanced material combined with the manufacturing techniques. The use of a reliable and easy to maintain structure is increasingly important for product competitiveness.

The priority requirement of the aircraft is to reduce the overall operating cost by reducing the weight and lessening manufacturing and assembly costs. The trade off between these two principles will become clearer during the concept selection. The availability of better maintenance practices combined with the accessibility of more reliable material automatically prioritises maintenance requirement during the decision making process or design trade off.

TABLE 4-5 PRIORITISATION MATRIX OF DESIGN REQUIREMENTS FOR FURTHER IMPROVEMENT

Outcome	Importance To:	Current Configuration	Target Satisfaction - Our Future Product	Improvement Factor	Overall Importance	Percent Importance
Mass - Fuel cost saving	5.0	3.0	5.0	1.4	7.0	8.8
Manufacturing Cost - material	5.0	3.0	4.0	1.2	6.0	7.6
Manufacturing cost - process	5.0	3.0	4.0	1.2	6.0	7.6
Company process availability	5.0	4.0	5.0	1.2	6.0	7.6
Maintenance cost	5.0	3.0	4.0	1.2	6.0	7.6
Material availability	5.0	5.0	5.0	1.0	5.0	6.3
Good resistance to damage growth	4.0	3.0	4.0	1.2	4.8	6.1
Damage Resistance	4.0	4.0	4.0	1.0	4.0	5.1
Protection against damage	4.0	4.0	4.0	1.0	4.0	5.1
Fatigue Resistance	3.0	3.0	4.0	1.2	3.6	4.5
Handling requirements	3.0	3.0	4.0	1.2	3.6	4.5
Assembly access	3.0	3.0	4.0	1.2	3.6	4.5
Modularity	3.0	3.0	4.0	1.2	3.6	4.5
Good fuel capacity	3.0	3.0	3.0	1.0	3.0	3.8
Manufacturing Cost - labour	5.0	5.0	3.0	0.6	3.0	3.8
Good Access	3.0	4.0	4.0	1.0	3.0	3.8
Detect-ability of damage	3.0	4.0	4.0	1.0	3.0	3.8

By sorting these requirements, it could be shown in the above prioritisation matrix, the priority of new design over current ones.

4.3 Concepts Generation

Several concepts of aircraft structure are designed to incorporate the features associated with the latest approach on design and damage tolerant structures. Therefore, the latest requirement in airworthiness regulation and current technology are fully utilised to meet the customer target, in which the different structural requirements of the aircraft component lead to a variety of constructions.

Loads acting on the structure and the environment where it will be operated dictate the type of configuration and material best suited for it. The synthesis process of airframe wing box utilises the above information to find the possible configuration. Airframe major structure and integration is assessed against the requirements. Advantages and disadvantages of each concept are brought to the next stage during concept selection to find optimum concept(s).

The design scenario for concept generation is firstly to gather information on similar existing aircraft and to break it down into each component related with the product breakdown structure. This is used as the concept baseline. The second step is to analyse the product to seek some possible improvement through the introduction of new concept material, process, assembly, and maintenance technology.

Lessons learned and best practises in industries on similar aircraft configuration are also utilised to generate the airframe configuration. This comprises the assessment of function, manufacturing and maintenance guide lines.

4.3.1 Load Path Design

Load distribution on wing surface is transferred to wingbox internal structures and then to the fuselage structure on wing root. Load path design combined with function requirements such as fuel tank volume, control surfaces area, and aerodynamic shape will dictate the shape of the internal structure. To have a greater understanding about the author's design space, a surface model is created in 3D Catia environment as shown in the following draft:

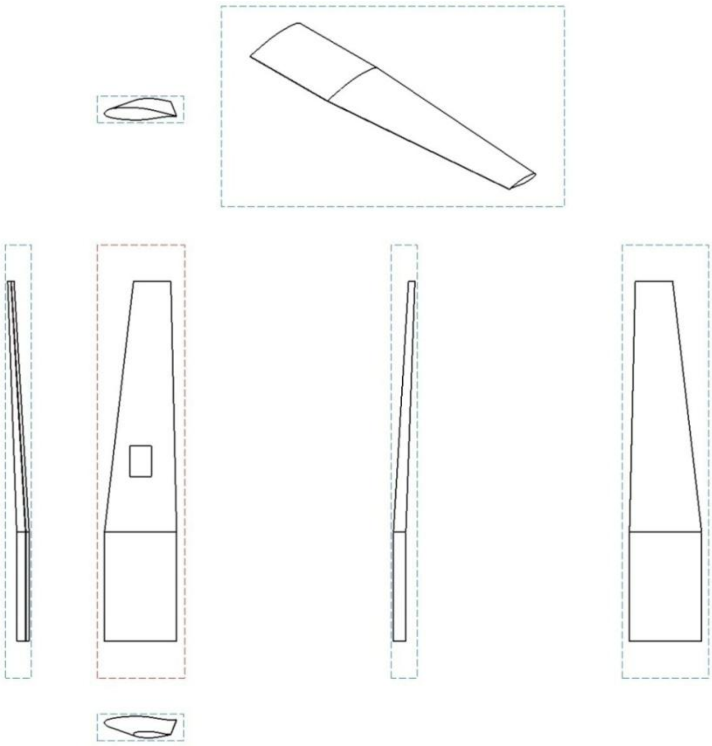


FIGURE 4-5 3D SURFACE MODEL OF WING

Later on, the result of initial sizing is used to create a more complete Catia solid model. The purpose and benefits are two-fold: the model could be used by Patran/Nastran software to perform static stress analysis in which the result will be compared with the result from developed software; additionally, the model will aid design illustration on intersection during assembly and access holes for maintenance.

4.3.1.1 Skin Panels

For this type of aircraft, when the load intensity is moderate to high, it becomes practical to use the upper and lower skins between the spars to provide a main reaction to the spanwise bending. Thus the skins are constructed to carry the end load by supporting their area with spanwise stringers. Upper and lower skin-stringer panel design is governed by the load type, i.e. compressive buckling load on the upper, and fatigue tensile dominant in the lower panel. An access panel will be required on the skin panel for maintenance purposes.

For the upper panel skin where the dominant load is compression, the buckling is becoming critical problem. ACDA provided information on material application - Boeing’s experience on its current aircraft (Airliners, 1998), for example - and supported by material data such as in Metallic Materials Properties Development and Standardization (MMPDS-01) (FAA, 2003). This is used during the material selection in this case study.

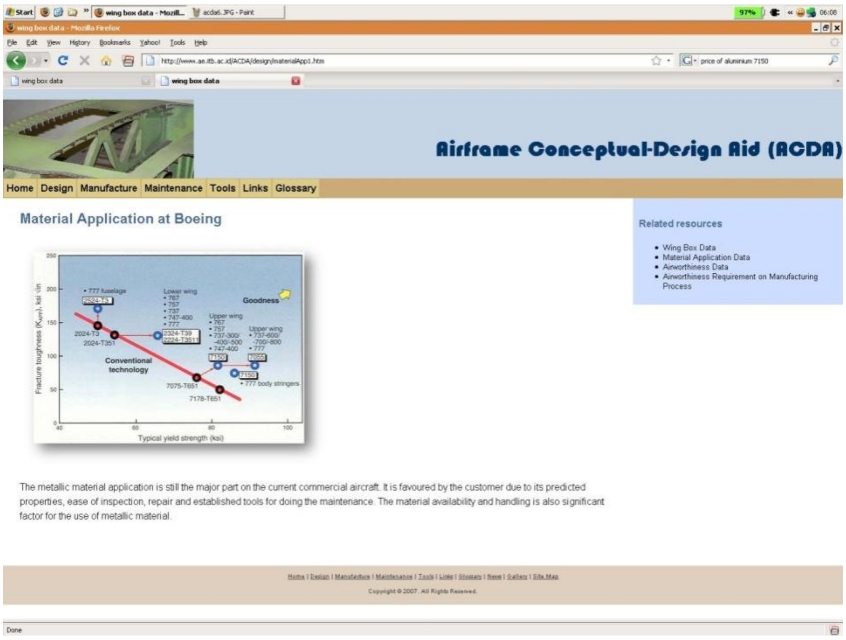


FIGURE 4-6 SCREEN SHOT ACDA: BOEING'S MATERIAL APPLICATION

Material with higher compressive strength such as 7xxx series alloy is considered.

TABLE 4-6 MATERIAL PROPERTIES (A-VALUE): ALUMINIUM 7XXX SERIES ALLOY (FAA, 2003)

Material Name		7075	7075	7150	7055
Temper		T651	T7351	T7751	T7751
Type		Plate	Plate	Plate	Plate
Raw material Thickness	(in)	2.0-3.0	2.0-3.0	2.0-3.0	0.375-1.25
Ultimate tensile Strength, f_{tu}	(MPa)	517	448	565	607
Yield tensile Strength, f_{ty}	(MPa)	455	359	524	565
Compressive yield, Proof Strength, f_2	(MPa)	427	345	517	558
Ultimate Shear Strength, f_s	(MPa)	303	269	324	338
Modulus Elastic, E	(MPa)	7.10E+04	7.10E+04	7.10E+04	7.10E+04
Density, ρ	(kg/m^3)	2.80E+03	2.80E+03	2.82E+03	2.82E+03

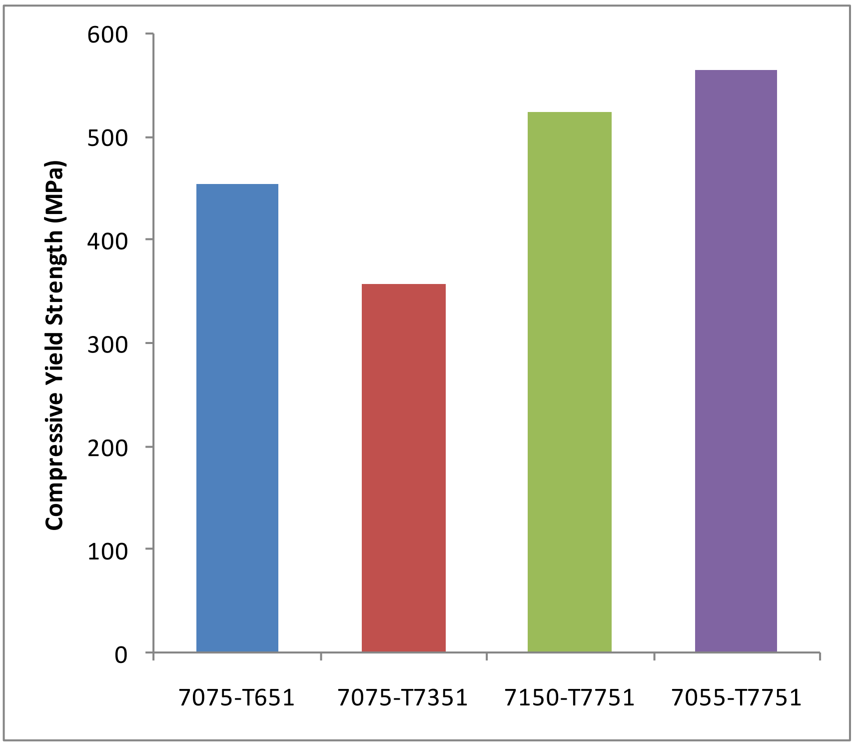


FIGURE 4-7 COMPRESSIVE YIELD STRENGTH OF 7XXX SERIES ALLOY (FAA, 2003)

TABLE 4-7 COMPARISON OF FRACTURE TOUGHNES (AT ROOM TEMP) AL-7055 AND AL-7150 (PH. LEQUEU ET AL., 2001)

	1 in. (25.4mm) Plate		1 in. (25.4mm) Extrusion	
Kic: ksi√in (MPa√m)	7150-T7751	7055-T7751	7150-T77511	7055-T7751
L-T	27 (29.7)	26 (18.6)	27 (29.7)	30 (33.0)
T-L	24 (26.4)	24 (26.4)	22 (24.2)	25 (27.5)

The AL-7150 and AL-7055 are suggested for the upper panel as it has better compression yield strength and is also more resistant to corrosion than AL-7075. In addition, it has similar fracture toughness values as the AL-2024. However, the AL-7055 has limited thickness for integral skin-stringer panel that requires raw material of more than 1.25 in.

For the lower skin, which is mostly under tension, the following material in 2xxx series is considered as it has good fatigue life and better damage tolerance than other series.

TABLE 4-8 MATERIAL PROPERTIES (A-VALUE): ALUMINIUM 2XXX SERIES (FAA, 2003)

Material Name		2024	2024	2124	2324
Temper		T351 Bare	T351 Clad	T851	T39
Type		Plate	Plate	Plate	Plate
Raw material Thickness	(in)	2.0-3.0	2.0-3.0	2.0-3.0	0.75-1.30
Ultimate tensile Strength, f_{tu}	(MPa)	414	400	448	476
Yield tensile Strength, f_{ty}	(MPa)	290	276	393	372
Compressive yield, Proof Strength, f_2	(MPa)	255	241	365	365
Ultimate Shear Strength, f_s	(MPa)	241	234	255	255
Modulus Elastic, E	(MPa)	7.38E+04	7.38E+04	7.17E+04	7.17E+04
Density, ρ	(Kg/m^3)	2.77E+03	2.77E+03	2.77E+03	2.77E+03

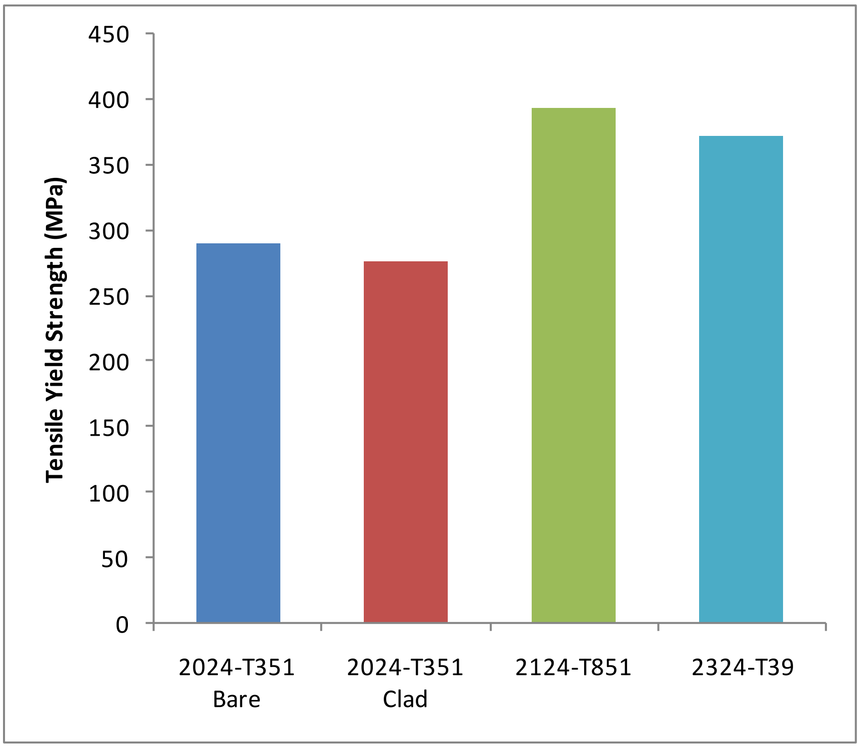


FIGURE 4-8 TENSILE YIELD STRENGTH OF 2XXX SERIES ALLOY (FAA, 2003)

TABLE 4-9 FRACTURE TOUGHNESS VALUES, KIC - KSI√IN (MPA√M) OF 2XXX SERIES ALLOY (PH. LEQUEU ET AL., 2001)

Test direction	L-T	T-L	S-L
2024-T851 Typical	22 (24.2)	20 (22.0)	17 (18.7)
Not Guaranteed	-	-	-
2124-T851 Typical	29 (31.9)	24 (26.4)	24 (26.4)
Guaranteed minimum 1.5-6.0 in. (38.1-127.0 mm) thickness	24 (26.4)	20 (22.0)	18 (19.8)

Alloy K_{ic}:	ksi√in (MPa√m)
2324-T39 0.750-1.300 in. (19.05-33.02 mm) thickness	35-40 (38.5-44.0)
2024-T351 0.500-1.000 in. (12.70-25.40 mm) thickness	34 (37.4)

It appears that the AL-2324 has the highest yield strength and better damage tolerance than the AL-2024 or AL-2124. However, it has limited thickness to be machined for integral skin-stringer panel and thus during analysis, it is only limited to build-up skin-stringer panel.

4.3.1.2 Front and rear spar web and caps

The front and rear spar are designed to have a built-in integral crack stopper and crack retarder. These reduce the stress level and the rate of crack propagation. In addition, a crack retarder should prevent any crack which may occur, from joining up across the frame position. Holes in the spars are provided for the slat track and the spoiler and the airbrake actuator passage. These spars holes also have integral reinforcing structure.

Considering the load types and environment where the part will be operated, for the front and rear spar web, material similar to that used for the upper skin panels is considered. Stiffeners across vertical and horizontal and on spar webs between ribs are employed to improve buckling strength whilst keeping the weight minimum.

4.3.1.3 Rib Web

Since the typical webs are supporting crushing load, buckling is major problem to these parts. Similar materials used for upper skin panels, i.e. 7xxx series, are considered. Stiffeners across vertical and horizontal rib web are employed to improve buckling strength whilst keeping the weight minimum.

4.3.1.4 Structural Joints

Structural joints are mainly provided by slug automatic riveting, but for critical fatigue areas and high stress areas, interference bolts such as hilok bull nose are used.

Experience from previous aircraft is utilised during the joint design; it shows a typical crack on the front spar fitting. The use of tension joint on the front spar is found to be sensitive on crack due to stress corrosion combined with load fatigue.

4.3.2 Manufacturing Process

The skin panel curvature is required by the aerodynamic profile definition. The construction is achieved using a combination of incremental forming by mechanical press and compound forming by shot peening. The top and lower skin panel are machined and where possible pocketed to save weight.

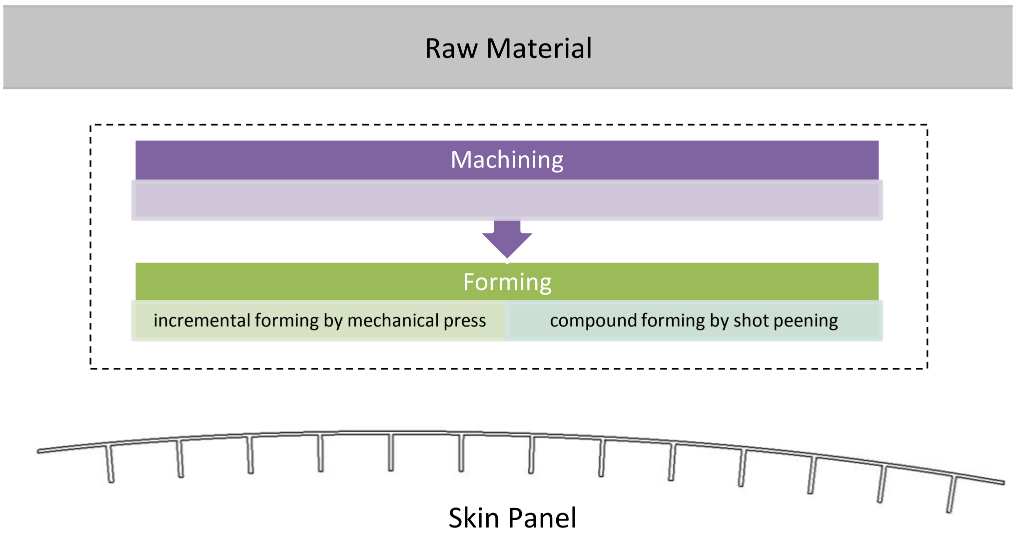


FIGURE 4-9 MANUFACTURING PROCESS

Each spar comprised an inboard and outboard section and are joined near the wing-kink section. Inner sections of both front and rear spars are integrally machined from a forged stretched plate and the outer section from a rolled stretched plate.

4.3.3 Maintenance Access and Structure Protection

4.3.3.1 Access

Wing access holes are provided in the skin panel and must be large enough for a person to enter to inspect and even reseal the inside if necessary. On the shallow wing section, access has also to be available in the lower surface to be acceptable for maintenance people to work in, even if they cannot climb in completely. Apart from the sealing problems associated with the lower access panel, it is primarily a tension skin and so introduces stress concentration in an area where crack propagation is a major consideration. In response, man-hole doors are machined elements and non-load carrying, except for load carrying doors in the outer wing. A non-load carrying door consists of an inner sealed door and outer door shaped to the wing profile.

4.3.3.2 Corrosion, damage, accident

Specific consideration is given to areas of high contamination and high condensation, where anodic corrosion between different materials could occur. Aluminium material is treated with Alodine or Chromic acid. Areas which are subject to contamination by aggressive fluids are primed and painted with primer and top coat which are resistant to the fluid. To avoid water accumulation, the drain holes are provided in the critical areas.

4.3.4 Wing Box Concepts

During this stage, the designer could apply information on current wing box configuration available in ACDA to generate concepts.

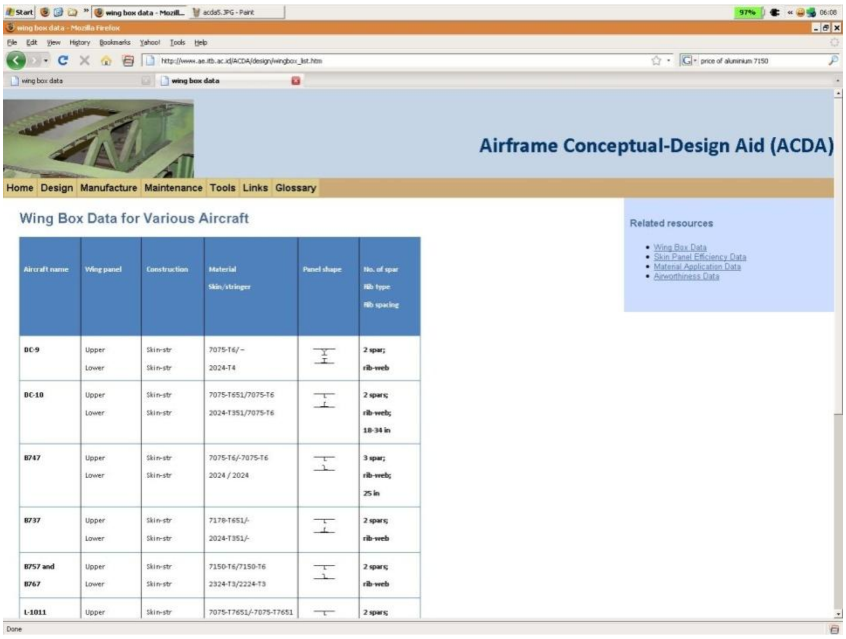


FIGURE 4-10 SCREEN SHOT ACDA: WING BOX CONCEPTS

Based on the above information, a few concepts are carried through to the next step during the design process to undergo more detailed assessment before the selection stage. These are:

- 2 Integral structure, blade and "J" type all metallic material
- 2 Build Up "Z" type structure
- 1 Co-cured composite structure
- 4 different materials for upper panels, spars and ribs;
- 4 different materials for lower skin
- 2 manufacturing process

Wing skin panel is a primary structure and contributes a large degree of the total weight. However, the use of advanced metallic materials and processes could be the solution to weight and cost reduction. Advanced metallic material is aimed to compete with composite on specific weight ratio. In addition to it, several manufacturing processes are being developed to minimise the number of parts.

