



CRANFIELD UNIVERSITY

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**An MDO Concept for Large  
Civil Airliner Wings**

CRANFIELD COLLEGE OF AERONAUTICS

PhD THESIS

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# Abstract

This thesis investigates the application of Multi-Disciplinary Design, Analysis and Optimisation to the design of a large civil airliner, similar in size as the future A3XX. For the first time structural optimisation, manufacturing cost and aerodynamic effects are simultaneously integrated within a realistic, complex aircraft design problem: the wing box of such a large airliner.

A novel multi-level system was developed to incorporate structural effects and manufacturing cost: mass is treated at a top-level while costs are treated at a structural sub-level. It allows a designer to study cost changes with respect to design changes and the interaction of cost with other disciplines such as structures and aerodynamics. The flexibility of the system allows companies to import their own results or cost data and to perform cost studies based on historical data or highly novel processes.

Structural optimisation of the wing box using MSc/NASTRAN and STARS, the development of a metal and composite cost model and the overall MDO methodology are being discussed.

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Kristof

The "MDO Project" (Multidisciplinary Design, Analysis and Optimisation of Aerospace Vehicles) was a collaboration between British Aerospace, Aerospatiale, DASA, Dassault, SAAB, CASA, Alenia, Aermachi, HAI, NLR, DERA, ONERA and the Universities of Delft and Cranfield. The project was managed by British Aerospace and funded by the European Union under the BRITE-EURAM initiative. (Project Ref.: Area 3: Aeronautics, Project No.:BE95-2056).

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# Notation

## Chapter 1

MDO Multi-Disciplinary Design, Analysis and Optimisation

## Chapter 2

$c_i$  Local analysis constraint  
 $F$  Objective function  
 $g_{ij}$  Compatibility constraint  
 $s$  Slack variable  
 $\bar{x}_i$  Domain specific design variable  
 $x_i$  Interdisciplinary coupling variable  
 $y_{ij}$  Coupling variable  
 $z$  System level design variable

## Chapter 3

TDMB Technical Data Modeller and Browser  
MMG Multi Model Generator  
 $M$  Bending moment  
 $F$  Force  
 $p$  load intensity  
 $\sigma$  Stress  
 $E$  Young's Modulus  
 $L$  Rib pitch  
 $\eta$  Efficiency of the compression panel  
 $j$  Safety factor  
 $t$  Thickness  
 $q$  Shear flow  
 $T$  Shear force  
 $I$  Moment of inertia  
 $h$  Height torsion box  
 $w$  Width torsion box

|               |   |
|---------------|---|
| <i>DOC</i>    | Direct Operating Cost                     |
| <i>OWE</i>    | Operating Weight Empty aircraft           |
| $\Delta OWE$  | the increment in OWE between the variants |
| <i>MTOW</i>   | Maximum Take Off Weight                   |
| $\Delta MTOW$ | The increment in MTOW                     |
| $\Delta Drag$ | The increment in drag                     |
| <i>i</i>      | Aircraft variant                          |
| <i>p</i>      | Parameter being varied                    |
| <i>w</i>      | Weighting factor                          |

#### Chapter 4

|                |                           |
|----------------|---------------------------|
| <i>f</i>       | Function                  |
| <i>x</i>       | Variable                  |
| <b>x</b>       | n dimensional vector      |
| $\nabla$       | Gradient                  |
| <b>H</b>       | Hessian                   |
| <b>[K]</b>     | Stiffness matrix          |
| <i>u</i>       | Displacement vector       |
| <i>P</i>       | Force vector              |
| <b>S</b>       | (Search) Direction vector |
| $\alpha$       | Step size                 |
| <i>W</i>       | Objective function        |
| <i>g</i>       | Constraint                |
| <b>I</b>       | Identity matrix           |
| <b>B</b>       | Positive definite matrix  |
| $\bar{\sigma}$ | Allowable stress          |
| <i>L</i>       | Lagrangian                |
| $\lambda$      | Lagrange multiplier       |

#### Chapter 5

|      |                                  |
|------|----------------------------------|
| CFRP | Carbon Fibre Reinforced Plastics |
| GFRP | Glass Fibre Reinforced Plastics  |

#### Chapter 6

|              |   |
|--------------|---|
| <i>i</i>     | Aircraft variant                          |
| <i>p</i>     | Parameter                                 |
| $\Delta W$   | Changes in aircraft weight                |
| $\Delta D$   | Changes in aircraft drag                  |
| $\Delta DOC$ | Changes in aircraft Direct Operating Cost |