4.4 Wing Box Structure Concept selection

Based on weight criteria, there are 5 candidates for wing box structure:

- Concept-1: the concept datum, a built-up skin-stringer panel based on the standard aluminium material and manufacturing processes.
- Concept-2: a built-up skin-stringer panel, utilising advanced aluminium material.
- Concept-3: an integral machined skin-stringer panel, utilising advanced aluminium material; after machined, the panels are mechanically formed and shotpeened. Based on manufacturing cost, the difference between blade and J integral panel are almost insignificant. The cost difference is due to processing time and additional tools required to create flange. The raw material volumes are still the same and also the cost of different materials is assumed to be the same.

- Concept-4: an integral machined skin-stringer panel, utilising advanced aluminium material; after machined, the panels are treated with an auto clave process to achieve the final form and strength, and minimise the exfoliation corrosion due to the machining processes.

- Concept-5: Wingbox, a composite structure in which components are fabricated individually and joined via subsequent bonding or co-bonding processes.

Table 4-10 Concept selection

	Concept-1	**Concept-2**	**Concept-3**	**Concept-4**	**Concept-5**
Upper Panel	"Z" Build Up	"Z" Build Up	Integral	Integral	"J" Co-cured
Material-upper	AL7075-T651	AL7055-T7751	AL7075-T651	AL7150-T7751	CFC
Lower Panel	"Z" Build Up	"Z" Build Up	Integral	Integral	"J" Co-cured
Material-lower	AL2024-T351	AL2324-T39	AL2024-T351	AL2124-T851	CFC
Spars	Integral AL7075-T651	Integral AL7055-T7751	Integral AL2024-T351	Integral AL7150-T7751	CFC Co-cured
Ribs	Integral AL7075-T651	Integral AL7055-T7751	Integral AL2024-T351	Integral AL7150-T7751	CFC Co-cured

During design selection processes, the candidates are assessed separately using 3 criteria:

- Performance
- Manufacturing
- Maintenance

The combination of concepts are then assembled and assessed as a whole using the design and manufacturing checklist available in ACDA:

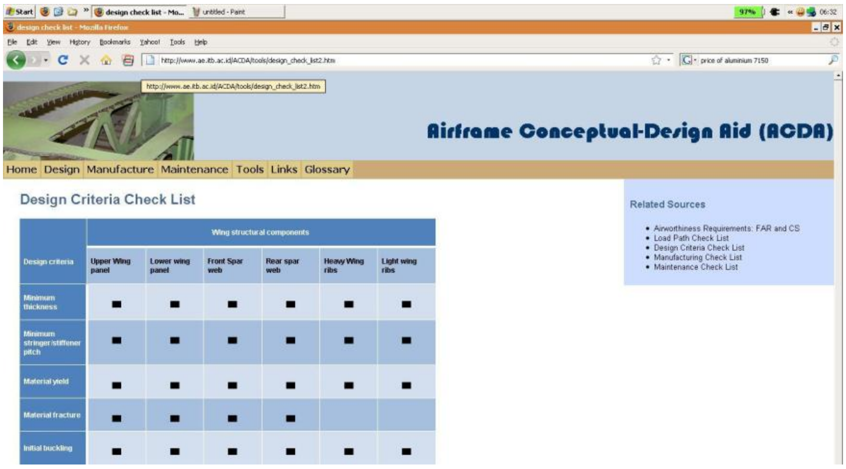


FIGURE 4-11 SCREEN SHOT ACDA: DESIGN CHECK LIST

The above design check list is then used to assess concepts based on Pugh's concept selection technique:

TABLE 4-11 CONCEPT SELECTION BASED ON PUGH'S TECHNIQUE

Design criteria	Concept 1	Concept 2	Concept 3	Concept 4	Concept 5
Performance:	DATUM				
Mass - Fuel cost saving		+	+	+	+
Good fuel capacity		S	+	+	+
Fatigue Resistance		+	+	+	+
Damage Resistance		+	+	+	+
Manufacture:					
Manufacturing Cost - material		-	-	-	-
Manufacturing Cost - labour		S	+	+	-
Manufacturing cost - process		S	+	-	-
Material availability		-	-	-	-
Company process availability		S	S	-	-
Handling requirements		S	S	S	-
Assembly access		S	+	+	+
Maintenance:					
Maintenance cost		+	+	+	-
Good Access		S	S	S	S
Good resistance to damage growth		+	+	+	+
Protection against damage		+	+	+	+
Detect-ability of damage		S	S	S	-
Modularity		S	-	-	-
Repair-ability		S	S	S	+
Total Rating:					
$\sum +$		**6**	**10**	**9**	**8**
$\sum -$		**2**	**3**	**5**	**9**
$\sum S$		**10**	**5**	**4**	**1**

From the above assessment it appears that concept-1 is inferior as four other concepts have better performance in terms of configuration, manufacturing, and maintenance. Concept-1 is therefore removed from the selection.

The composite wing box, concept-5, has the potential to make an improved structure. However, the risk estimation of new technology for the company is still high. The reduced risk of using advanced aluminium and manufacturing processes provides a more realistic option. It is based on the estimation that the company and supplier will be able to provide it during product development phase with the test result being available to confirm it.

Concept 4 is therefore the most feasible concept from a design and manufacturing point of view to proceed to the next stage where more detailed analysis on structural and manufacturing parameters will be performed. However during the initial sizing and parametric synthesis, all the concepts, except composite structure, are analysed in term of performance and DOC. The result are shown in section 4.6.

4.5 Initial Sizing and Mass Estimation

The wing box structure outboard of the wing root consists of a machined skin-stringer panel, machined front and rear spars together with 25 machined ribs. All are made from aluminium alloy material. The fuel tank extends from the rib at the wing root to rib 14.

The following figure shows the general arrangement of the wing box structure:

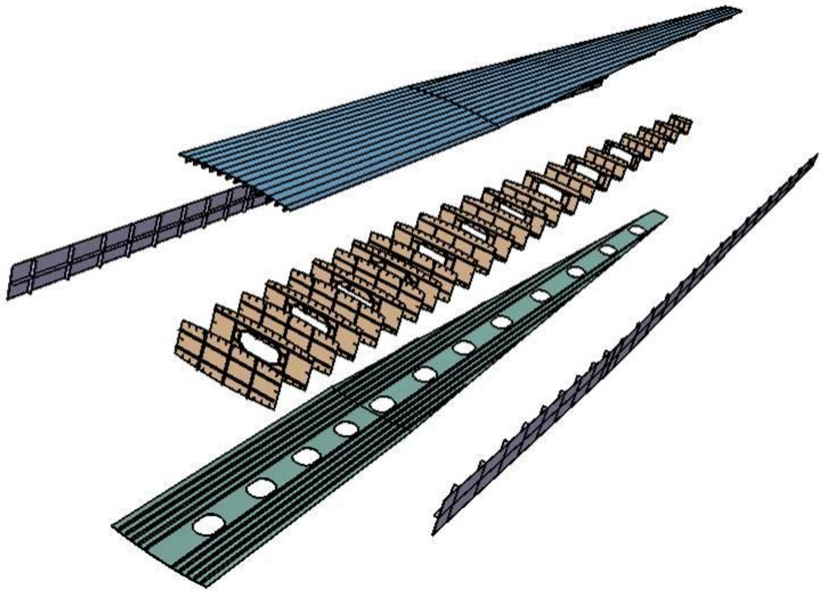


FIGURE 4-12 GENERAL ARRANGEMENT OF WING BOX STRUCTURE

Loads acting on the structure and environment where it will operate dictates the type of configuration and material best suited for the skin panel.

4.5.1 n-V Manoeuvre and Gust Diagram

One requirement according to CS25.333/335/337 on constructing Flight Manoeuvring Envelope and CS25.341 for Gust Loads, is to create the critical load factor boundary for the aircraft designed. By the superposition of manoeuvre and gust envelopes, maximum and minimum vertical load factor on the aircraft structure is then determined. It is shown that at cruise speed, V_c, positive load factor due to gust is more critical than the manoeuvre, i.e. $n_z^+ = 2.54$. For negative load factor, it is critical due to manoeuvre requirements which is $n_z^- = -1.0$

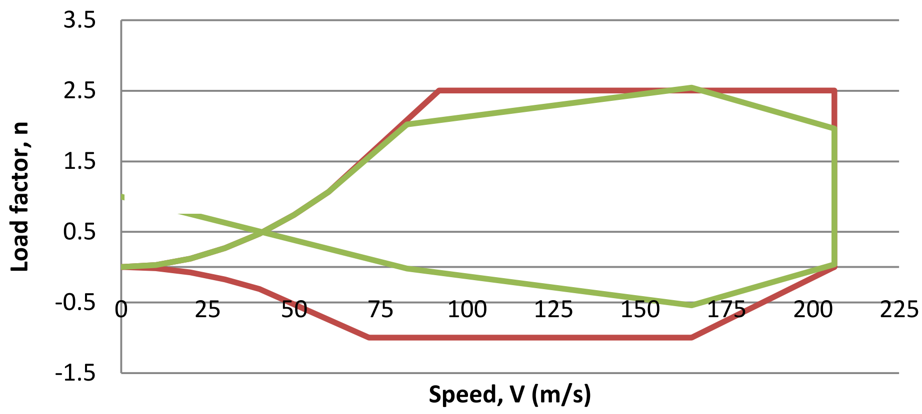


FIGURE 4-13 MANOEUVRE AND GUST ENVELOPES

n-V diagram and wing aerodynamic load distribution is calculated by employing a small program developed in ACDA as shown in the following:

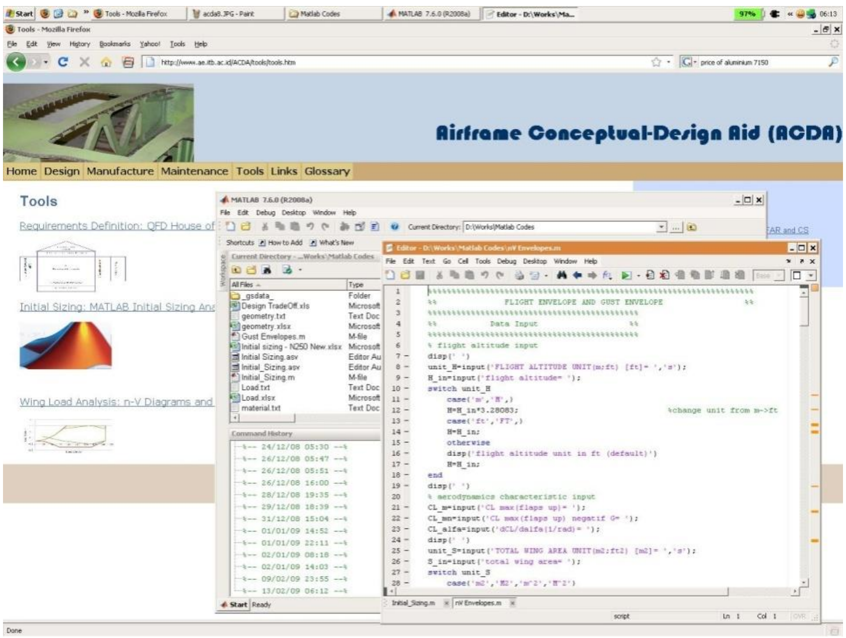


FIGURE 4-14 SCREEN SHOT ACDA: WING LOAD MODULE

4.5.2 Total Wing Load Distribution

Design load is defined as the critical load acting on the structure and therefore used in the structural design process. Wing load consists of shear force, bending moment, and torsional moment distribution of wing span. Before obtaining them, we need to calculate wing aerodynamic load and inertia relieve load due to the fuel and engine if they are placed on the wing. Once the airframe mass distribution is known, the total wing load can then be revised to include the airframe inertia load. Load factor for vertical acceleration is selected from whichever is greater between critical flight cases of manoeuvre and gust envelope.

Shear Force, Bending Moment, and Torsional Moment stress distribution is calculated from integrating small elements of forces acting on the wing box structure, taking into account external aerodynamic load, fuel, engine, and initial structure mass distribution. Figure 4-15 demonstrates the wing load distribution at MTOW, cruise speed , and n_z = 2.54. Detailed wing load calculation is shown in Appendix C.

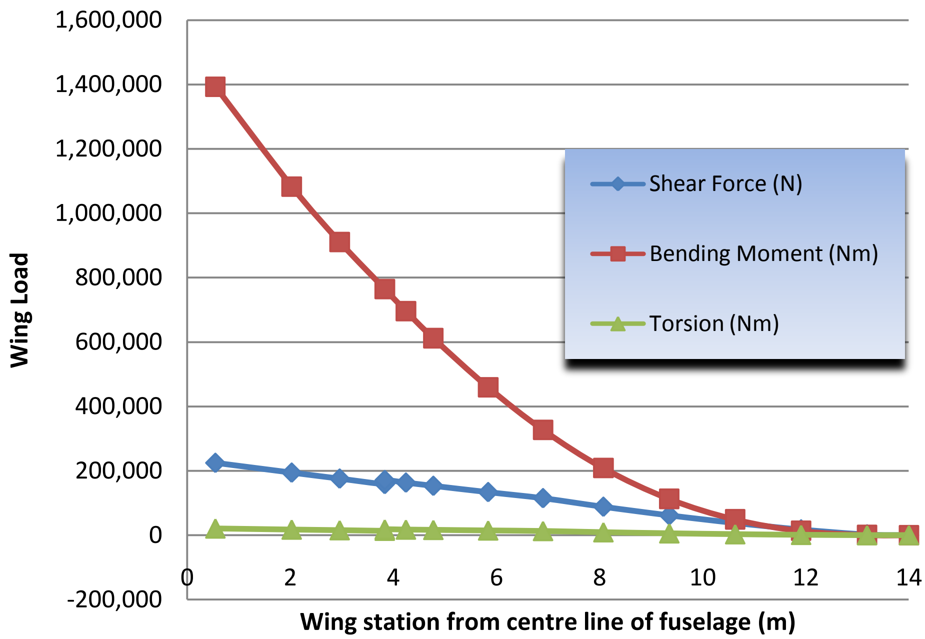


FIGURE 4-15 WING LOAD DISTRIBUTION, AT MTOW, V_C, $n_Z = 2.54$

4.5.3 Initial Sizing Result

Initial sizing is performed using the load distribution from the previous section. It follows the initial sizing procedure as laid down in chapter 3.5.4 and the design criteria, and Chapter 3.5.3, for the major components of the wing box: skin panels upper and lower; spar front and rear; and typical ribs web. The initial sizing result is then translated into the wing box's mass. The initial sizing procedure for main wing box components, such as wing skin panel, spar web and rib, was performed using a developed computer program written using MATLAB, which is available from ACDA for designers to use. An example of detailed initial sizing calculation for upper skin thickness is shown in Appendix D. The screen shot of Matlab initial sizing tool is shown below:

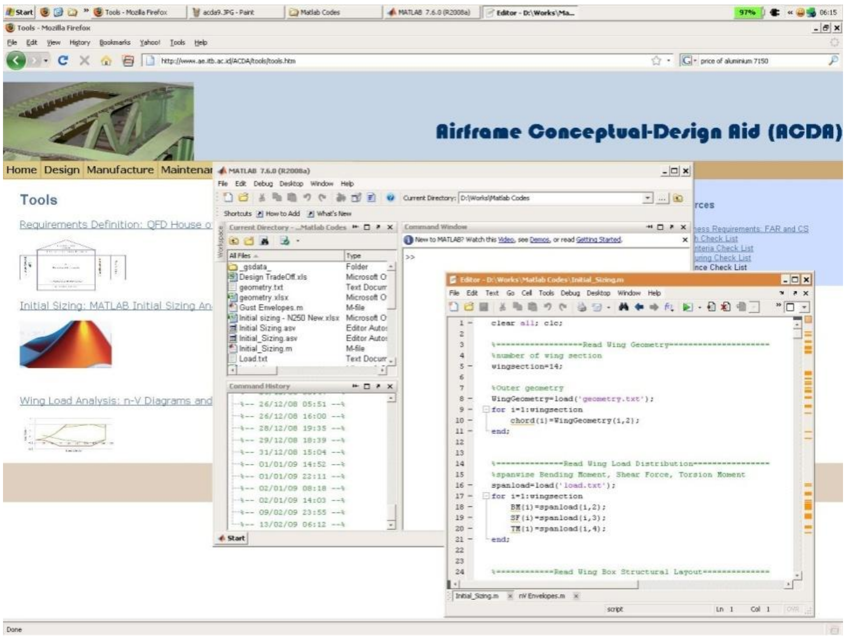


FIGURE 4-16 SCREEN SHOT ACDA: INITIAL SIZING MODULE

4.5.4 Mass Estimation Result

The mass estimation result from Matlab's initial sizing program is then inputted back into the internal load calculation as the input of the inertia force. The correction on internal load is subsequently used to refine the result of the sizing process, which implies an iterative process. From the result of case study, the airframe mass effect on the load distribution is found to be small, and therefore gives a little effect on the final result.

4.6 Parametric Study

Optimisation for minimum weight is performed around the following parameters:

TABLE 4-12 LIST OF PARAMETER FOR OPTIMISATION

Parameter	Description	Number of Combination
Material Type	4 of 7000 series 4 of 2000 series	4
Skin-Stringer Panel	Integral I, J Build-up Z	3
Stringer Pitch	0.09 – 0.16 m	8
Skin to Stringer Areas Ratio	0.5 – 2.0	4
Rib Spacing	0.3 – 0.7 m	8
No of Ribs Stiffeners	3 ver & 1 hor	1
No of Spar Stiffeners	1 ver & 1 hor	1

The above combination creates 3072 cases which are then run using the program developed.

Study 1: Material types, Rib Spacing and Skin to Stringer Areas Ratio

The following figures show the effect of rib spacing on the mass of each wing box component for four different combinations of material and skin to stringer areas ratio.

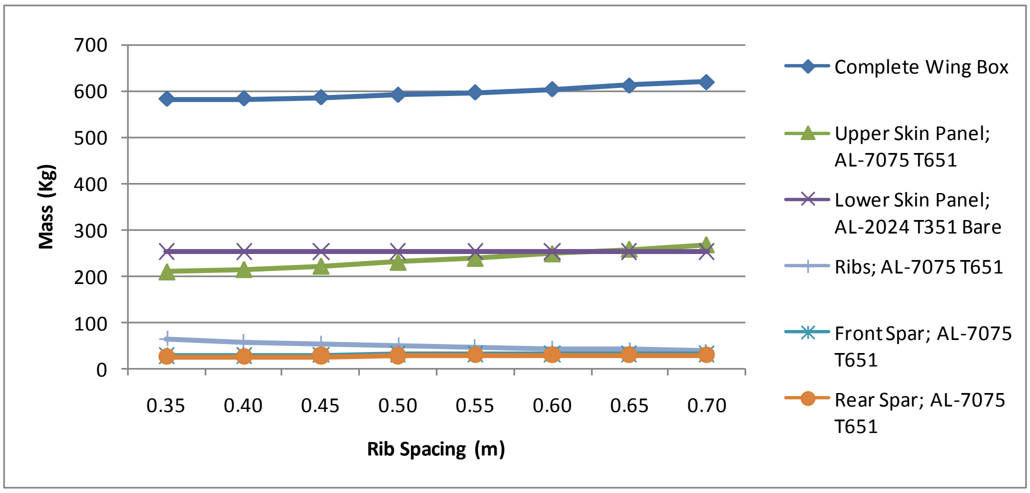


FIGURE 4-17. EFFECT OF RIB SPACING ON MASS OF WING COMPONENT STRUCTURE (*MATERIAL COMBINATION 1*)

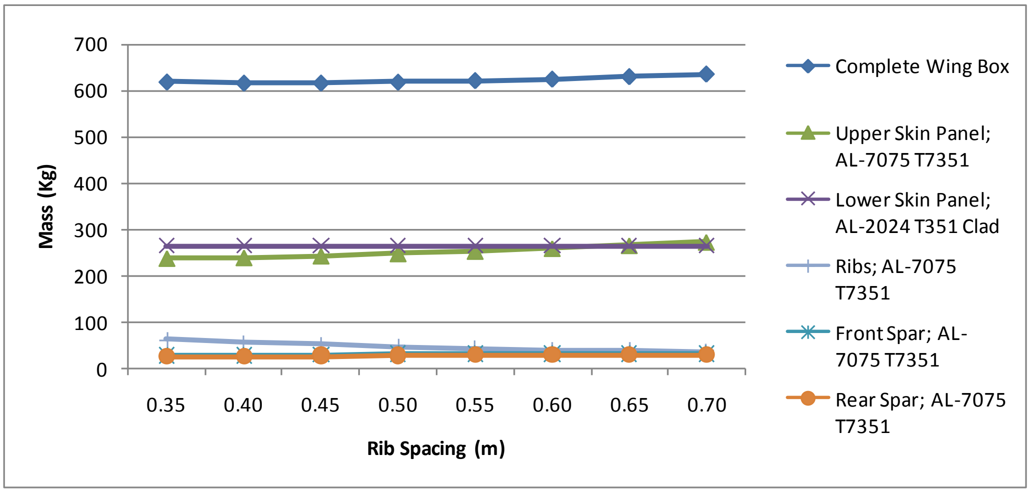


FIGURE 4-18. EFFECT OF RIB SPACING ON MASS OF WING COMPONENT STRUCTURE (*MATERIAL COMBINATION 2*)

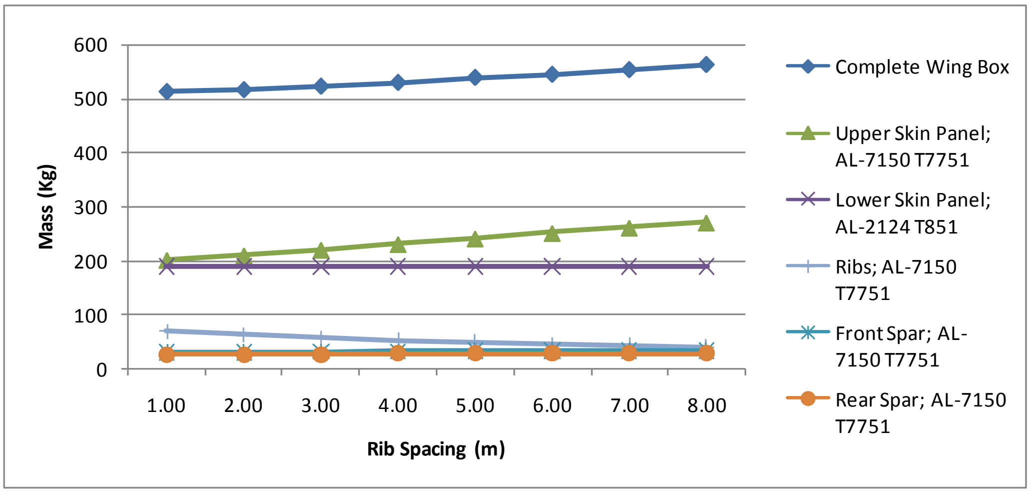


FIGURE 4-19. EFFECT OF RIB SPACING ON MASS OF WING COMPONENT STRUCTURE (*MATERIAL COMBINATION 3*)

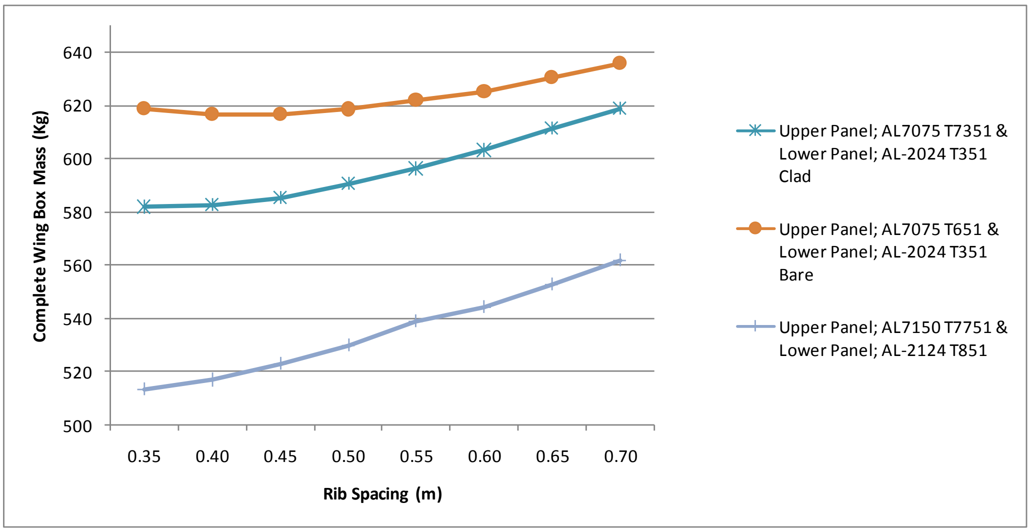


FIGURE 4-20 EFFECT OF RIB SPACING AND 3 MATERIAL COMBINATIONS ON MASS OF WING BOX STRUCTURE

The above graphs show that an increase of rib spacing will reduce the rib weight. However it is also accompanied by an increasing of upper skin panel which outweighs the weight saving of rib mass.

The lowest wing box mass is obtained when the rib spacing is around 0.4m, which is close to the result of initial sizing.

Study 2: Skin-Stringer Panel Type and Stringer Pitch

The effect of stringer type and pitch are shown below.

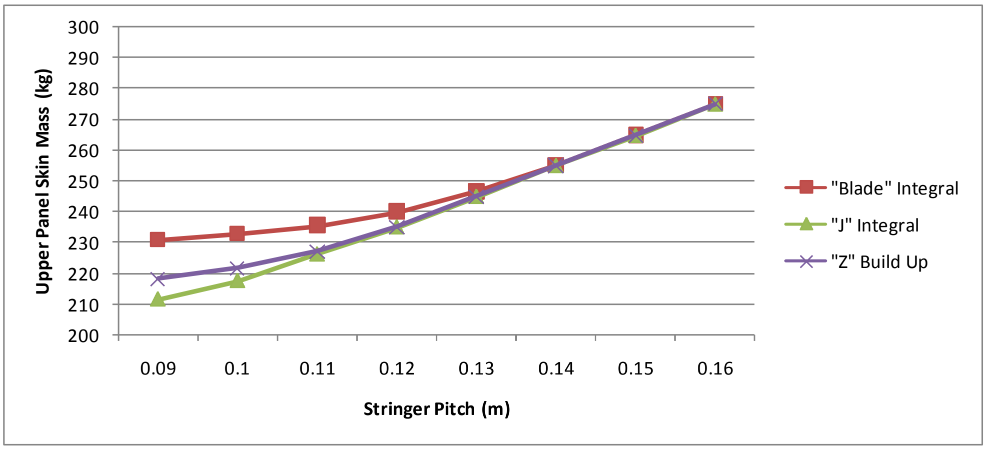


FIGURE 4-21 EFFECT OF STRINGER TYPE AND PITCH ON MASS OF UPPER WING PANEL

The above parametric study shows that the minimum weight of the upper panel is obtained using J integral panel with stringer pitch close to 90mm. The following figure shows the complete mass of wing box for 3 different stringer types and stringer pitch. The same conclusion is obtained.

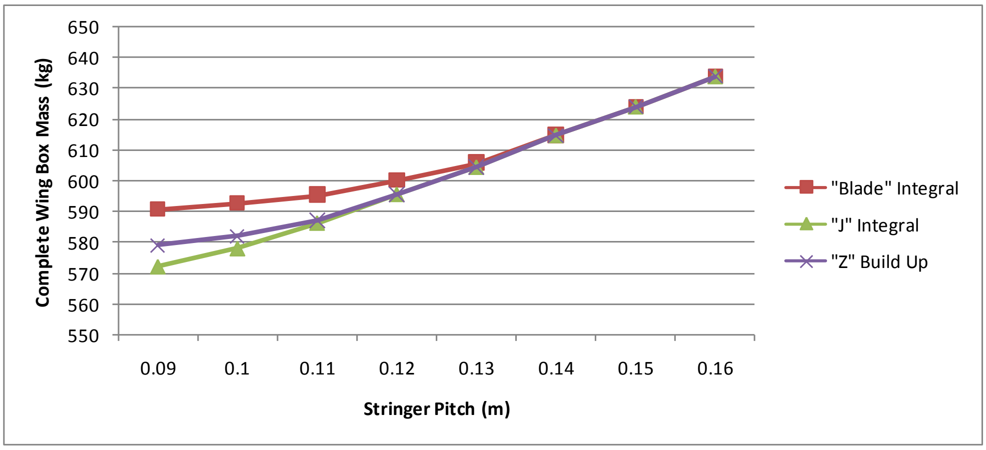


FIGURE 4-22 EFFECT OF STRINGER TYPE AND PITCH ON MASS OF COMPLETE WING BOX

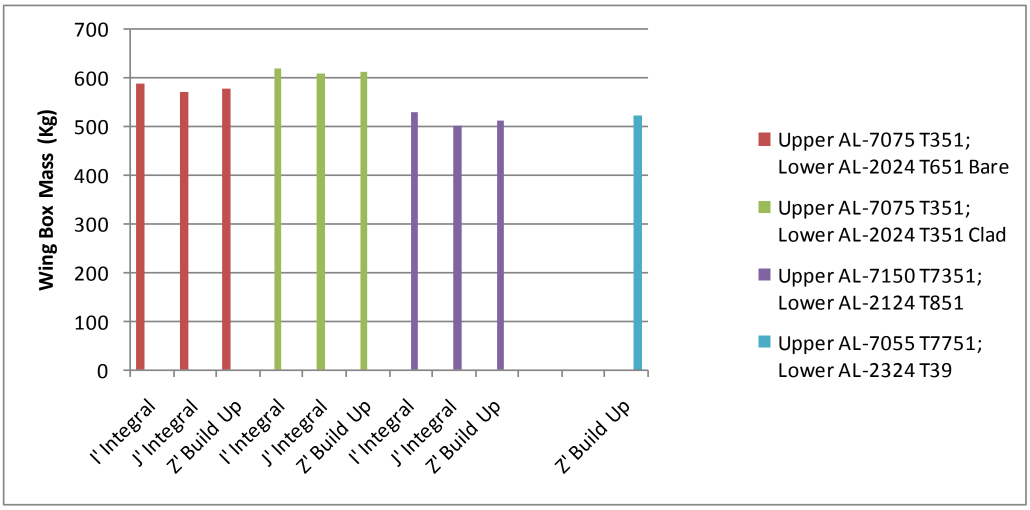


FIGURE 4-23 EFFECT OF MATERIAL AND STRINGER TYPES ON WEIGHT OF WING BOX

Since the raw materials of AL-7055 and AL-2324 in plate form have limited thickness, it can be used for the build up type but for the integral skin-stringer panel. The above figures show that minimum weight of wing box is achieved when material AL7150 and Al2124 is combined, using an integral J stringer type panel.

It is interesting to compare this result with the previous investigation on the design of the upper skin panel of typical aircraft (Niu, 2002). Niu has indicated that an integrally stiffened section can attain an exceptionally high degree of structural efficiency. A weight reduction of approximately 10-15% was realised by the use of an integrally stiffened structure. However, initial sizing on two integral skin panels and one built-up skin panel has shown differently. This can be explained as the mass estimation on the built-up panel in the case study does not include the weight of rivets, sealing and clips, which could increase the weight. However, it is important for the designers to remember this additional weight if their final decision should choose the built-up concept.

Study 3: Skin to Stringer Areas Ratio (A_{st}/A_{sk})

The effect of skin to stringer areas ratio is shown below.

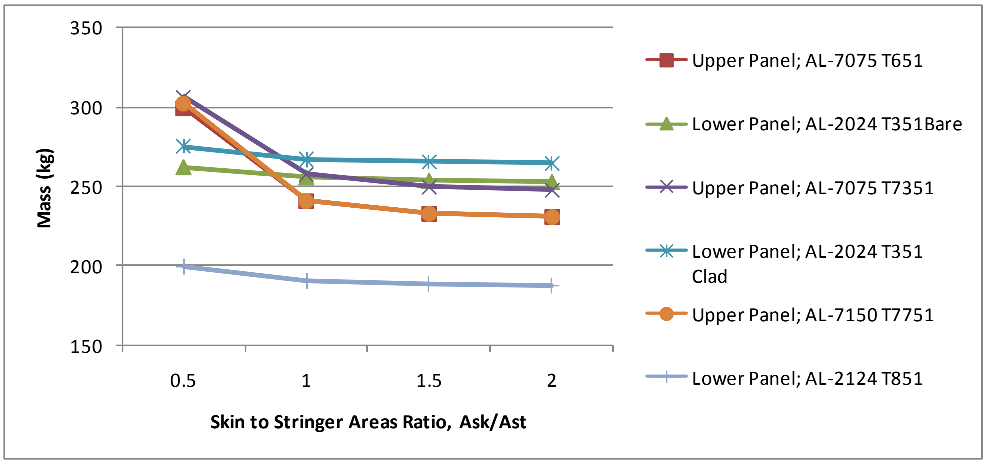


FIGURE 4-24 EFFECT OF SKIN TO STRINGER AREAS RATIO TO MASS OF SKIN PANELS

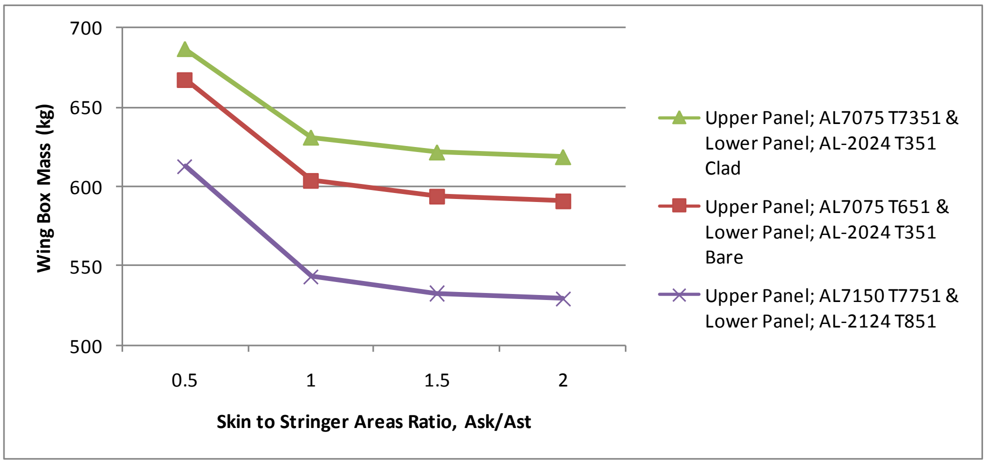


FIGURE 4-25 EFFECT OF SKIN TO STRINGER AREAS RATIO TO MASS OF COMPLETE WING BOX

Wing box structure reaches its minimum weight at skin to stringer area ratio of 2.0. It has a second benefit in that the increased thickness of skin will improve the torsional stiffness of wing box and therefore delay the possibility of flutter.

4.7 Minimum Weight and Fuel Cost Saving

From 3072 cases run, the program sorts the mass calculation for each configuration to find the minimum weight. For each material, the program gives several combinations for the airframe designer to select. These are:

TABLE 4-13 MINIMUM WEIGHT CONFIGURATION FOR 3 DIFFERENT MATERIALS

Material Type	Rib Pitch (m)	Str. Pitch (m)	Str. Type	Ask/Ast	Upper Panel (Kg)	Lower Panel (Kg)	Front Spar (Kg)	Rear Spar (Kg)	Ribs (Kg)	Wing Box (Kg)
3	0.450	0.090	2	2.0	199	188	31	27	58	**503**
3	0.500	0.090	2	2.0	203	188	31	27	53	**504**
3	0.400	0.090	2	2.0	196	188	30	26	63	**505**
3	0.550	0.090	2	2.0	210	188	32	28	50	**507**
3	0.400	0.090	3	2.0	200	188	30	26	63	**508**
1	0.500	0.090	2	2.0	212	252	31	27	50	**572**
1	0.450	0.090	2	2.0	210	252	30	27	54	**573**
1	0.550	0.090	2	2.0	216	253	32	28	47	**574**
1	0.600	0.090	2	2.0	220	252	32	28	43	**576**
1	0.450	0.090	3	2.0	213	252	30	27	54	**577**
2	0.600	0.100	2	2.0	245	265	32	28	42	**611**
2	0.450	0.090	2	2.0	238	265	30	27	52	**611**
2	0.500	0.090	3	2.0	241	265	31	27	48	**612**
2	0.450	0.090	3	2.0	238	265	30	27	52	**612**
2	0.700	0.090	2	2.0	249	265	33	28	38	**612**

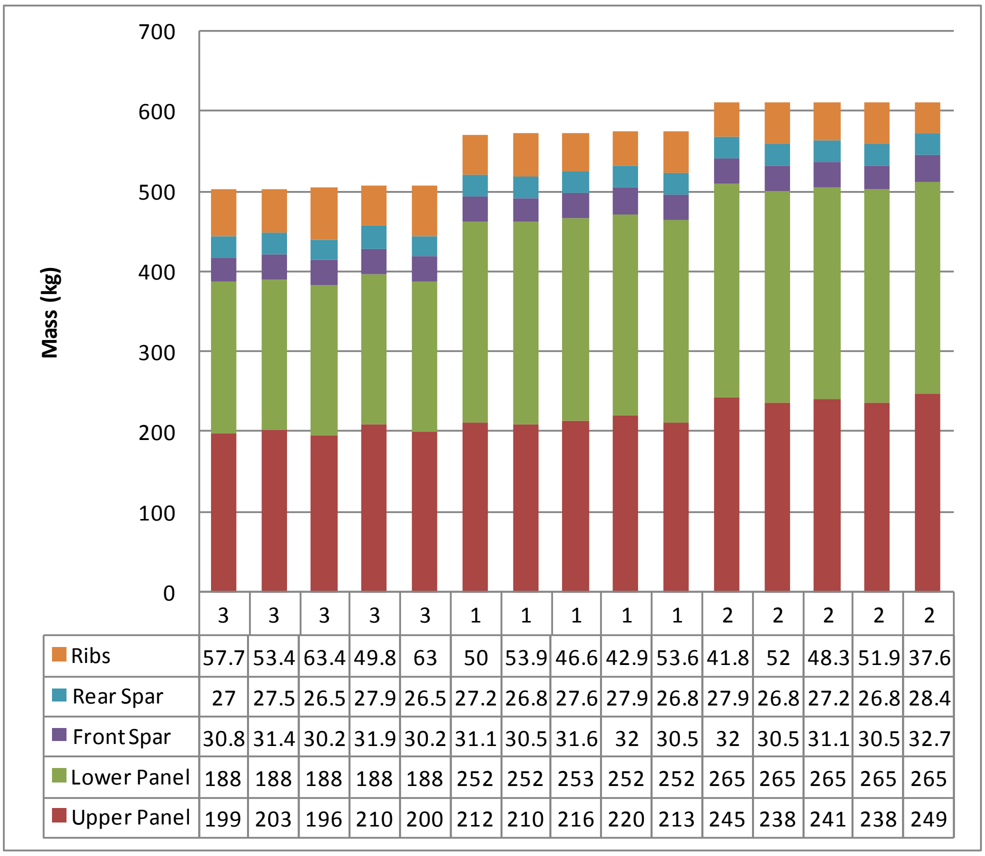


FIGURE 4-26 MINIMUM WEIGHT CONFIGURATION FOR 3 DIFFERENT MATERIALS

This aircraft is designed for 40,000-50,000 cycles during a 20-year lifespan. By taking the fuel price at $0.5/litre (IATA, 2007), the fuel cost of the aircraft is then USD460 / FH, since the maximum aircraft utilisation is from 2000-2500 FH/Yr. Therefore, the fuel saving for the entire life cycle of the aircraft for three different aircraft configurations and compared to the baseline aircraft is shown in the following figure:

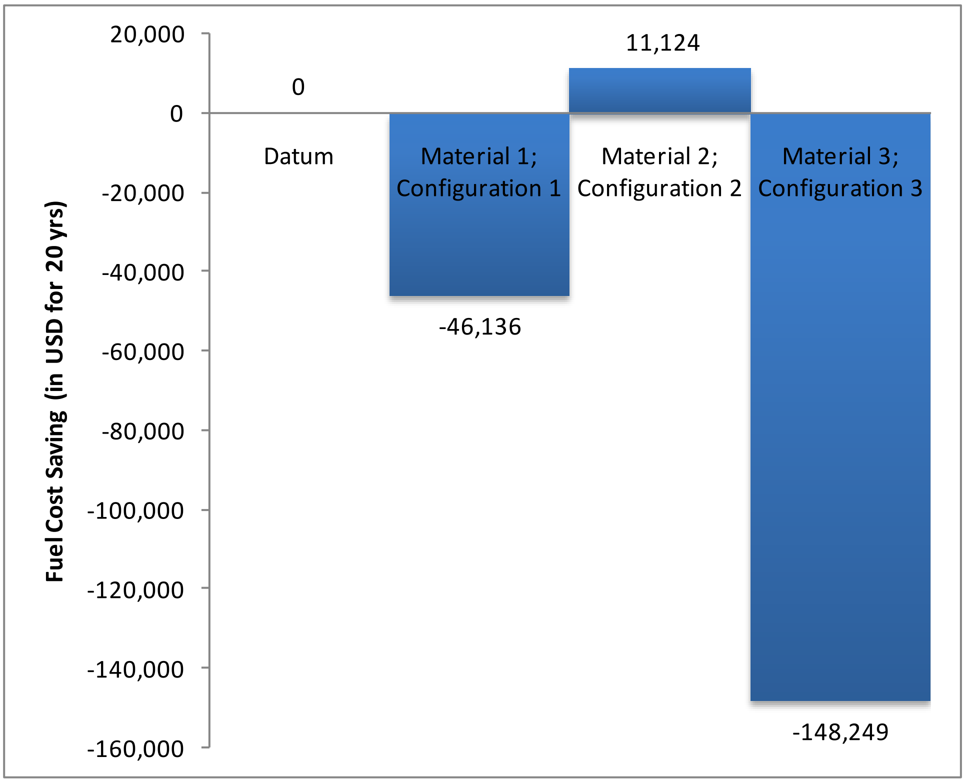


FIGURE 4-27 FUEL COST SAVING DUE TO WEIGHT REDUCTION IN WING BOX STRUCTURE (20 YEARS AND 2000FH/YR)

Configuration 3 has given the airlines a fuel saving of USD 148k during the 20-year lifecycle of aircraft or USD 7400 / yr. However, by examining the fuel price trend of constant increase, the saving will be even higher. In addition, higher fuel saving will be available to the aircraft operator once the weight reduction program is applied to the whole aircraft structure.

If the analysis is focused on the effect of new material to weight saving from the above figure, the fuel cost saving is USD 102k (or USD148k minus USD46k) for the life of the aircraft or USD 5000 / yr.