# Chapter 1

## Introduction

*'MDO will provide designers with a new array of tools and approaches that will take them closer to the exclusive goal: an optimal aircraft...'*

[1]

### 1.1 Concurrent Engineering and MDO

On the verge of the 21st century, today's extremely competitive market has initiated major changes in the production and development of new products. Balancing the conflicting goals of systems superiority and systems affordability is currently a challenge larger than ever before. Hence, design for affordability through implementation of Concurrent Engineering (CE) and Integrated Product and Process Development (IPPD) environments has become a target activity for many industrial and governmental organisations. [2]

CE and IPPD embody the simultaneous application of both system and quality engineering methods throughout an iterative design process. Its application results in the time-conscious, cost-saving development of products and systems. [3] The IPPD approach is an expanded CE approach which includes also business practices as well as engineering manufacturing and support processes as part of the product development. [4]

CE and IPPD concentrates on global product optimisation and provides a systematic approach to the integrated and simultaneous design of products, taking account of their related processes such as manufacturing and support. The aim is to improve functionality, reduce costs and reduce time to market. Thus, the integrated design environment is a multidisciplinary concept, encompassing people, processes and tools. [2]



In order to obtain the concept of 'global optimum product', close collaboration between the various activities throughout the product life cycle is necessary. A competitive product quality, reduced product life cycle cost and reduced acquisition time can only be achieved if the simultaneous consideration of the different engineering and commercial disciplines occurs early in the product development process. Hence, from the earliest stages in the development of a new product, product designers are forced to consider all elements of the product life cycle: from concept to realisation, including quality, cost, schedule and user requirements. [4] [5] [6]

Within the Concurrent Engineering and Integrated Product and Process Development environment, Multi-Disciplinary Design, Analysis and Optimisation (MDO) has become an increasingly important methodology for solving complex engineering problems. It has turned into a 'key word' for today's high technological enterprises, such as the automotive, the aircraft and the space industry, which are systematically trying to apply the CE and IPPD philosophy. MDO has been defined as 'a formal design methodology based on the integration of disciplinary analysis, sensitivity analysis, optimisation and artificial intelligence, applicable at all stages of the multidisciplinary design' [1]. In other words, MDO is a way of getting engineers from various disciplines, such as aerodynamics, structures, control systems, propulsion, manufacturing, marketing, etc. to work together. The mechanism by which this 'working together' can be accomplished is by using mathematical methods (e.g. sensitivity analysis), modelling methods, expert systems and optimisation solvers. It offers new approaches for managing trade-offs in complex design situations where conflicting objectives drive the design in opposite directions [7]. The result is a process that can both reduce the design cost and the flow time, as well as improve the quality of complex engineering systems.

The Aerospace industry has proved to be a perfect field for the implementation of MDO. A typical design process for an aerospace vehicle involves:

- a large number of specialists from different disciplines (e.g. structures, aerodynamics, control, manufacturing, economic analysis, etc.; see figure 1.1)
- the transfer of mountains of data
- many design iterations
- multiple interactions where one specialist or specialist team is constrained by requirements from other individuals or groups. [8] [9]

Traditionally, these design problems with high degrees of complexity have been solved by focusing on the individual disciplines. Due to the theoretical,