The following figure shows the fuel cost saving for configuration 3 (maximum weight saving) due the variation of aircraft utilisation in flight hours per year:

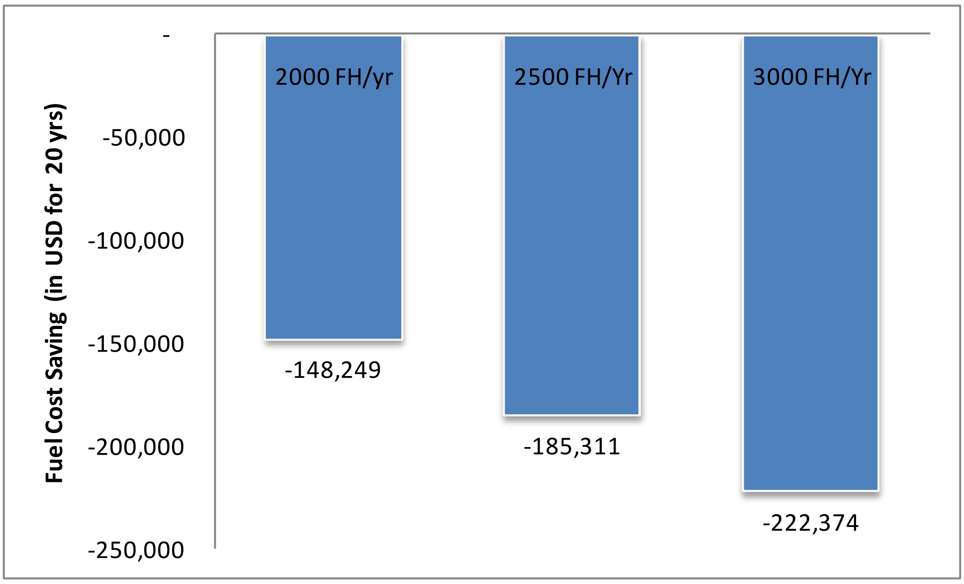


FIGURE 4-28 FUEL COST SAVING VS. THE VARIATION OF AIRCRAFT UTILISATION PER YEAR

4.8 Manufacturing Cost

Manufacturing cost estimation for the wing box is performed around the following parameters:

- Labour rate increase
- Material price variation
- The effect of the number of aircraft before Break Even Point (BEP)
- Increase complexity of stringer

TABLE 4-14 LIST OF PARAMETERS FOR MANUFACTURING COST ESTIMATION

Parameter	Description	Number of Combination
Labour rate increase	100% - 1000%	10
Material price variation	85% - 125%	10
The effect of the number of aircraft before Break Even Point (BEP)	100 - 200	11
Complexity	I and J Panel	2

There are 2200 cases of variation of the above parameters in the analysis of manufacturing cost. The results are shown below:

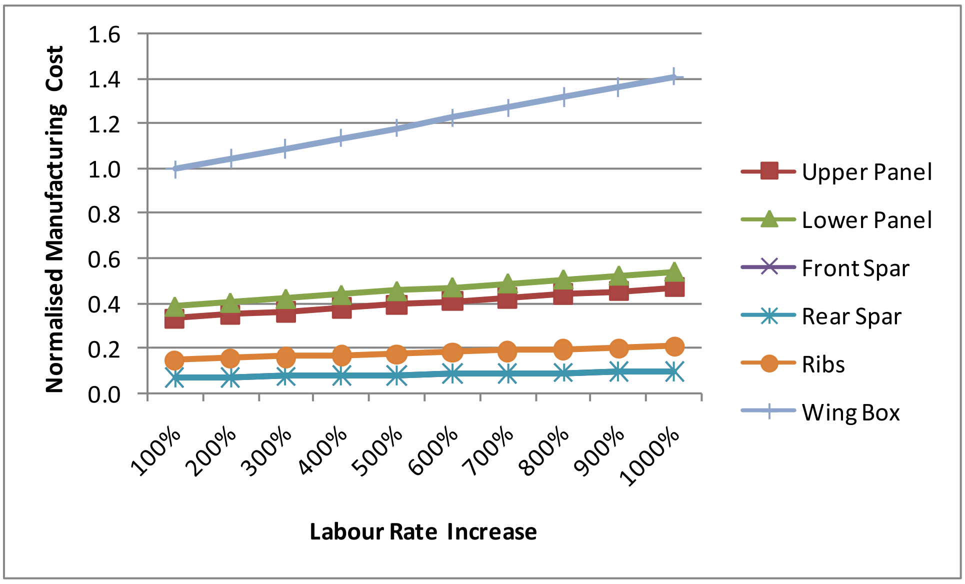


FIGURE 4-29 EFFECT OF LABOUR RATE INCREASE ON MANUFACTURING COST

From the above study, the increase of labour cost 10 times of the cheapest rate will increase manufacturing cost by 40%. This will obviously be quite significant for aircraft manufacturers in the USA and Europe, where their labour cost could reach 10 times higher than other country's labour cost; to move their manufacturing plant or outsource their manufacturing work.

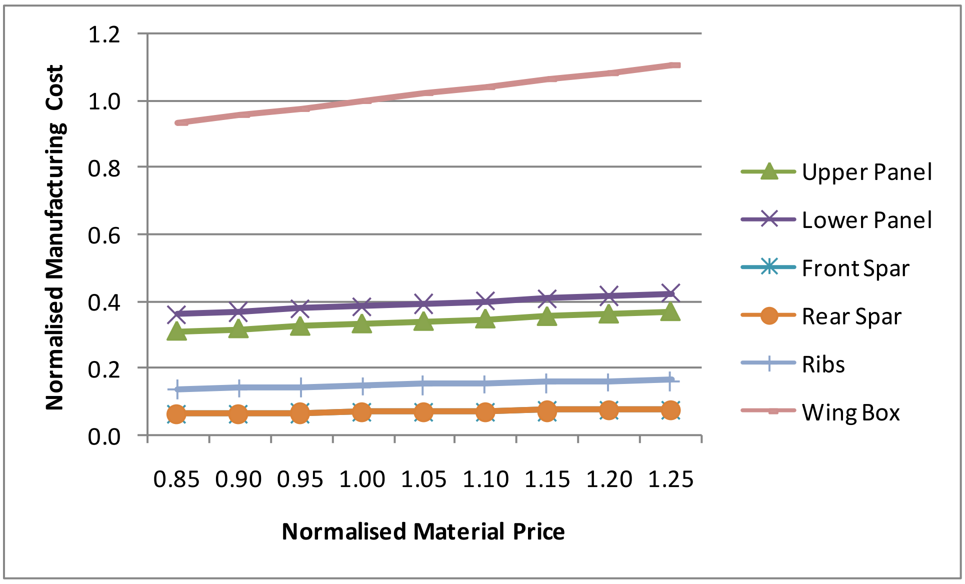


FIGURE 4-30 EFFECT OF MATERIAL PRICE VARIATION ON MANUFACTURING COST

At around 50%, material price is the biggest contributor to the total manufacturing cost. In this study, the variation is limited to 0.85 - 1.25 of the material cost in 2003. Moreover, since there are very limited material suppliers in the world, the manufacturer has limited flexibility to get a cheaper price.

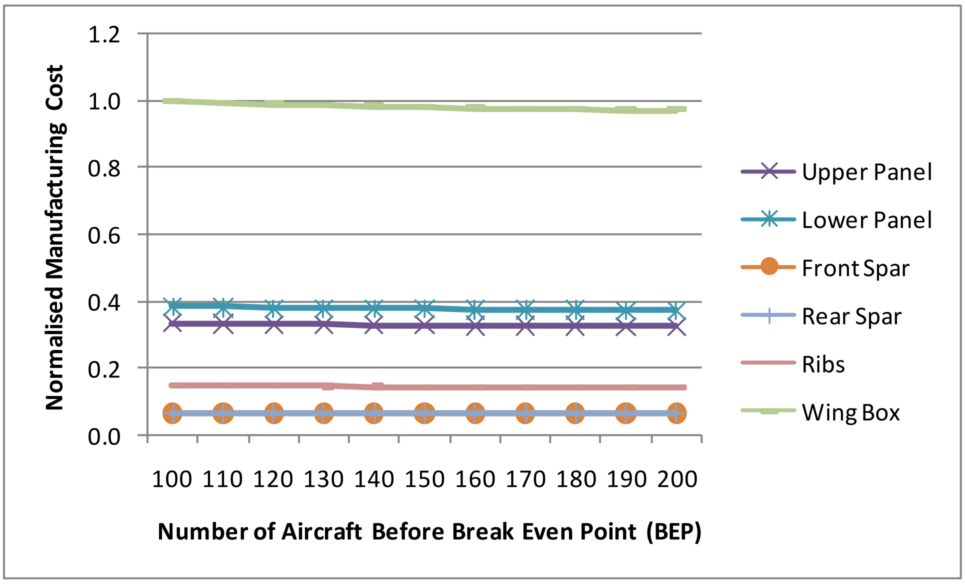


FIGURE 4-31 EFFECT OF NUMBER OF AIRCRAFT BEFORE BEP ON MANUFACTURING COST

This is the cost breakdown for major components of wing box by improving the machining rate compared to the standard tool. The following figure shows the comparison of material, labour and inspection cost for two different manufacturing processes.

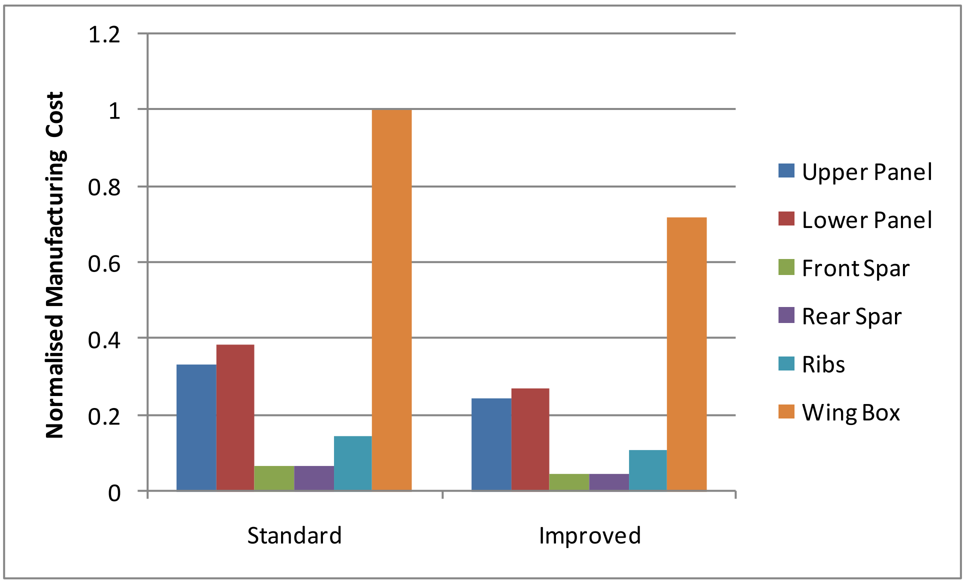


FIGURE 4-32 COMPARISON OF STANDARD AND IMPROVED MACHINING PROCESS

It can be seen from the above figure that by improving the process, a great opportunity also exists to reduce the manufacturing cost. Nevertheless, this study is focussed on improving removal rate by using better and bigger cutters, and with regard to this, the reduced manufacturing time is calculated directly in machine rate and labour rate, which gives the total manufacturing cost.

4.9 Design Trade Off Based on DOC

As shown in tables 4.4 – 4.6, and with reference to customer requirements, an optimum design selection is determined by its impact on the DOC. It is a process of selecting the combination of important parameters of design and manufacturing which gives the lowest DOC.

4.9.1 Breakdown of DOC per Flight Hour of the Original Aircraft

The calculation of DOC of the original aircraft, with the baseline data as in table 4-1, is based on assumptions of the costs of fuel and oil, maintenance, crew cost, depreciation (purchase price), and insurance per flight hour. The target purchase price of this aircraft is US$21.00m based on the price of similar type of aircraft in year 2008, i.e. Bombardier Q300 and Q400, (Airline Fleet & Network Management, 2008). The DOC breakdown of the original aircraft is based on the unpublished data in year 1998, but adjustments have been made on the purchase price and fuel cost to year 2008. DOC values at various fuel prices are shown in the following figure:

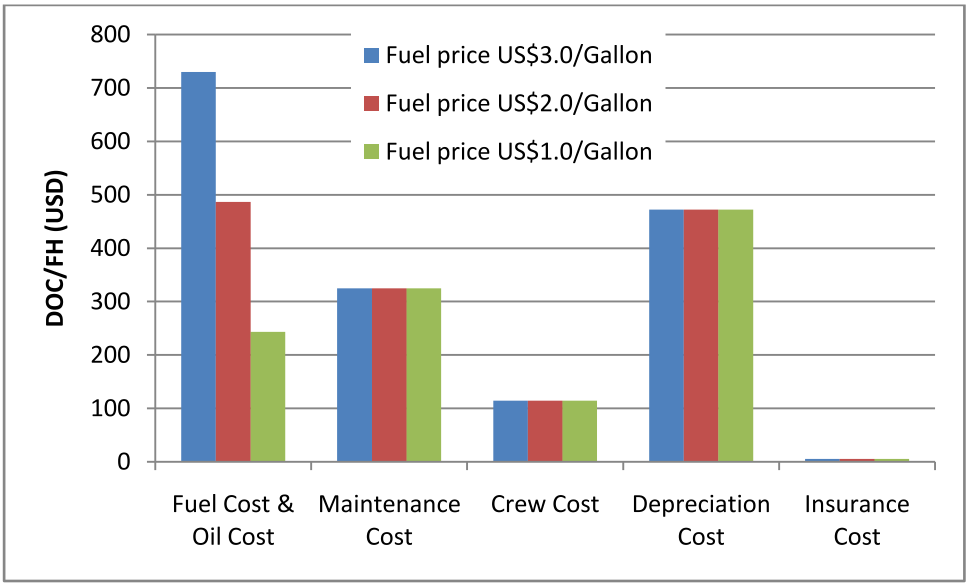


FIGURE 4-33 DOC/FH (USD/FH) OF THE ORIGINAL AIRCRAFT AT DIFFERENT FUEL PRICES

4.9.2 The Impact of Weight Reduction on DOC

It has been shown in figure 4-27 in section 4.7 that the new configuration and the use of new materials have reduced the wing box weight and have thus directly minimised the fuel consumption. The impact of these parameters is then translated into percentages of relative DOC per flight hour and per seat and shown in figure 4-34:

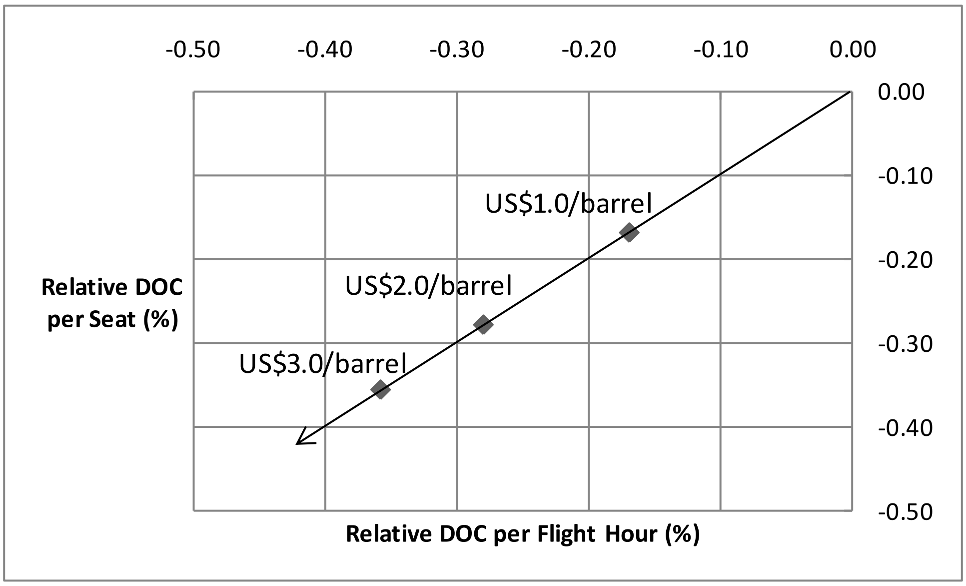


FIGURE 4-34 RELATIVE DOC (%) DUE TO FUEL SAVING (WING WEIGHT REDUCTION) AT VARIOUS FUEL PRICES CONDITION

It can be seen that the weight reduction is giving DOC reductions of less than 0.4%, which is relatively small. However it is predicted that the contribution from fuselage and tail structures redesign (based on proportion of the weight of those structures to the wing weight) could bring the total DOC reduction up to around 1.2%.

4.9.3 The Impact of Manufacturing Cost on DOC

The use of better materials is usually associated with a higher price of raw materials. As the price of raw material from suppliers is affected by several parameters, such as availability, form, quantity, etc, therefore during this study it is not fixed into a number but as a comparison to a reference price of basic materials available in literature (Swift & Booker, 2003). Figure 4-30 simulates the scenario of variations of manufacturing costs versus material price increases from 50% to 150% of basic material costs. For an aircraft utilisation of 2000FH/yr, then the designers could investigate the trade-off between price of new materials used and consequent fuel savings, due to the wingbox weight reduction. This would give evidence to make decisions if changes were acceptable DOC reductions:

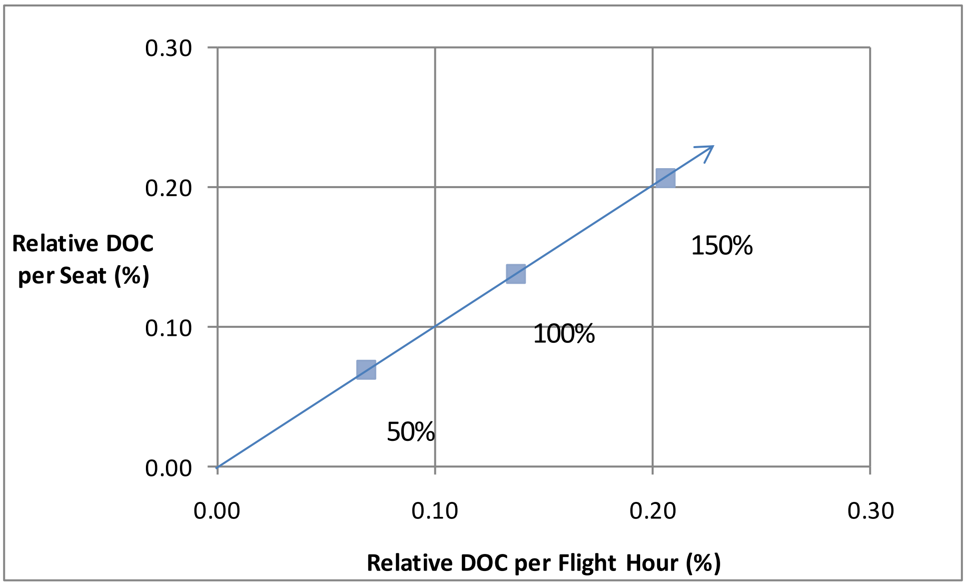


FIGURE 4-35 RELATIVE DOC (%) DUE TO NEW MATERIAL PRICE INCREASE (50-150%)

To compensate for the effect of higher new material costs, manufacturers may outsource the production to a country with lower labour rates, as shown in figure 4-22, chapter 4.7. The effect of labour costs reduction (in percentage) on DOC is shown in the following figure:

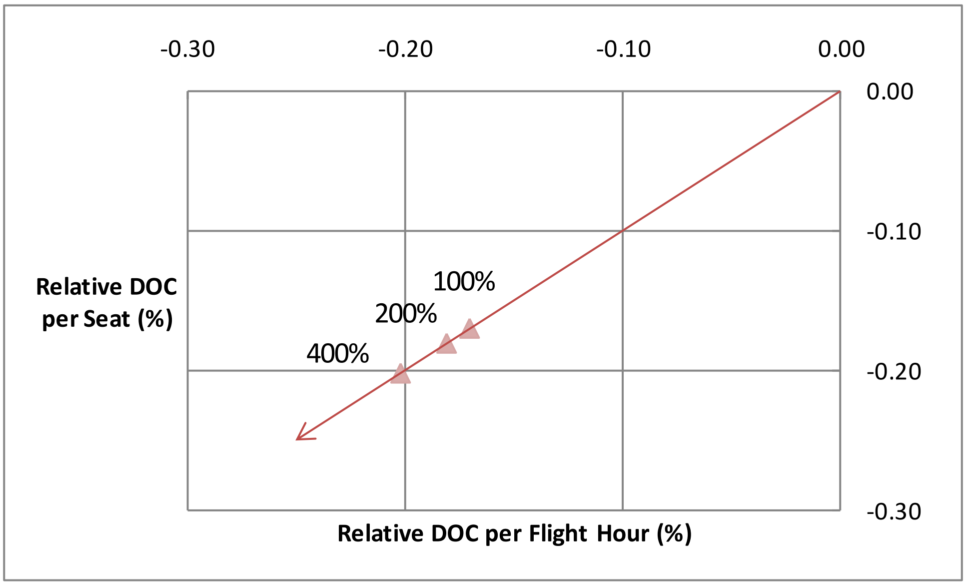


FIGURE 4-36 RELATIVE DOC (%) DUE TO LABOUR COST REDUCTION (100-400%)

It can be seen from figure 4-37 below that by improving the manufacturing process, such as using high speed machining process, an opportunity also exists to reduce the DOC.

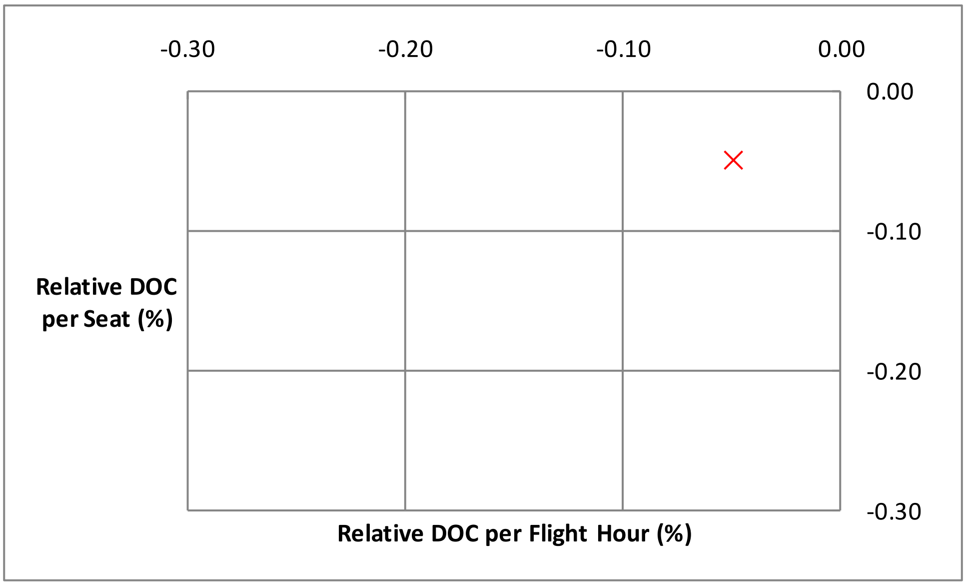


FIGURE 4-37 RELATIVE DOC (%) DUE TO HIGH SPEED MACHINING PROCESS

4.9.4 The Impact of Fuel Cost Saving, Material Price Increase, Labour Rate Reduction and Manufacturing Process on DOC

It is obvious that the overall DOC reduction is the result of the total impact of relative DOC effects due to fuel cost saving, material prices, labour rates, and manufacturing process improvements. Within the range of the calculated parameter values, the overall DOC reductions could be as much as 0.64% relative DOC. The impact of each parameter on the DOC is shown in the following figure. It appears that fuel prices, material cost and labour rate give greater impacts on DOC than high speed machining processes.

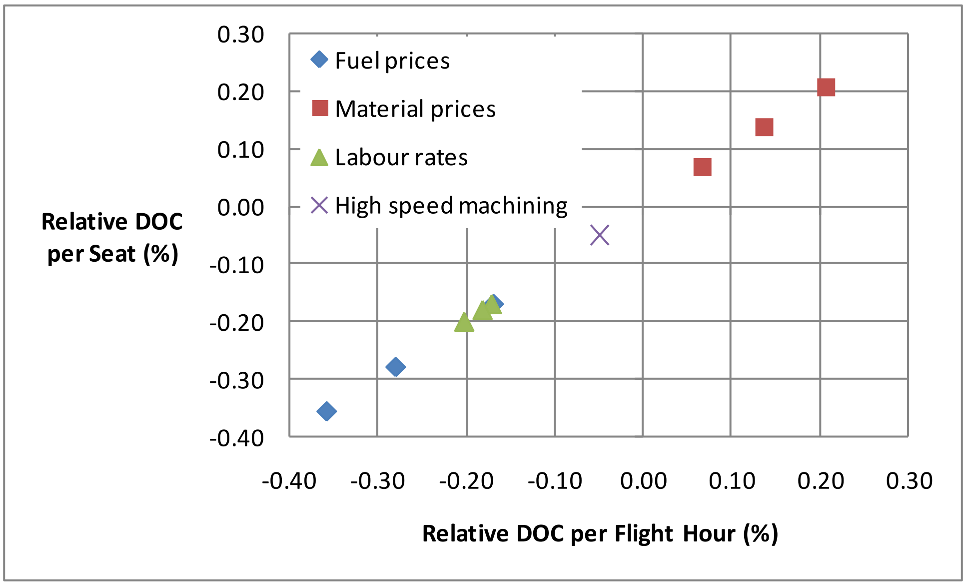


FIGURE 4-38 RELATIVE DOC (%) DUE TO FUEL PRICES, MATERIAL PRICES AND LABOUR RATES

The maximum DOC reduction is 0.61% DOC, achieved on the condition that the fuel price is the highest at US$3.0/gallon, in which there is a little material price increase, let say 50%, and the product manufacturing is outsourced to a country where the labour cost is only a quarter of current rate, in addition to these, the high speed machining process is utilised.

The minimum DOC reduction is on the assumption that the price of advanced material is 150% higher than the original one, and the aircraft is operating at the minimum fuel price, say US$ 1.0/gallon, and no machining process improvement or outsourcing to a country with cheaper labour rate have been incorporated. The DOC/FH is increases by small percentage of 0.04%. In this case, the cost of redesigning the wingbox structure using new advanced materials outweighs the fuel savings due to weight reduction.

It is important to remember that the above DOC analysis is only for wingbox structure redesign. Additional DOC improvement can also be obtained by redesigning the fuselage and the tail structures using similar approach.

4.9.5 The Impact of Maintenance Cost on DOC

Due to the use of advanced aluminium, maintenance cost is predicted to be less. It has better fatigue life and fracture toughness than the standard aluminium and therefore will increase the aircraft maintenance period for inspection and reduce repair costs due to slower crack damage growth. This cost saving contributes in reducing the life cycle cost of the aircraft. In addition, the number of crack stoppers could be reduced, therefore minimising weight and manufacturing cost. These benefits, however, have not been analysed.

4.10 Implementation and Validation of Developed Approach and Tool

Initial sizing procedure has been shown to give a conservative estimation for the upper skin panel. The lower skin panels are shown to be closer to the real aircraft thickness. Static strength analysis shows the requirements are met. However, fatigue and damage tolerance analysis, which is not done, will correct the sizing. The sizing of spar and ribs are also limited to static strength based on the material shear stress. The subsequent process of buckling assessment for the spar and the ribs web shows that although the thicknesses are not enough to meet the local buckling requirements, the use of additional stiffeners gives the necessary strength.

A finite element model was created in Patran and then submitted to Nastran software to validate the stress analysis from the software developed.

The following screenshots show the stress distribution on wing box and each major component under limit load due to gust.

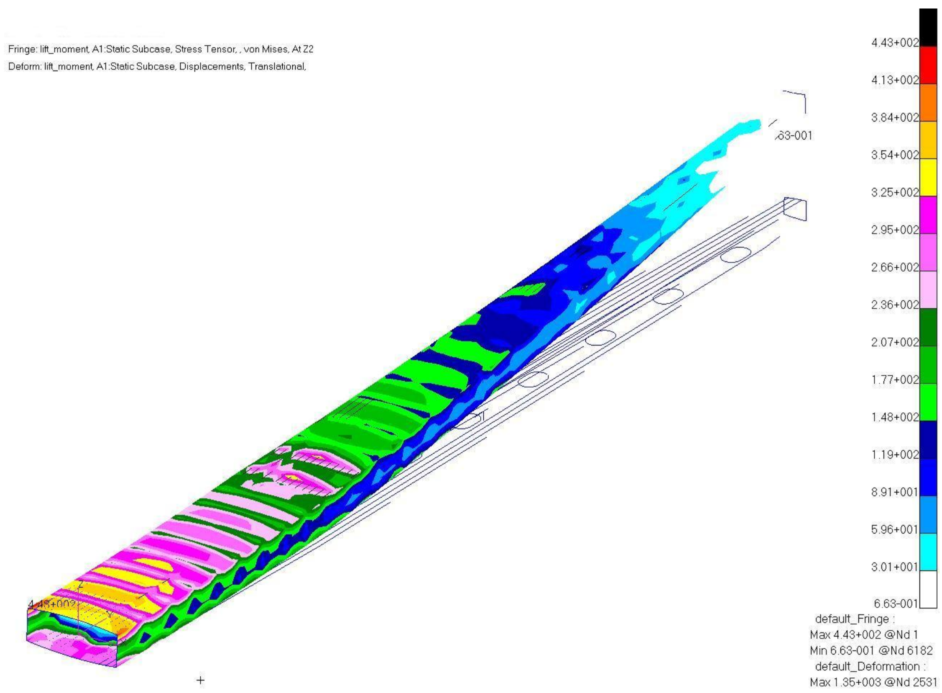


FIGURE 4-39 STRESS DISTRIBUTION ON WING BOX AT LIMIT LOAD DUE TO GUST

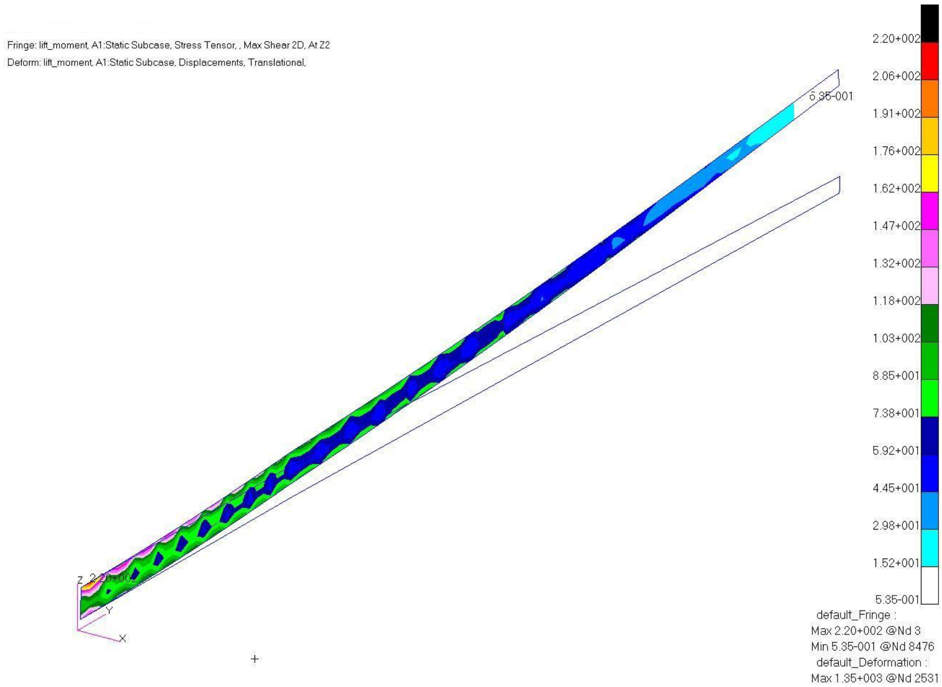


FIGURE 4-40 STRESS DISTRIBUTION ON REAR SPAR AT LIMIT LOAD DUE TO GUST

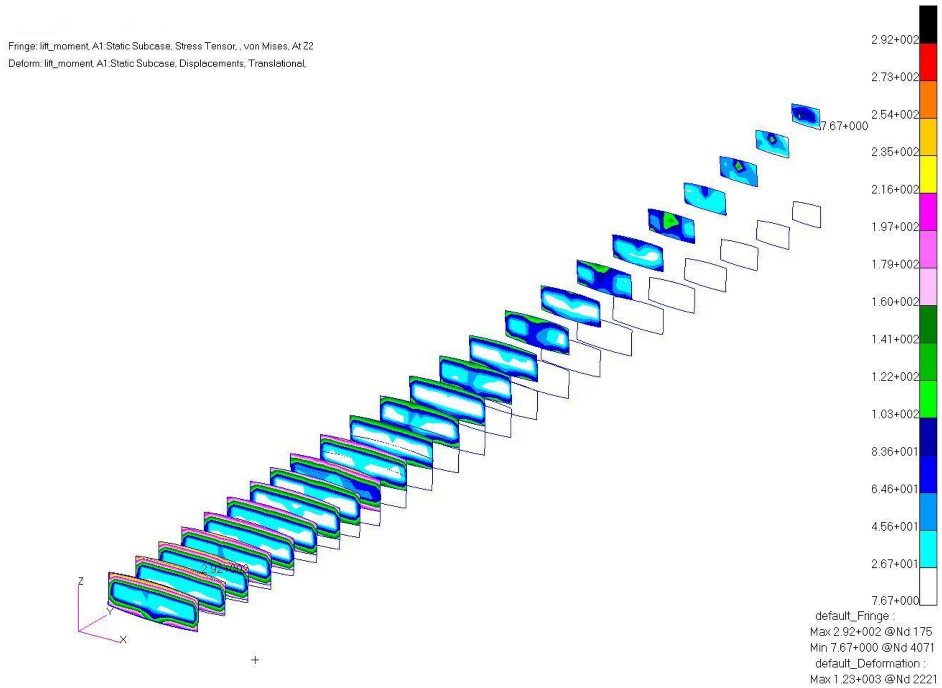


FIGURE 4-41 STRESS DISTRIBUTION ON RIBS AT LIMIT LOAD DUE TO GUST

TABLE 4-15 STRESS COMPARISON BETWEEN PROGRAM DEVELOPED AND NASTRAN FEA

Element	Stress type	Program (MPa)	Nastran (MPa)	Difference
Upper panel at root	Von-mises	299	271	9%
Upper panel at kink	Von-mises	214	227	-6%
Lower panel at root	Von-mises	193	196	-2%
Lower panel at kink	Von-mises	193	200	-4%
Front spar at root	Max Shear	79	75	5%
Front spar at kink	Max Shear	65	59	9%
Rear spar at root	Max Shear	116	111	4%
Rear spar at kink	Max Shear	95	92	3%

It shows that the analytical approach in analysis tool is quite accurate in predicting the stress distribution of the wing box. On average, a difference of less than 10% is acceptable for conceptual design.

The result of the current study was also compared with a study performed by N250's industrial team on the same aircraft. A comparison is made for one wing box configuration: the upper and lower panel of inboard and outboard inner wing are stiffened by integrally machined blade-type stringers. The upper surface is made of aluminium Al-7150-T7751, while the lower surface is made of aluminium Al-2024-T351. The spars and ribs are made of aluminium 2024-T351.

TABLE 4-16 COMPARISON OF STRUCTURAL THICKNESS (IN MM) BETWEEN CURRENT WORK AND INDUSTRIAL STUDY

Element	Thickness	Current	Industry	Difference
Upper panel at root	(mm)	3.8	4.0	-5%
Upper panel at kink	(mm)	3.4	3.2	6%
Lower panel at root	(mm)	7.8	8.0	-3%
Lower panel at kink	(mm)	5.1	5.0	2%
Front spar at root	(mm)	3.0	2.9	3%
Front spar at kink	(mm)	2.7	2.8	-3%
Rear spar at root	(mm)	3.1	2.9	7%
Rear spar at kink	(mm)	3.0	2.8	7%

It could be seen from the above table that the analysis tool gives a close result compared to a previous study done by N250's team. The percentage difference is within 10%. This comparison clearly illustrates that the proposed approach and tool is providing an acceptable result, and therefore demonstrates the attempt to get an optimum design at early stage of product development process to be a valid one.

Chapter 5 Discussion

ON THE CASE STUDY AND THE CONTRIBUTION TO THE KNOWLEDGE

5. DISCUSSION

In this chapter, the discussion is focused on the result of implementation the developed approach and tools in redesigning the wing box structure of transport aircraft. It then shows whether the objectives of the study have been achieved and describes the contribution of this study to the knowledge.

5.1 Results of the Case Study

The case study was redesigning a wing box structure of a 64-passenger turboprop low subsonic aircraft. The design target was to improve the DOC of the aircraft through reducing the fuel consumption and manufacturing cost. The developed approach and tools were used to assist the airframe design and manufacturing integration process during the conceptual design stage.

In chapter 4.2.1 to 4.2.3, the process of requirements definition required the designer to understand the aircraft characteristics, airworthiness and customer requirements before attempting the redesign process. The process was started with gathering the information from available resources, such as in ACDA, then creating a House of Quality matrix for prioritising the requirements using the QFD tool. During this process, the designer faced the issues such as how to identify the main parameters relevant to the airframe structures which included function of configuration, material and manufacturing process, etc. At this stage, an inexperienced engineer or a student could be supported by the airframe data base in ACDA which provides them the collection of necessary information. This increased the effectiveness of communication between the inexperienced engineers with more experienced designer on more important issues. The advantage of ACDA as a support tool was shown.

However, during the case study, several issues were found related the effectiveness of the database in ACDA and QFD tool. The database is currently at the early stage of development, there are several critical issues related with how the user could interact more efficiently with the tool. The tool does not provide an automatic suggestion or warning on specific choice or decision made by the user. However, the tool provides the static checklists as guidance and controls to the process. The additional issues on the use of the QFD tool were on making decision to put an 'importance rating' on each requirement of the airframe. Ideally this process is performed by a multi-disciplinary team, such as designer, stress, aerodynamics, customer, manufacture and maintenance, etc. The team will give better judgment than a single designer. However, from the case study it was found that even without this ideal situation, the user still is able to produce a quite comprehensive result in requirements definition.

During concept generation stage, in section 4.3, the database provided much information required by the inexperienced designer and students in solving various issues in generating the concepts of structure configurations, materials types, and manufacturing processes. For example, the designers were provided with various type of skin-stringer panels for different aircraft. They could also see the effect of different types of stringers on buckling effectiveness. The size of typical panels, including skin thickness and stringer sizing, were also provided for the user as their first estimate on the concepts that they designed.

The material information database was critical during the conceptual generation stage. In industry, the designer usually goes to the company database or the material handbooks for selecting the candidates of materials for their design. The developed support tool in ACDA provided an additional checklist to designer so that they could explore alternative materials used by current aircrafts or those being developed by suppliers and possibly research institutions. This kind of practice, where designer was reminded to explore new materials will give an opportunity to improve their design relating to existing aircraft. This highlighted the importance of exploring new technology through bringing specific suppliers into the design process from the very early design stage. In addition, a close cooperation with supplier plays significant factors in deciding whether new technology will be available during at the end of design stage.

The ACDA database on material currently consists of links to electronic format of Material Handbooks, and the information gathered from the available sources on the material application and the development of new materials.

Information on the development in new manufacturing processes were also provided to the user. For example the use of high speed machining was incorporated in the concepts generated, in which during manufacturing cost assessment, this new process could reduce the manufacturing cost of the wingbox structure.

During concept generation, 5 different concepts for wingbox structure were produced. For inexperienced engineers and students the achievement of producing various concepts could be seen as their ability to explore many possibilities within their limited experience on previous projects. This not only gives the opportunity for the user to develop their knowledge on the latest technology in the airframe structural design but also speeds up their time in learning from the experience of senior designers and opens up the possibilities to gather much needed wisdom of their seniors.

The author felt that with a mixed process of using the database in ACDA and at the same time having ‘real discussions’ to more experienced engineers is the best way for the inexperienced engineer and students to acquire tacit knowledge, which is known to be difficult to acquire, from experienced designer. In this research no attempts has been made to automate the knowledge transfer using KBE’s commercial tool.

In section 4.4, concept selection, the technique proposed by Prof. Pugh were shown to be quite effective at the early stage whereas very little information is available to the user. However, the simplicity of the technique would then allow for a broad assessment based on structural performance, manufacturing and maintenance to be made. Similarly to the QFD technique, this concept selection process is ideally performed by a multi-disciplinary team. However, by performing the assessment, the user could then select the most feasible concept in a more effective way than just based on the tendency to follow the 'normal way of doing' in the company.

During parametric synthesis, the initial sizing and parametric study on critical structural parameters were performed. This is one of very important processes for the inexperienced designer or student to quickly develop their 'sense' of different parameters of the structure and it's impacts to certain targets, such as weight and cost. Supporting tools developed by the author using the MATLAB language eliminate most of the burden of an already difficult situation from the shoulders of this type of user, in doing some repetitive analytical calculations. The user could perform as much calculation as they like to produce better designs and at the same time increase their understanding on the characteristics of their design.

The result of the analysis is quite close in comparison with the more elaborate and detailed analysis result by the team in Industry, i.e. within 10% margin. This gives confidence to the user in using the tools to explore structural parameters in order to achieve a better design.

In addition to this, the integration of design-manufacturing assessment in the developed tools helps the design process to be done more effectively and quickly even by the inexperienced engineers. They could easily perform trade-off analysis between fuel saving, due to weight reduction, and manufacturing cost variation, due to labour rate, material prices, and manufacturing processes, for the proposed concepts to achieve greater DOC reduction.

During the parametric synthesis, more than 3000 cases on structural design and more than 2000 cases on manufacturing assessment were investigated using the developed tools. As explained in 4.6-4.9, the new design could reduce the wing box structure weight by 16% compared to the original design. Fuel cost saving during 20 year of aircraft operation is up to US$ 200,000. Purchase price of the aircraft could be also be reduced due to using cheaper labour rate and new manufacturing processes by up to US$ 96,000 from wingbox structure only. If these cost saving was converted into DOC reduction, then the DOC reduction constitutes -0.36% of DOC due to fuel saving and 0.25% of DOC due manufacturing cost saving. The result confirms the findings by Fielding (1999) and Kinder (1995) on the impact of weight reduction and manufacturing cost on DOC reduction.

It is interesting to note from the case study results in section 4.9 that fuel prices, material cost and labour rate give greater impacts on DOC than high speed machining processes. Since, there are many possibilities in the current market situation, it is very important to assess the conditions in which redesigning the airframe structure will give the maximum DOC reduction, as is discussed in chapter 4.9.

The whole activities and the experience of the inexperienced engineer or the students in airframe design process could significantly improve their understanding about the design and could contribute to the possibility of creating innovative product through a thorough conceptual design process. In the long term the tool could help the process of knowledge transfer from senior experienced engineers to less-experienced engineers and students.

The analysis tool is currently limited to low subsonic transport aircraft. For high subsonic, non-conventional wing shape and composite structure, the user has to use different tools. However, the approach on design-manufacture integration is generic and therefore could be used.

5.2 Contributions to Knowledge

The result of this study in designing a wing box structure of a transport aircraft were discussed in the previous section. The author compared them to the how well the approach and tools tackled the major issues and whether the research contribution as mentioned in section 1.3 has been met.

By integrating the manufacturing cost parameters into the conceptual structural design process, as developed in the approach, the designer could make a better and more comprehensive decisions than just focusing on achieving minimum weight. The capability of the approach is shown as the extension of the tools to analyse the impact of fuel saving. These allow study on the effect of new configurations, advanced material, and manufacturing cost variations due to labour cost, material prices, etc, on DOC reduction. The first and second objectives of this research are thus satisfied.

By developing the Airframe Conceptual-Design Aid, ACDA, and Matlab wing loads, initial sizing, and manufacturing assessment tools, the important information based on the experience of senior engineers, and supported by test results and previous studies which are relevant to the work being done could be gathered more quickly. Therefore, it helps in retaining the valuable information in structural design and also improves the design process through better results and less time. Therefore, the third and final objective has been met.

The study developed a different approach and tools compared to the current state of the art as mentioned in chapter 2. The approach can be used generically for conceptual design process of wing box structure by integrating the design and manufacturing as suggested in the current studies. The tools were specifically developed to solve the issues laid down in section 1.3, such as it contributed to the process of retaining the airframe design knowledge from experienced engineers and pass them to inexperienced engineers and students. The developed airframe design tools are accessible through the internet and also open in which widen the target audience.

Chapter 6 Conclusion

AND RECOMMENDATION

6. CONCLUSIONS AND RECOMMENDATIONS

6.1 Conclusions

To produce a better airframe design, it is mandatory to investigate the problems of design and manufacturing integration early on at conceptual design stage. An approach and tool were required to aid the designer for future product development, which is expected to introduce difficulties due to increasing complexities.

The present work focuses on the development of an approach and design aids for designing wingbox structures. It facilitates the production of alternative structural concepts based on an existing product but also has the ability to capture the effects of advanced materials and manufacturing process on selecting structural concepts. It extends previous studies by the inclusion of manufacturing analysis in the wing box conceptual design process whilst keeping the analysis relatively simple to be performed using a personal computer. In addition, the use of a web-based approach for the supporting tools help the knowledge retention and transfer from experienced engineers to inexperienced engineers and students.

The following conclusions present the aspects of development, implementation and validation within the case study. Finally recommendations for further work will be given.

- Airframe design could be approached quite comprehensively at conceptual stage as shown in the case study.
- The use of Airframe Conceptual-Design Aid, ACDA, simplifies the synthesis process by providing important information to create concepts.
- The integration of new material and manufacturing analysis into the structural design process improves quality through weight reductions and fuel saving.
- The variation of raw material cost due to new material and labour cost due to variation of labour rate could be incorporated into the earlier design process. These manufacturing cost parameters have affected the selection of an airframe design concept.
- The accuracy of the developed software for initial sizing compared with the results of more elaborate work using FEA method is quite acceptable as the difference is within 10% margin.
- Comparison with the more accurate procedures used by the design team in the company designing the baseline aircraft showed that the percentage difference in airframe sizing is also within 10%. The proposed approach and tool therefore could be used to perform parametric studies, to obtain optimum concepts at the early stage of product development stage.
- The use of relative DOC as a design target helps in investigating the impact of fuel saving and manufacturing cost on the new design.

- The use of decision making technique has been beneficial in the trade-off process for selecting an optimum design.
- In the case study of 64 passenger low subsonic turbo prop aircraft, the new design could reduce the wing box structure weight by 16% using new configuration and advanced metallic material.
- For an optimum concept, fuel cost saving during 20 year of aircraft operation is up to US$ 200,000.
- Purchase price of the aircraft could be also be reduced due to using cheaper labour and new manufacturing processes by up to US$ 96,000 from wingbox structure only.
- If these cost saving was converted into DOC reduction, then the DOC reduction constitutes -0.36% of DOC due to fuel saving and 0.25% of DOC due manufacturing cost saving.
- The maximum DOC reduction is 0.61% DOC, achieved on the condition that the fuel price is the highest at US$3.0/gallon, in which there is a little material price increase, let say 50%, and the product manufacturing is outsourced to a country where the labour cost is only a quarter of current rate, in addition to these high speed machining process is utilised.
- The worst possibility scenario is on the condition that the price of advanced material is 150% higher than the original one, and the aircraft is operating at the minimum fuel price, let say US$ 1.0/gallon, and no machining process improvement or outsourcing to a country with cheaper labour rate have been incorporated. The DOC is increasing by small percentage of 0.04%. In this case,

redesigning wingbox structure using new advanced material outweighs the fuel saving due to weight reduction.

- The DOC reduction due to redesigning wing box structure is up to 0.54% DOC, which is relatively small. It is predicted that the contribution from fuselage and tail structures redesign could bring the total DOC reduction up to around 1.5% DOC.

6.2 Recommendation for Future Works

Several aspects within manufacturing assessment as well as model creation of complex structures have been addressed in this work. However it is still required to create a seamless integration of database and analysis tool, particularly on the application of advanced material and processes database.

The current state of the tool graphical user interface (GUI) which automatically stores and displays the selected information and decision making process will be required to speed up the design process. Development of database is an ongoing process. A more comprehensive database will improve the design process to achieve the target.

The case study on this project is limited to a conventional configuration, due to the availability of aircraft input data and detail manufacturing cost data for validation purposes. Further work is required for validating the approach and tool for different types of aircraft. It is also important to develop the tools further to allow the use of composite materials to be assessed, based on structure weight and manufacturing costs.

It is suggested to include the analysis of maintenance cost reduction for future work. This cost saving contributes to reduction of the life cycle cost of the aircraft since the material has better fatigue life and fracture toughness than the standard aluminium, and therefore will increase the aircraft maintenance period for inspection and repair due to slower crack damage growth. In addition, the number of crack stoppers could be reduced, therefore minimising weight and manufacturing cost.

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Appendix A
QFD Method

TRANSLATING CUSTOMER REQUIREMENTS INTO ENGINEERING CHARACTERISTICS

A. QFD METHOD

QFD is a planning and problem solving method that translates customer requirements into the engineering characteristics of a product. It is a graphic method that systematically examines the elements that go into the product development as a group effort (Dieter, 2000):

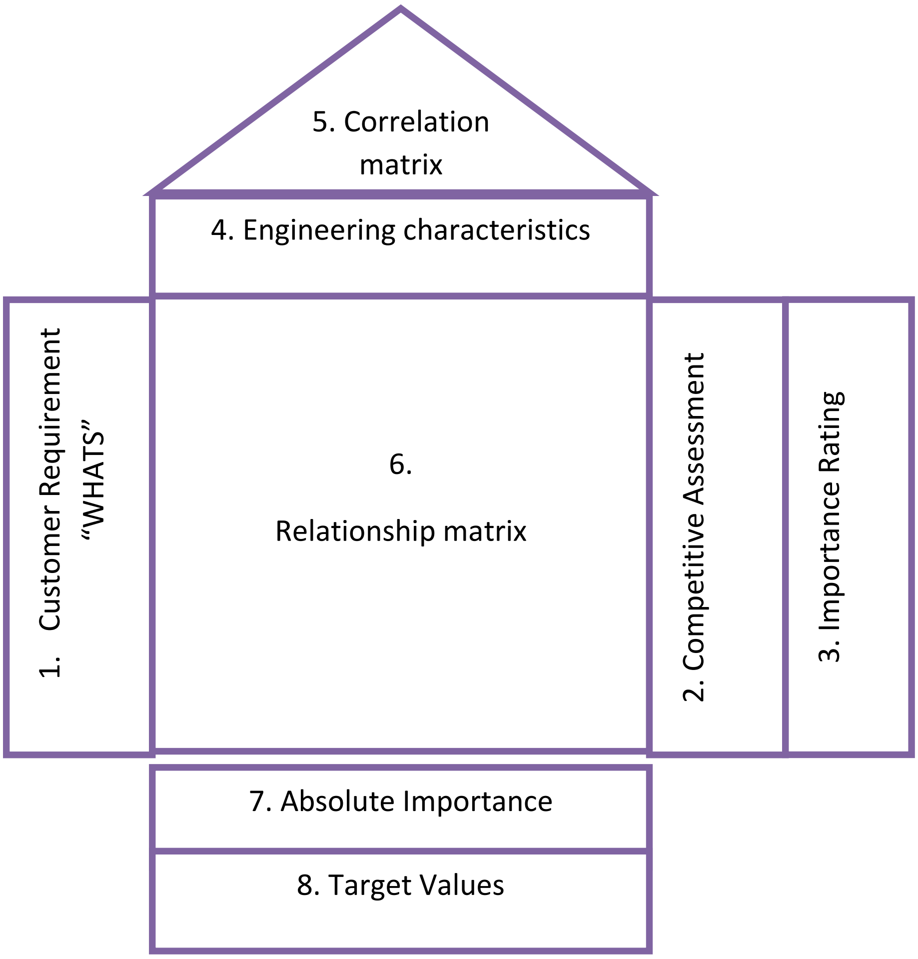


FIGURE A-1 HOUSE OF QUALITY MATRIX

The following is a description of the content of each QFD matrix box:

1. Customer requirements (WHATS) is defined based on the input from the customer
2. Competitive assessment shows the position of each customer requirement compared to the competition.
3. Importance rating shows the rating of each customer requirement
4. Engineering characteristics describes solutions to achieve the customer requirements
5. Correlation matrix shows the interaction between each solution in engineering characteristics
6. Relationship Matrix determines the correlation between the engineering characteristics and the customer requirements
7. To determine absolute importance is by multiplying the number in relationship matrix and number in importance rating
8. Target values is by knowing the target values for each engineering characteristics

QFD Matrix Program (Hales, 1995) was written using the Excel Program and has been embedded into ACDA. Hales (Hales, 1995) proposed the following steps are used to develop QFD Matrix:

1. Type the list of Outcomes (WHATs) and Metrics (HOWs) into the OutDescMet worksheet. These will automatically appear where needed. The user may also include a description if the Outcome might need more explanation.

2. Change the column headings in OutPrior to reflect the source of importance data and what are the rated competitors. The user can expand and add more columns if need be, but they will probably have to change the formulas for the calculations. If the number of columns is unchanged, the priorities will show up where needed on other sheets.
3. Sort the prioritized outcomes on the OutSort sheet. Sorting the data by the Overall Importance column will show the Outcomes deserve the most attention.
4. Define the relationships between the Outcomes and the Metrics using the Prioritization Matrix sheet. By default there is a 9 down the diagonal indicating which key Metric directly drives customer satisfaction relative to each Outcome. It is recommended that the user use the following criteria for setting relationship values: If a change in the value of the Metric causes a predictable change in the level of satisfaction of the Outcome, put a 9 in the appropriate cell. If the resulting change is moderately predictable, give it a 5. If a change will probably result, but it will be very small and unpredictable, give it a 1.
5. The resulting priorities for the metrics will be displayed in MetPrior. Use these priorities to determine what level of target value that should set. Set those target values directly on the MetPrior screen.
6. Define the interactions between the metrics using the Roof Matrix worksheet. Evaluate across the rows asking, "To what degree does a change in the Independent Metric impact the value of the Dependent Metric".

7. On the Alternatives screen, define any alternative design concepts that need evaluation. These will automatically show up on the Selection Matrix screen.
8. Finally, evaluate the alternative design concepts against the prioritized metrics on the Selection Matrix screen. Two effective approaches exist to evaluate alternatives. The first is to rate all of the alternatives by how well delivering the target for each Metric. If it can deliver the Metric's target, give it at least a 5. If it can easily exceed the target, give it a higher rating (to a 10 if possible). If it cannot deliver the target, give it a lesser value (to a zero, if necessary). The second is to pick one concept and rate all concepts relative to that baseline concept. If the concept is better than the baseline, give it a 1. If the concept is worse, give it a -1.

Appendix B
Airframe Database Design Aid (ADDA)

WEB BASED TOOL FOR SUPPORTING AIRFRAME DESIGN PROCESS

B. AIRFRAME CONCEPTUAL-DESIGN AID (ACDA) AND ANALYSIS TOOLS

Airframe Conceptual-Design Aid, ACDA, was developed for to assist the designer during wing box structural design process. This tool provides information for the inexperienced engineers and the students on the product development process and the implementation of design lessons learned in the following areas:

- critical issue of structural arrangement;
- characteristic of material;
- methods of fabrication and production cost; and
- in-support service requirements.

Together with ACDA, several analysis tools were also developed to assist the inexperienced engineers and the students in performing wing load analysis, initial sizing and static failure modes analysis, weight estimation, fuel cost analysis, manufacturing cost analysis and also DOC assessment.

The following diagram shows the structure of ACDA and Tools:

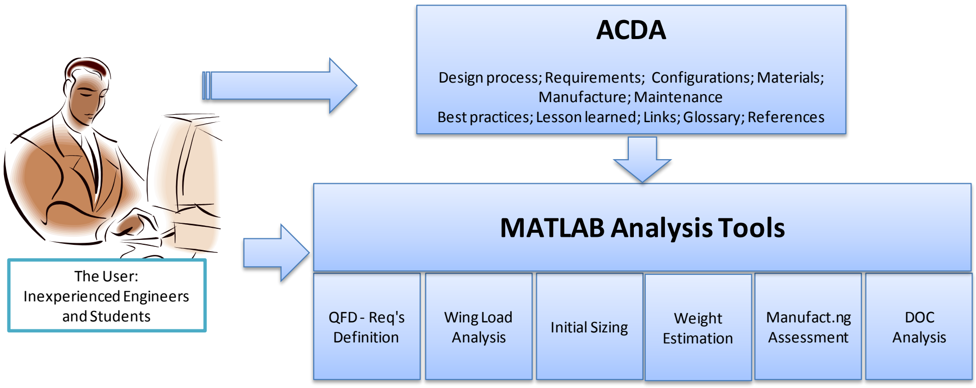


FIGURE B-1 ACDA AND ANALYSIS TOOL STRUCTURE

The information was collected from established literature, such as journals, working group papers, etc, and combined with material from visits and discussion with experts in industry and academia.

The database was developed electronically and can be accessed on the internet/intranet so that any necessary information, which is not normally available to designer without extensive surveys, will only be a click away. This database potentially could be useful to retain as much knowledge as possible from the experts.

The most challenging task during the tool development was to make the information adaptable for different product development stages, and to not overwhelm the user.

At this stage, the method of structuring the information gathered from the sources is following the procedure of design approach. There is no technique to convert engineering knowledge into a knowledge model, such as, MOKA, CommonKADS and the 47-Step Procedure has been utilised in developing the ACDA. This ACDA is simply a reservoir of relevant information, gathered from relevant sources, that is provided to the inexperienced engineer or student to improve their airframe design.

It could be summarised that the tool is developed around the proposed framework and should have the following characteristics:

- easy to use and non-restrictive in order to allow the designer to explore more creative thinking
- consisting of design information on industrial practice (visit and discussions), academic design projects, books, regulations, journals, and papers
- structured to support the airframe design approach but flexible so that it can also be used by designers employing other processes
- accessible through the internet and at any computer platform

The information in ACDA includes:

- Design approach
- Airworthiness requirements
- Past and existing design information
- Best practices
- Check lists
- Case study

Screenshots on the use of ACDA and MATLAB analysis tools have been shown throughout the case study in chapter 4.

Appendix C
Wing Loading Analysis

DEFINING WING LOAD DISTRIBUTION

C. WING LOADING ANALYSIS

This appendix shows the detailed calculations of wing load distribution using the procedure and formulas in chapter 3.

The airfoil used for this aircraft is NASA MS317, a 17 percent thick medium speed airfoil designed for general aviation applications (McGhee and Beasley, 1980):

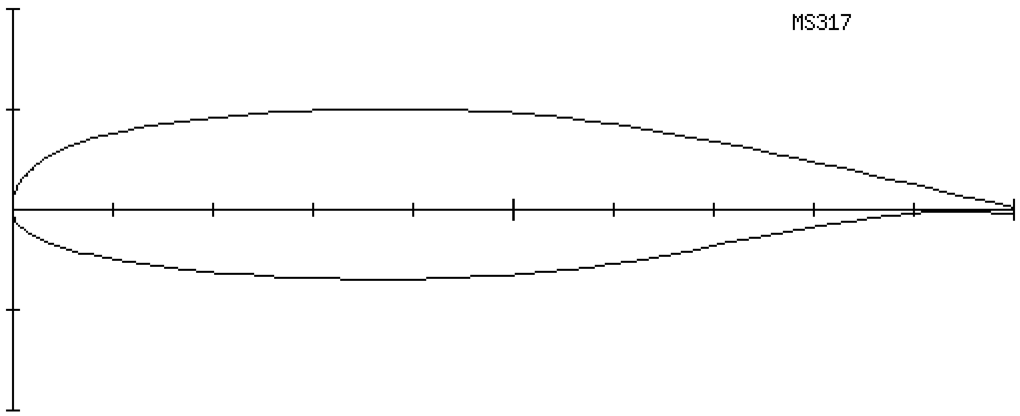


FIGURE C-1 NASA MS317 AIRFOIL SECTION (MCGHEE AND BEASLEY, 1980)

The aerodynamic characteristic of airfoil MS317 used for the calculation are (McGhee and Beasley, 1980):

Slope of lift coefficient, $m_o = 0.125/deg.$

Moment coefficient of aerodynamic, $c_{m\text{-}ac} = -0.07$

Design load is defined as the critical load acting on the structure and therefore is used in the structural design process. It consists of shear force, bending moment, and torsional moment distribution of wing span. Before obtaining it, we need to calculate wing aerodynamic load, inertia relieve load due to fuel, and the landing gear and engine, if they are placed on the wing. Once the airframe mass distribution is known, the total wing load can then be revised to include the airframe inertia relieve load. The process of defining design wing load is shown in figure 3-9.

As explained in chapter 3, the aerodynamic load distribution for wings with aerodynamic twist is obtained in two parts. The first part, called the basic lift distribution, is obtained for the angle of attack at which the entire wing has no lift. The second part, called the additional lift, is obtained by assuming the wing has lift but no aerodynamic twist. Therefore, the total lifts coefficient distribution is:

$$c_l = c_{lb} + c_{la} \qquad \text{C-1}$$

The method of calculating the additional lift coefficient consists simply of averaging the lift forces obtained from an elliptical lift distribution with those obtained from a planform lift distribution

$$c \cdot c_{la1} = \frac{1}{2}\left(\frac{m_o}{\overline{m_o}} c + \frac{4S}{\pi b} \sqrt{1 - \left(\frac{2y}{b} \right)^2} \right)$$

$$\overline{m_o} = \frac{\int_0^{b/2} m_o c\, dy}{\frac{S}{2}}$$

Since the airfoil type used is uniform along the wing span, i.e. MS317, from equation above: $\overline{m_o} = m_o = 0.125/deg.$

Then using equation 3-5 as quoted above, additional lift coefficient distribution, c_{la1}, can be calculated as shown in the following table:

TABLE C-1 ADDITIONAL LIFT COEFFICIENT DISTRIBUTION

y (1)	2y/b (2)	c (3)	$\sqrt{1-\left(\frac{2y}{b}\right)^2}$ (4)	$\frac{4S}{\pi b}\sqrt{1-\left(\frac{2y}{b}\right)^2}$ (5)	cc_l (6)	c_{la1} (7)
					0.5*[(3)+(5)]	(6) / (3)
0.000	0.000	2.800	1.000	2.957	2.879	1.028
0.550	0.039	2.800	0.999	2.955	2.877	1.028
2.030	0.145	2.800	0.989	2.926	2.863	1.022
2.960	0.211	2.800	0.977	2.890	2.845	1.016
3.835	0.274	2.800	0.962	2.844	2.822	1.008
4.245	0.303	2.800	0.953	2.818	2.809	1.003
4.777	0.341	2.726	0.940	2.780	2.753	1.010
5.840	0.417	2.579	0.909	2.688	2.633	1.021
6.904	0.493	2.432	0.870	2.573	2.502	1.029
8.075	0.577	2.269	0.817	2.416	2.342	1.032
9.353	0.668	2.092	0.744	2.200	2.146	1.026
10.631	0.759	1.915	0.651	1.924	1.920	1.002
11.909	0.851	1.738	0.526	1.555	1.646	0.947
13.188	0.942	1.561	0.336	0.992	1.277	0.818
14.000	1.000	0.000	0.000	0.000	0.000	0.000

It is important to note that c_{la1} in the above equation is relative lift coefficient where reference maximum lift coefficient is 1.0. The actual value of $c_{l\text{-}actual}$ is calculated by multiplying c_{la1} with lift coefficient of aircraft C_L that is calculated from equilibrium of flight condition, i.e.:

$$C_L = \frac{W}{\frac{1}{2}\rho V^2 S}$$

Therefore the actual additional lift coefficient is:

$$c_{la} = c_{la1} \cdot C_L$$

The basic lift coefficient distribution is obtained from the following equation:

$$c \cdot c_{lb} = \tfrac{1}{2} \cdot c \cdot m_o \cdot \alpha_a$$

And to calculate the wing angle of attack for zero lift is obtained from the following equation:

$$\alpha_{w0} = \frac{\int_0^{b/2} m_o \alpha_{aR} c\,dy}{\int_0^{b/2} m_o c\,dy}$$

Where an arbitrary reference plane is assumed and α_{aR} is measured from this plane to the zero-lift chord of each section, α_{w0} is the angle from this reference plane to the plane of zero lift for the wing.

The aircraft wing has a -3 deg twist at the tip to improve stall characteristics. By assuming the twist angle is not sudden, but changing gradually from centreline to the tip then the basic lift coefficient along the wing span can be calculated.

Using equation 3-10 above and the linear integration for small elements of wing span, calculate the wing zero-lift plane for the wing:

$$\alpha_{w0} = \frac{-41.612}{32.5}$$

$$\alpha_{w0} = -1.28\,\text{deg}$$

For the whole wing span section, the calculation of basic lift distribution is shown in the following table. Lift curve slope of airfoil MS317, $m_o = 1.125/deg$

TABLE C-2 BASIC LIFT COEFFICIENT DISTRIBUTION

		from ref. line			from zero plane	$c.c_{lb}$ =	
y	c	α_{oR}	$c*\alpha_{oR}$	$c*\alpha_{oR}\,dy$	α_a	0.5 c. m_o. α_a	c_{lb}
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)
			(2) x (3)		(3) - (- 1.28)	0.5x(2)x0.1x(6)	(7) / (2)
0.000	2.800	0.00	0.00		1.28	0.224	**0.080**
0.550	2.800	-0.12	-0.33	-0.09	1.16	0.203	**0.073**
2.030	2.800	-0.44	-1.22	-1.15	0.85	0.148	**0.053**
2.960	2.800	-0.63	-1.78	-1.39	0.65	0.113	**0.040**
3.835	2.800	-0.82	-2.30	-1.78	0.46	0.080	**0.029**
4.245	2.800	-0.91	-2.55	-0.99	0.37	0.065	**0.023**
4.777	2.726	-1.02	-2.79	-1.42	0.26	0.044	**0.016**
5.840	2.579	-1.25	-3.23	-3.21	0.03	0.005	**0.002**
6.904	2.432	-1.48	-3.60	-3.64	-0.20	-0.030	**-0.012**
8.075	2.269	-1.73	-3.93	-4.42	-0.45	-0.064	**-0.028**
9.353	2.092	-2.00	-4.19	-5.20	-0.72	-0.095	**-0.045**
10.631	1.915	-2.28	-4.36	-5.48	-1.00	-0.119	**-0.062**
11.909	1.738	-2.55	-4.44	-5.64	-1.27	-0.138	**-0.079**
13.188	1.450	-2.83	-4.10	-5.48	-1.55	-0.140	**-0.097**
14.000	0	-3.00	0.00	-1.71	-1.72	0.000	**0.000**

The total lift coefficient distribution, is:

$$c_l = c_{lb} + c_{la} \qquad \text{C-2}$$

Therefore, the total lift coefficient distribution is shown in the following table:

TABLE C-3 TOTAL LIFT COEFFICIENT DISTRIBUTION

(1)	(2)	(3)	(4)	(5)	(6)
			$C_L = 1$		$C_L = 1.07$
y	c_{lb}	c_{la1}	c_l	$1.07\, c_{la1}$	c_l
			(2) + (3)	1.07 x (3)	(2) + (5)
0.00	0.080	1.028	1.108	1.107	1.187
0.55	0.073	1.028	1.100	1.107	1.180
2.03	0.053	1.022	1.075	1.101	1.154
2.96	0.040	1.016	1.057	1.095	1.135
3.84	0.029	1.008	1.037	1.086	1.114
4.25	0.023	1.003	1.026	1.081	1.104
4.78	0.016	1.010	1.026	1.088	1.104
5.84	0.002	1.021	1.023	1.100	1.102
6.90	-0.012	1.029	1.016	1.108	1.096
8.08	-0.028	1.032	1.004	1.112	1.084
9.35	-0.045	1.026	0.981	1.105	1.060
10.63	-0.062	1.002	0.940	1.080	1.017
11.91	-0.079	0.947	0.868	1.020	0.941
13.19	-0.097	0.818	0.721	0.881	0.784
14.00	0.000	0.000	0.000	0.000	0.000

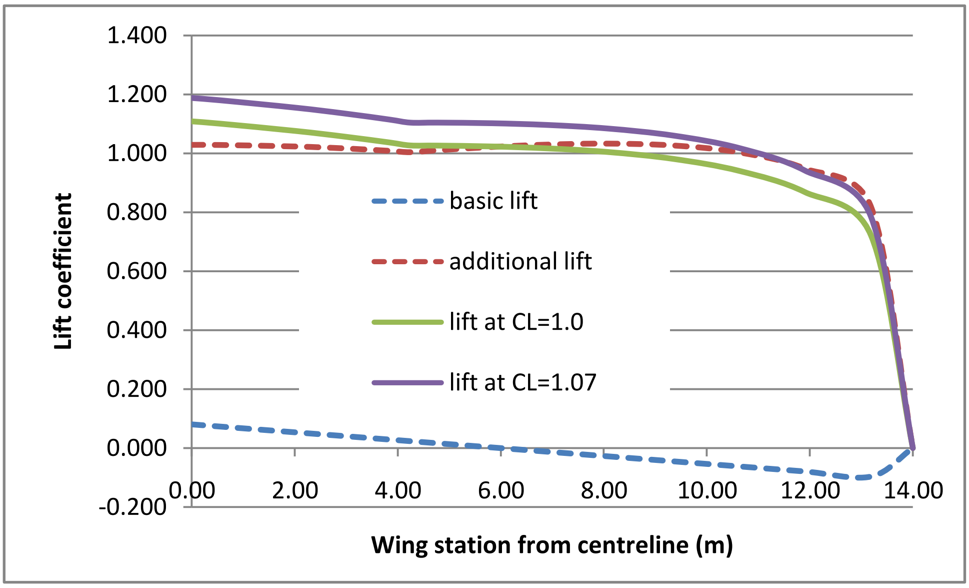


FIGURE C-2 LIFT COEFFICIENT DISTRIBUTION

Shear Force, Bending Moment, and Torsional Moment distributions are calculated from the integration of small elements of forces acting on the wing box structure. The wing is assumed to have a fixed-end condition at the wing root position, with the tip being a free condition. Again, it is important to note that wing structure mass and fuel load distribution are likely not linear; therefore, the above lines are only for illustration.

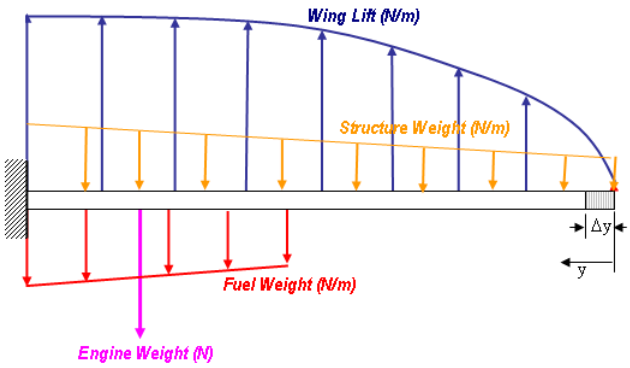


FIGURE C-3 MODELING OF FORCES DISTRIBUTION ACTING ON AIRCRAFT WING

When considering small elements from wing tip to the root, and using equilibrium equation for each element, then:

$$SF_{i+1} = SF_i + \Delta L_{i+1} - W_{fuel}\Delta y - W_{engine} - W_{Structure}\Delta y \qquad \text{C-3}$$

And bending moment distribution can be calculated using the following equation:

$$BM_{i+1} = BM_i + \Delta BM_i$$

and $$\Delta BM_i = (SF_{i+1} + SF_i)\frac{\Delta y}{2}$$

Torsional Moment at the shear centre of each section is calculated as follows:

$$T_{i+1} = T_i + \Delta T_i$$

and $$\Delta T_i = \Delta M_{ac} + \Delta L_i \left(\frac{(c_{FS} + c_{RS})}{2} - 0.25c \right)$$

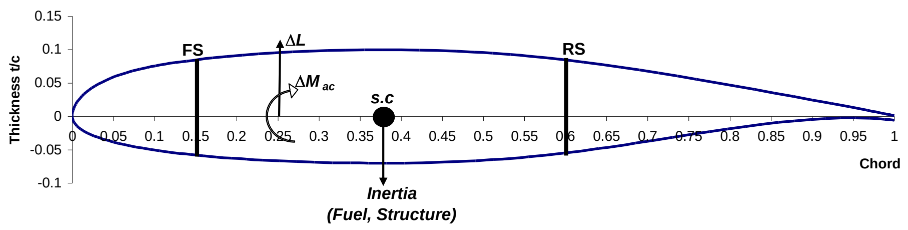


FIGURE C-4 WING BOX CROSS SECTION

Whereas:

BM_i, *T_i*,and *SF_i* represent bending moment, torsion, and shear force at any point in the spanwise section. *L* and *M_{ac}* are the lift and the moment of aerodynamics around the aerodynamic centre.

$$\Delta L = \tfrac{1}{2}\rho V^2 \Delta S c_l \qquad \textbf{AND} \qquad \Delta M_{ac} = \tfrac{1}{2}\rho V^2 \Delta S c_{MAC} c$$

c_{FS} and c_{RS} are the position of front and rear spars from the leading edge.

The detailed design load calculation and associated data used are shown in the following table:

TABLE C-4 INTERNAL STRESSES DISTRIBUTION

N250: 64 Pax, 2 x RR AE2100C Engine

WEIGHT				DIMENSIONS			SPEED		
Take Off Weight	25000	Kg		Wing Area	65	m2	Cruise Speed (TAS)	165	m/s
				$C_{m\text{-}ac}$	-0.07		Air Density (20000ft)	0.652925	Kg/m3
Engine weight (each)	500	Kg	ws3835	C_L at Cruise MTOW	0.424	(at Vc, 20k ft)			
gravity constant	9.8	m/s2		C_L at max n_z	1.077	(C_L x n_z)			
				Front Spar	15%	chord			
max vertical load factor	2.54		due to gust at Vc	Rear Spar	60%	chord			
Fuel Capacity	4000	Kg	ws550 to ws6900						
Fuel density	800	Kg/m3							
Tank capacity	0.80								

y	Δy	c	c_{av}	$C_L = 1$	C_L at $n_{Z\,max}$	c_{lav}	$(0.5\,\rho V^2 c_l)_{av}$	ΔSF	ΔSF	ΔSF	ΔSF	ΔSF	SF	ΔBM	BM	ΔT	T
(m)	(m)	(m)	(m)	c_l	1.08			(N)	(N)	(N)	(N)	(N)	(N)	(Nm)	(Nm)	(Nm)	(Nm)
								(Aero)	(Fuel)	(Engine)	(Airframe)	(Total)	Σ ΔSF		Σ ΔBM		Σ ΔT
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)	(15)	(16)	(17)	(18)
					1.07x(1)			(8)x(2)x(4)				(9)+(10)+(11)+(12)					
0.000		2.800		1.11	1.19								240,846		1,520,823		23,951
	0.550		2.800			1.18	10,519	16,200				16,200		128,010		2,987	
0.550		2.800		1.10	1.18								224,647		1,392,812		20,964
	1.480		2.800			1.17	10,372	42,980	-12,570		-256	30,410		309,973		3,425	
2.030		2.800		1.08	1.15								194,236		1,082,839		17,539
	0.930		2.800			1.14	10,173	26,491	-7,899		-142	18,592		171,994		1,971	
2.960		2.800		1.06	1.13								175,644		910,845		15,569
	0.875		2.800			1.12	9,996	24,489	-7,431		-117	17,058		146,226		1,702	
3.835		2.800		1.04	1.11								158,586		764,619		13,866
	0.000		2.800			1.11	9,904	0	0	-12,446	0	-12,446		0		-4,356	
3.835		2.800		1.04	1.11								171,032		764,619		18,222
	0.410		2.800			1.11	9,857	11,316	-3,482		-49	7,834		68,517		742	
4.245		2.800		1.03	1.10								163,198		696,102		17,480
	0.532		2.763			1.10	9,811	14,421	-4,400		-59	10,021		84,156		934	
4.777		2.726		1.03	1.10								153,177		611,946		16,546
	1.063		2.653			1.10	9,801	27,636	-8,102		-104	19,534		152,445		1,823	
5.840		2.579		1.02	1.10								133,643		459,501		14,723
	1.064		2.506			1.10	9,766	26,034	-7,236		-85	18,799		132,196		1,732	
6.904		2.432		1.02	1.10								114,845		327,306		12,991
	1.171		2.351			1.09	9,687	26,662			-72	26,662		118,872		3,809	
8.075		2.269		1.00	1.08								88,182		208,433		9,182
	1.278		2.181			1.07	9,527	26,548			-56	26,548		95,733		3,456	
9.353		2.092		0.98	1.06								61,635		112,700		5,726
	1.278		2.004			1.04	9,231	23,636			-38	23,636		63,665		2,728	
10.631		1.915		0.94	1.02								37,998		49,035		2,999
	1.278		1.827			0.98	8,703	20,314			-25	20,314		35,581		1,985	
11.909		1.738		0.87	0.94								17,684		13,454		1,013
	1.279		1.594			0.86	7,667	15,632			-18	15,632		12,621		1,093	
13.188		1.450		0.72	0.78								2,052		833		-80
	0.812		0.725			0.39	3,486	2,052			-1,022	2,052		833		-80	
14.000		0.000		0.00	0.00								0		0		0

The result of the above calculation can also be shown in the following usual plot of shear force, bending moment and torsion along the wing span.

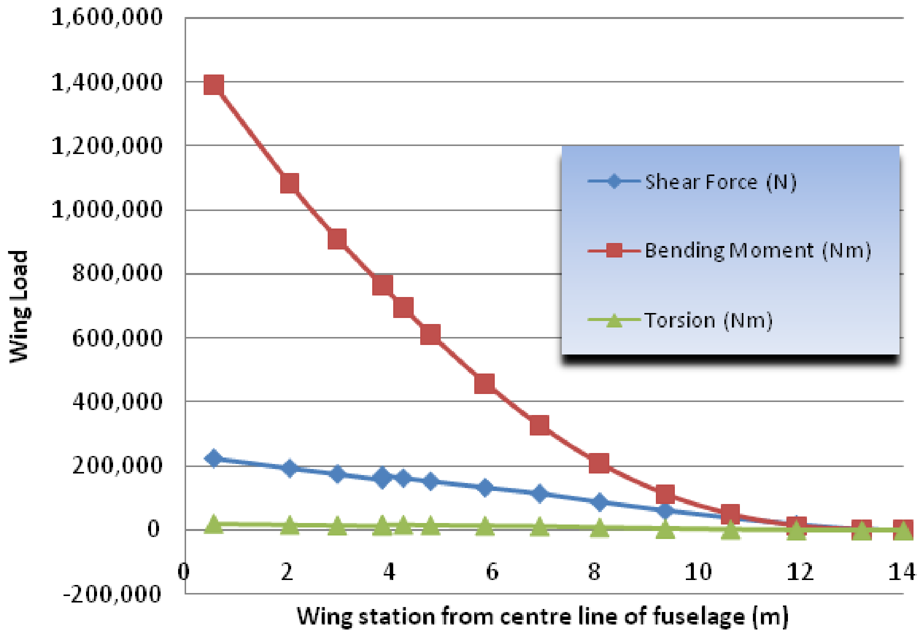


FIGURE C-5 WING LOAD DISTRIBUTION, AT MTOW, V_C, $n_z = 2.54$

Appendix D
Initial Sizing and Static Failure Modes Analysis

INITIAL SIZING AND STATIC FAILURE MODES ANALYSIS

D. INITIAL SIZING AND STATIC FAILURE MODES ANALYSIS

This appendix shows the detail calculation of initial sizing, and failure modes analysis using the procedure and formulas in chapter 3. In this section, the detailed analysis is shown for the upper skin panel at station 550 (at wing root), in which the subsequent rib is set at 500mm. The upper skin panel has an integral blade stringer configuration, made from Al-7150 T7751. The lower skin panel also has an integral blade stringer configuration, using Al-2024 T351. The material properties are obtained from MMPDS-01 (FAA, 2003), shown in the following table:

TABLE D-1 SKIN PANEL CONFIGURATIONS AND MATERIAL PROPERTIES (A-VALUE) (FAA, 2003)

PART DESCRIPTION			
Part Name		Upper Skin Panel	Lower Skin Panel
Location	(STA mm)	550-13188	550-13188
Skin Panel Configuration		Integral 'blade'	Integral 'blade'

MATERIAL			
Material Name		7150	2024
Temper		T7751	T351 Bare
Type		Plate	Plate
Raw material Thickness	(in)	2.0-3.0	2.0-3.0
Ultimate tensile Strength, f_{tu}	(MPa)	565	414
Yield tensile Strength, f_{ty}	(MPa)	524	290
Compressive yield, Proof Strength, f_2	(MPa)	517	255
Ultimate Shear Strength, f_s	(MPa)	324	241
Modulus Elastic, E	(MPa)	71016	73774
Density, ρ	(kg/m3)	2823	2768

Referring to Appendix C, design loads used for the sizing of wing box structure:

TABLE D-2 WINGBOX DESIGN LOADS: LIMIT AND ULTIMATE

Limit Load			
STA	*BM*	*SF*	*T*
(m)	*(Nm)*	*(N)*	*(Nm)*
0.00	1,531,378	240,875	44,383
0.55	1,403,295	224,884	40,319
2.03	1,092,628	194,936	33,962
2.96	919,882	176,560	30,123
3.84	772,790	172,098	31,000
4.25	703,825	164,317	29,419
4.78	619,060	154,347	27,420
5.84	465,347	134,858	23,617
6.90	331,873	116,034	20,096
8.08	211,665	89,273	14,533
9.35	114,655	62,543	9,408
10.63	49,985	38,662	5,251
11.91	13,743	18,053	2,062
13.19	854	2,103	39
14.00	0	0	0

Ultimate Load = 1.5 x Limit Load			
STA	*BM*	*SF*	*T*
(m)	*(Nm)*	*(N)*	*(Nm)*
0.00	2,297,068	361,312	66,574
0.55	2,104,942	337,326	60,478
2.03	1,638,942	292,404	50,943
2.96	1,379,824	264,840	45,184
3.84	1,159,185	258,147	46,500
4.25	1,055,737	246,476	44,129
4.78	928,590	231,521	41,130
5.84	698,021	202,287	35,425
6.90	497,809	174,052	30,144
8.08	317,498	133,909	21,800
9.35	171,982	93,815	14,112
10.63	74,977	57,993	7,877
11.91	20,615	27,080	3,092
13.19	1,281	3,154	58
14.00	-	-	-

The procedure for upper skin panel sizing is explained in section 3.4.3, and therefore used in here. The following detailed calculation is performed on the original configuration of wing box structure at upper skin panel at Sta. 550. The following data is taken from the original configuration. The purpose of this calculation is firstly to check the accuracy of the proposed formulae, and secondly to check whether the initial configuration is buckling free.

Stringer pitch, $b = 0.086\ m$; Rib pitch, $L = 0.500\ m$

Ratio of Skin area to stringer area, $\frac{A_{sk}}{A_{st}} = c = 1.0$;

Wingbox section depth, $h = 0.428\ m$;

Wingbox width, $w = 1.260\ m$;

Wingbox section area, $A = 0.585\ m^2$

Young modulus Al 7150-T7751, $E = 71,016 \times 10^6\ N/m^2$

Ultimate design loads: $M = 2,104,942\ Nm$ and $T = 60,478\ Nm$

ESDU 70003 (Anon, 1970), $c = 1.0;\ h/b = 0.58;\ t_s/t = 1.6$:

from fig.1: $K_c = 4.0;\quad \eta = 0.9$

ESDU 71005 (Anon, 1971), for simply supported $b/L = 0.18$,

from fig.1: $K_s = 5.0;$

$Q_T = T/2A = 60,478/(2x0.585) = 51,690\ N/m$

$$A = \frac{\eta^2 {K_c}^2 {K_s}^2 E^2}{b^4} = 2.987 \times 10^{28}$$

$$B = -\frac{\eta K_c {K_s}^2 EMc}{b^2 hw(1+c)} = -1.686 \times 10^{21}$$

$$C = -{K_c}^2 {Q_T}^2 = -4.275 \times 10^{10}$$

$$({t_s}^3) = \frac{-B \pm \sqrt{B^2 - 4AC}}{2A} = 5.643 \times 10^{-8}$$

$t_s = 3.83 \times 10^{-3} m = 3.8\ mm$

The skin thickness of original upper skin panel at station 550 is 4.0 mm. By comparing the result of analysis and the original dimension, it could be concluded that the proposed approach to obtain initial sizing for skin panel under combined loading, compression and shear, gives a close result. The proposed approach gives an optimum thickness, in which RF = 1.0, in this case the difference on the skin thickness may be caused by the company allowance for additional safety on the wingbox structure. The same procedure was also applied for other station of upper skin panel.

The Matlab sizing program has been developed following this procedure, as specifically shown in section 3.4.3, and used for initial sizing and failure modes analysis of upper and lower skin panel, front and rear spar web, and typical rib web.

Appendix E
The Cost Support Data

E. THE SUPPORT COST-DATA

This appendix shows the support data used during fuel saving, manufacturing cost assessment and DOC analysis.

TABLE E-1 TYPICAL DIRECT OPERATING COST N250

TYPICAL DIRECT OPERATING COST (N250)			
Assumptions			
1	Aircraft Price	21,000,000	US$
2	Aircraft Utility	2,000	FH/Year
3	Depreciation Period	20	Years
4	Residual Value of Aircraft	10	%
5	Rate of Insurance	1.00	% aircraft price/year
6	Crew Salary		
	Pilot Salary	3,000	US$/month
	Co-pilot Salary	2,250	US$/month
7	Maximum Crew Utilisation	1,050	FH/Year
8	Fuel Price	**2.25**	US$/gallon
9	Average Fuel Consumption	700	kg/Hrs
10	Oil Cost	2.5	% fuel cost
11	Maintenance Cost		
	Part & Material	133	US$/FH
	Engine & Propeller Reserve	150	US$/FH
	Man Hour Cost	42	US$/FH
Direct Operating Cost Break-down			
1	Fuel Cost & Oil Cost	548	US$/FH
2	Maintenance Cost	325	US$/FH
3	Crew Cost	114	US$/FH
4	Depreciation Cost	473	US$/FH
5	Insurance Cost	5	US$/FH
Direct Operating Cost (DOC)		1,465	US$/FH
Fuel price datum during calculation		**2.25**	US$/gal
		0.50	US$/l

Fuel price information source:
http://www.iata.org/whatwedo/economics/fuel_monitor/index.htm

The following assumption is used as the baseline data for manufacturing cost assessment of the wing box structure:

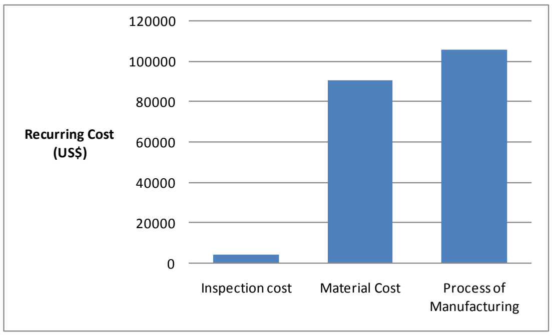


FIGURE E-1 RECURRING COST (USD) OF WINGBOX STRUCTURE

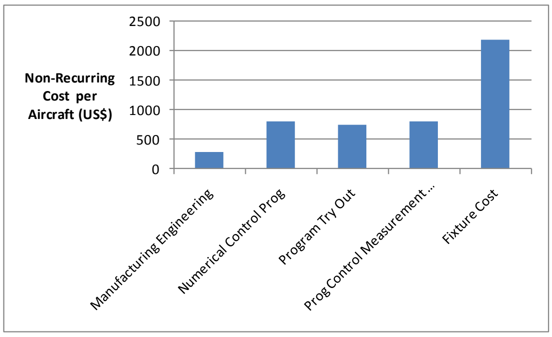


FIGURE E-2 NON-RECURRING COST (USD) OF WINGBOX STRUCTURE

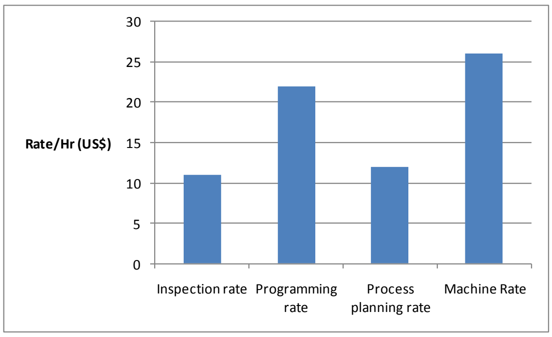


FIGURE E-3 COST/HOUR (USD) OF LABOUR AND MACHINE

Appendix F
Finite Element Model

COMPARING THE ANALYTICAL SOLUTION WITH THE FINITE ELEMENT MODEL

F. FINITE ELEMENT MODEL

The objective of developing a finite element model of wing box structure using finite element package PATRAN/NASTRAN and analysing the stress distribution is to validate the accuracy of the analytical solution developed in MATLAB and based on the thin cell theory.

This appendix presents the process of structure modelling of skin, spars and ribs of wingbox, and the assumption taken on the structure modelling. The result of stress distribution on each component is shown at the end of this appendix.

F.1 Skin, Spar and Rib Element Model

The shell element, CQUAD4 is used for the wing skin which represents the shell element within MSC/NASTRAN. Each element can be used to model membranes, plates, and thick or thin shells. Their properties, which are defined using the PSHELL entry, are used in conjunction with the membrane and bending properties of the skin.

Considering the experiences, the aspect ratio (or length/width) of the QUAD4 should not be greater than 3 to give a good stress distribution.

Considering the purpose of the elements, the spar elements are split between caps and web. The caps were designed to take horizontal tensile and compression (in-lane) loading and the web was designed to take the vertical shear load and some proportion of bending moments.

There are two ways of modelling the spar caps: firstly, by using the appropriate shell element and by meshing the upper and lower caps into several small elements; and secondly, by using the BAR element.

The consequence of the first is an increase in the number of elements on the skin to match the caps meshing, which is more time consuming. Employing the second way lessens the number of elements needed and therefore is selected.

The same as like spar the web element was modelled using CQUAD4 entry and the spar using the CBAR entry.

F.2 Results

Four examples of NASTRAN's images of the stress distributions on skin, spar and ribs at design load are shown in figures F-1 to F-4 Below. These show normal stress distribution on top and bottom skin, shear stress on spar web and shear stress on rib.

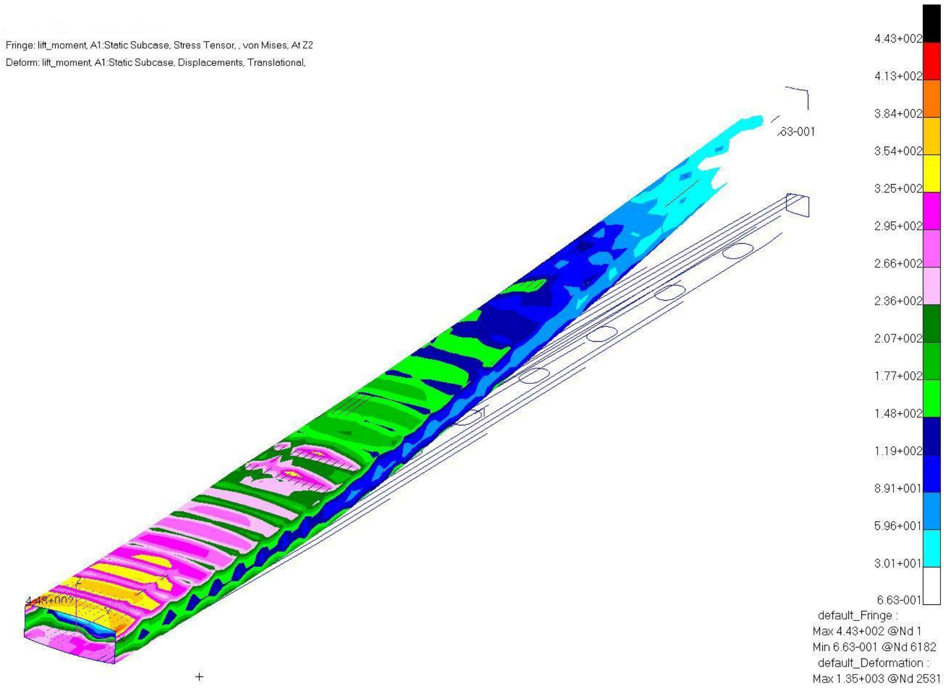


FIGURE E-1 STRESS DISTRIBUTION ON WING BOX AT LIMIT LOAD DUE TO GUST

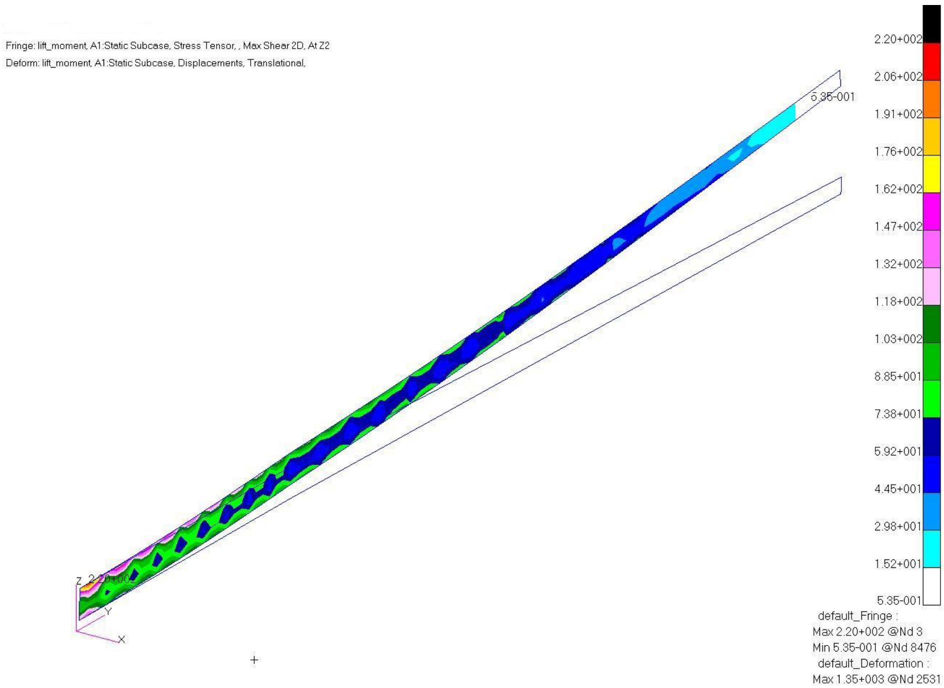


FIGURE E-2 STRESS DISTRIBUTION ON REAR SPAR AT LIMIT LOAD DUE TO GUST

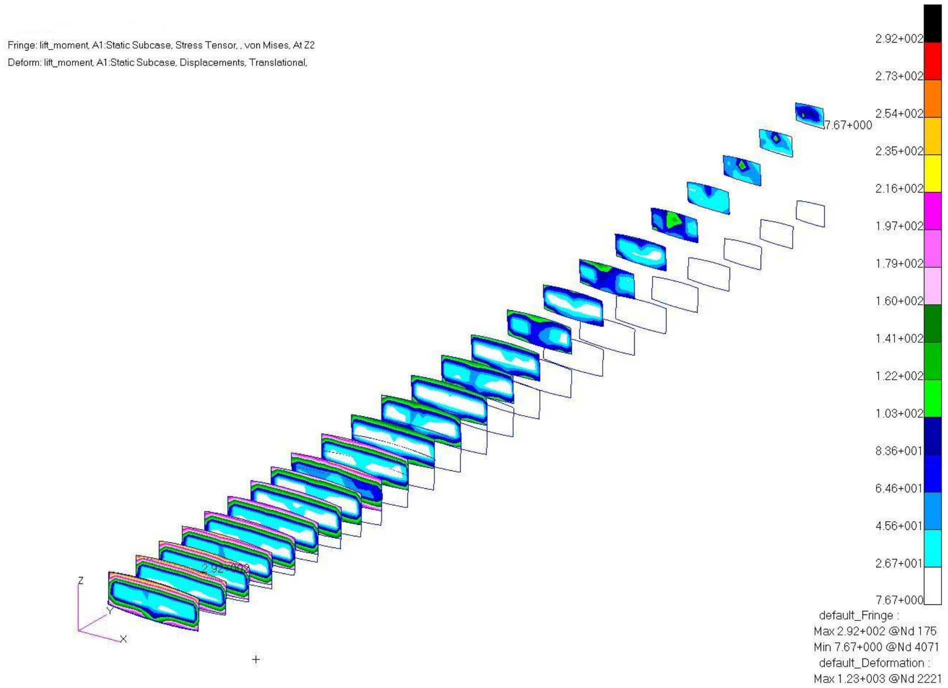


FIGURE E-3 STRESS DISTRIBUTION ON RIBS AT LIMIT LOAD DUE TO GUST

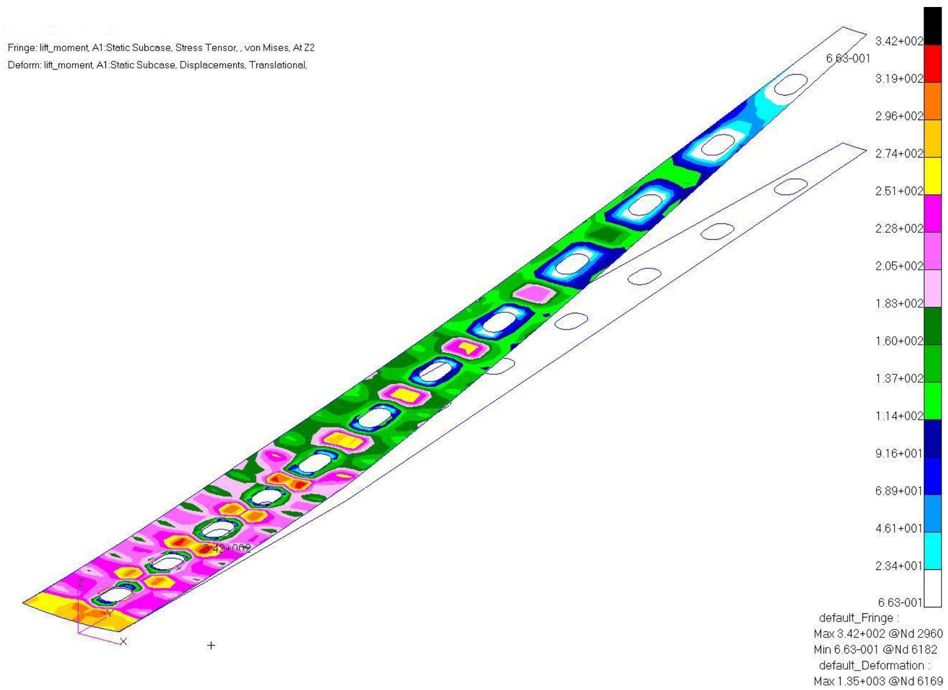


FIGURE E-4 STRESS DISTRIBUTION ON LOWER SKIN AT LIMIT LOAD DUE TO GUST

G. EXAMPLE OF USE OF THE SYSTEM IN A TUTORIAL CONTEXT

The following is another example of the use of the system in a tutorial context. The user is asked to design an upper skin panel of wing box structure of the same aircraft in the case study.

During concept generation stage, the tool provided information required by the inexperienced designer and students in solving various issues in generating the concepts of structure configurations, materials types, and manufacturing processes. In this example, the designers were provided with various type of skin-stringer panels for different aircraft.

Based on design loads acting on wing box structure, the upper skin-stringer panel is critical to compression stress. Therefore the selection of skin-stringer configuration will be based on the value of buckling efficiency as shown in the following screenshot:

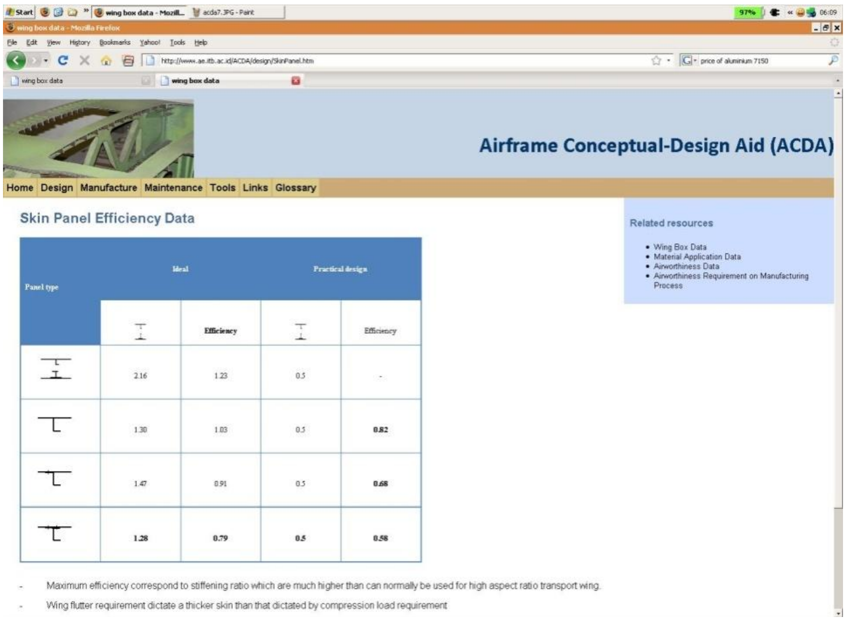


FIGURE G-1 SCREENSHOT ACDA: SKIN-STRINGER CONFIGURATIONS

They could also see the effect of different types of stringers on buckling effectiveness. The size of typical panels, including skin thickness and stringer sizing, were also provided for the user as their first estimate on the concepts that they designed.

Then they have to decide which material is suitable for the upper skin panel looking at the application on the existing aircraft. The properties is obtained from the link in ACDA to an electronic format of material handbook database 5H:

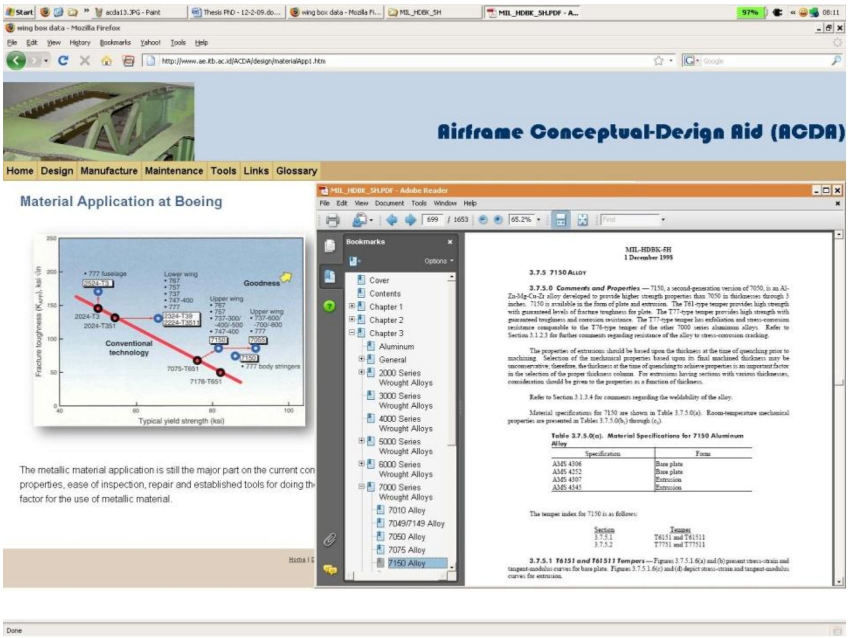


FIGURE G-2 SCREENSHOT ACDA: MATERIAL APPLICATION AND MIL-HDBK 5H

The material information database was critical during the conceptual generation stage. In industry, the designer usually goes to the company database or the material handbooks for selecting the candidates of materials for their design. The developed support tool in ACDA provided an additional checklist to designer so that they could explore alternative materials used by current aircrafts or those being developed by suppliers and possibly research institutions. This kind of practice, where designer was reminded to explore new materials will give an opportunity to improve their design relating to existing aircraft. This highlighted the importance of exploring new technology through bringing specific suppliers into the design process from the very early design stage. In addition, a close cooperation with supplier plays significant factors in deciding whether new technology will be available during at the end of design stage.

During the design process, the user could use the ‘links’, such as to FAA, or EASA to find relevant information on the airworthiness regulation for this aircraft. The ‘glossary’ menu in ACDA may also be used to get relevant meaning on certain terms used for skin panel design.

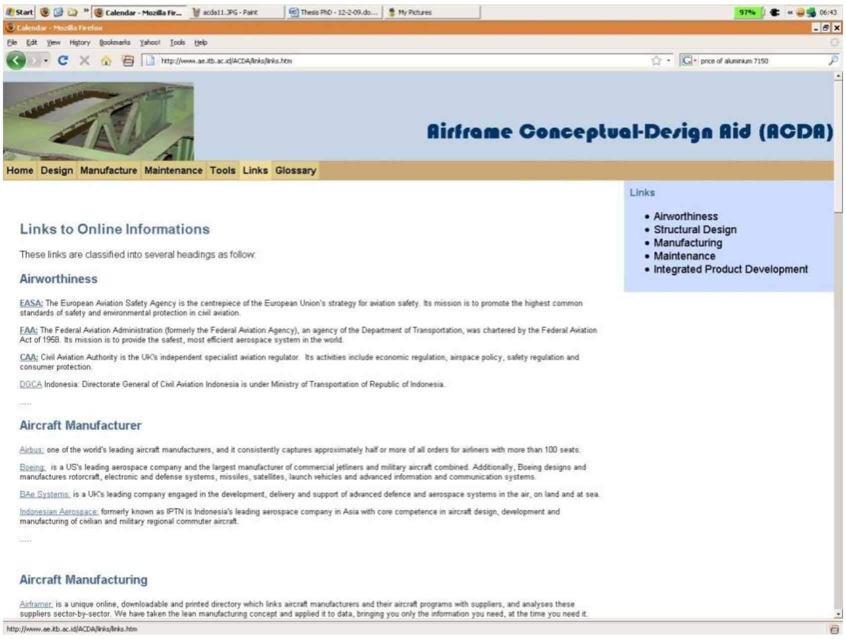


FIGURE G-3 SCREENSHOT ACDA: LINKS TO ONLINE INFORMATION

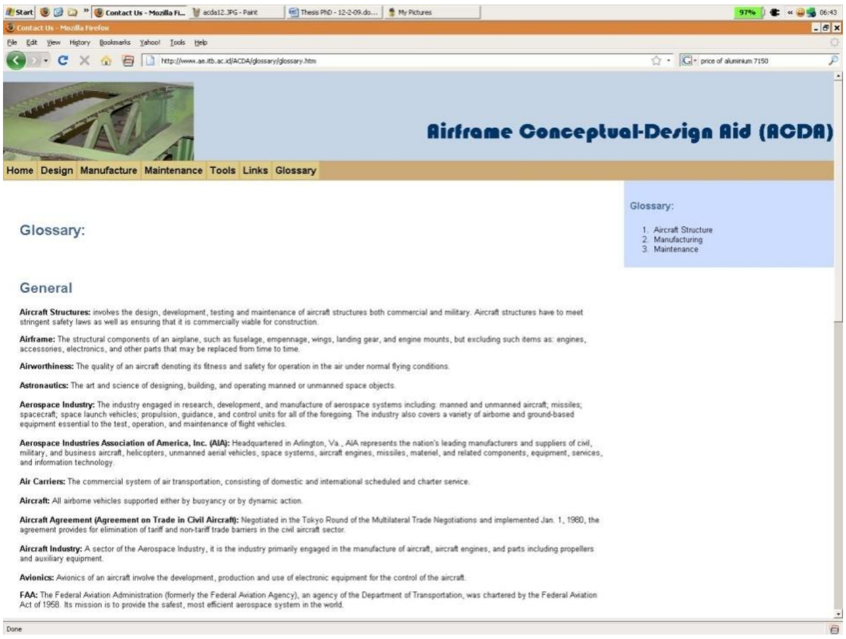
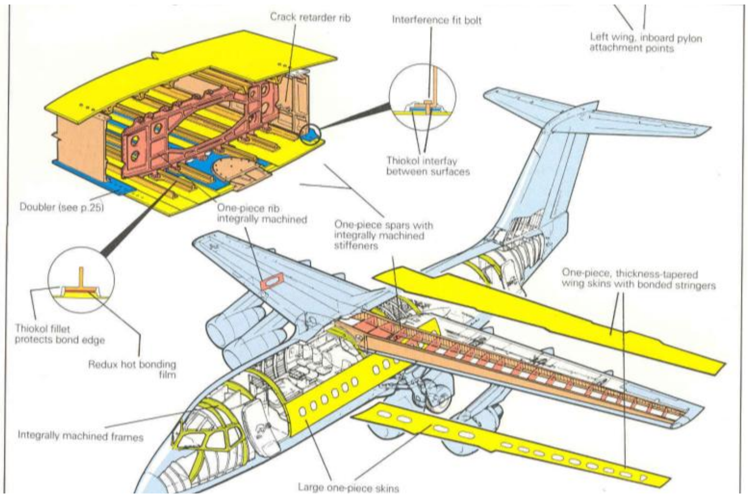
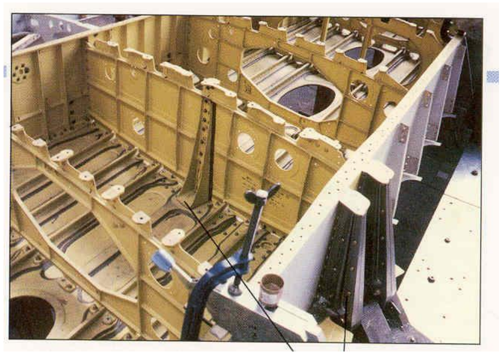
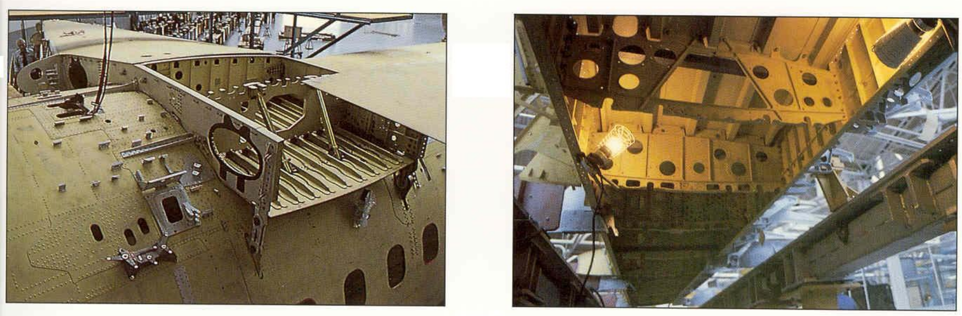


FIGURE G-4 SCREENSHOT ACDA: GLOSSARY

The information of existing wing box configuration available in ACDA, such as in the following figures, may be used as the example prior to creating the concepts:







One-piece upper wing skin is apparent as left wing is offered to wing box. Upper surface bonded stringers (right) serve as fuel tank vent pipes.

FIGURE G-5 BAE SALES BROCHURE ON RJ146 (BAE 1995)

During concept generation, different concepts for upper skin panel of wingbox structure can be produced. For inexperienced engineers and students the achievement of producing various concepts could be seen as their ability to explore many possibilities within their limited experience on previous projects. This not only gives the opportunity for the user to develop their knowledge on the latest technology in the airframe structural design but also speeds up their time in learning from the experience of senior designers and opens up the possibilities to gather much needed wisdom of their seniors.

The author felt that with a mixed process of using the database in ACDA and at the same time having 'real discussions' to more experienced engineers is the best way for the inexperienced engineer and students to acquire tacit knowledge, which is known to be difficult to acquire, from experienced designer. In this research no attempts has been made to automate the knowledge transfer using KBE's commercial tool.

Having all this information available to the user, then the concept selection, the technique proposed by Prof. Pugh were shown to be quite effective at the early stage whereas very little information is available to the user. However, the simplicity of the technique would then allow for a broad assessment based on structural performance, manufacturing and maintenance to be made. Similarly to the QFD technique, this concept selection process is ideally performed by a multi-disciplinary team. However, by performing the assessment, the user could then select the most feasible concept in a more effective way than just based on the tendency to follow the 'normal way of doing' in the company.

Then user can start to calculate wing load distribution to obtain SF, BM, and T load distribution along the wing span using Matlab Wing load analysis tool as explained in Appendix C. The wing load analysis tool, embedded in ACDA, incorporates the effect of the aerodynamics twist and the different type of airfoil on the load distribution. Therefore the designer could design the most suitable structural configuration according to the aerodynamics configuration of the wing.

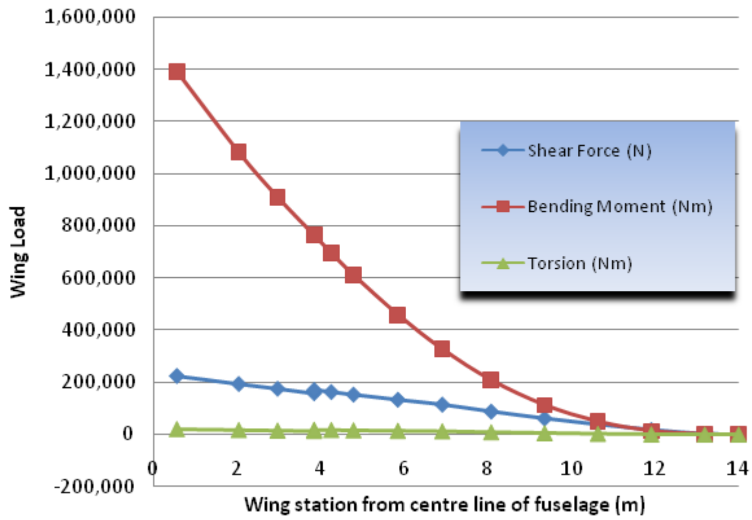


FIGURE G-6 WINGLOAD DISTRIBUTION

Then, the rest of the process, initial sizing, weight estimation, cost assessment and DOC analysis, is actually similar to the process as described in detail in chapter 4.