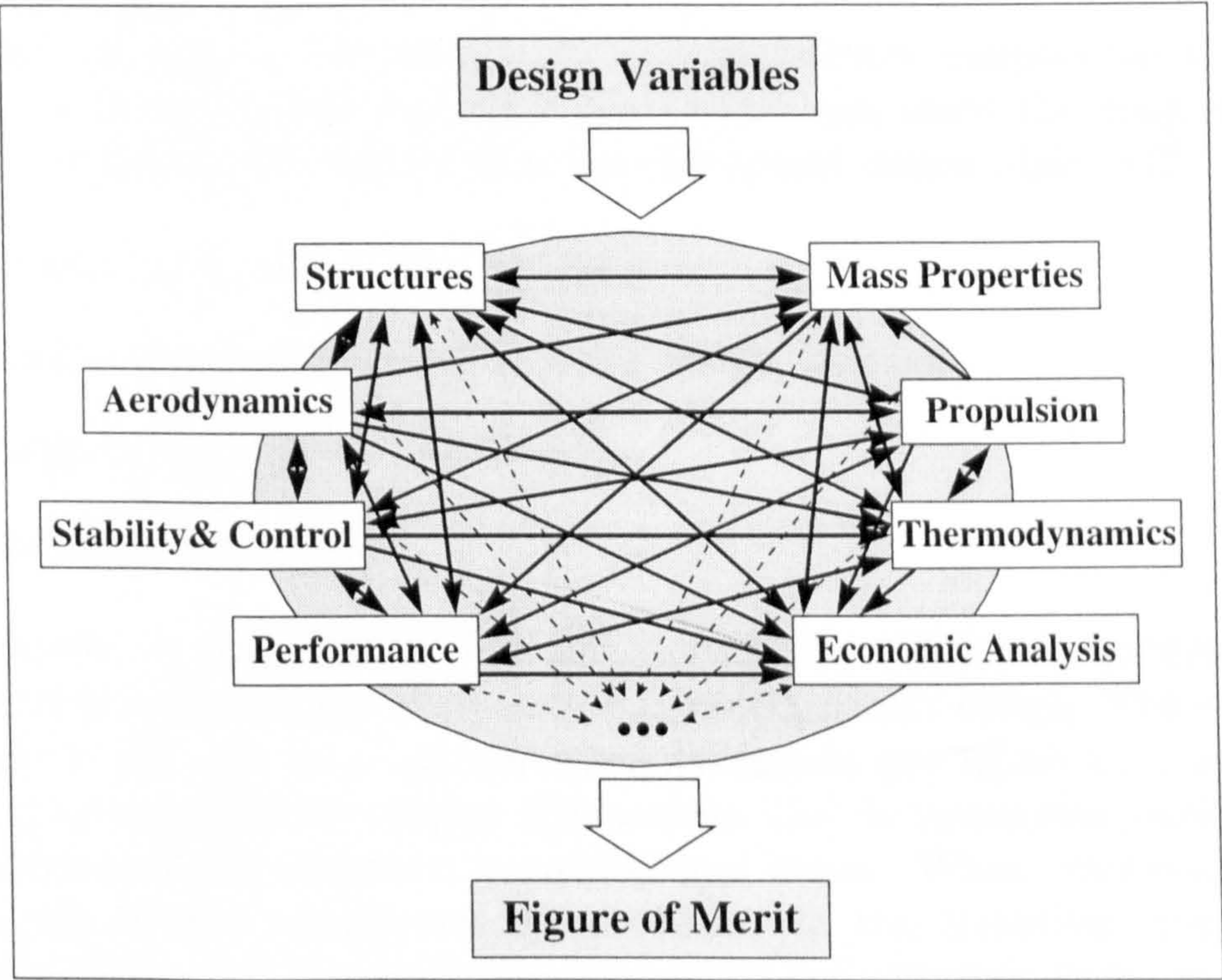


Figure 1.1: Disciplines involved in aircraft design and their interdisciplinary interactions



computational and methodological developments which have taken place in the past fifteen years, great advances have been made in the techniques for handling these individual disciplines. This has led to an increase in the number of functional disciplines which can now be simultaneously considered in today's aerospace vehicle design studies. As a consequence of this evolution, the complexity and improved accuracy of these sophisticated methods, used in each discipline, has increased the number of interdisciplinary couplings that must be resolved. Unfortunately, this has often resulted in a decrease in the awareness of the influence which a given specialist's decision has on other disciplines. [10] [11]

MDO will help to overcome these interdisciplinary complexities by providing a new set of tools and approaches which can assist the designer. In addition, applying this approach in the conceptual design phase will:

- reduce the development time-scale
- lead to more accurate preliminary design decisions
- improve the aircraft performance
- reduce the cost. [12]

However, as promising as the MDO prospective might be, progress has been slow in achieving the objectives of interdisciplinary design. The obvious difficulty is the vast scope of individual disciplines (see figure 1.1 ) and the inability to comprehend complex interactions. [13] As mentioned, each of the disciplines presents numerous computational issues. When combined their interaction further complicates the problem. In this situation, defining a merit function, a set of appropriate constraints and selecting design variables that can handle the requirements of all the competing disciplines, is extremely difficult. These difficulties are not only limited to aerospace vehicle design, but are common to the design of automobiles, civil and mechanical products and in process control.

In order to overcome these difficulties, some realistic practical applications are required. As was identified by Schrage [4] before 1994 the CE, IPPD and MDO research focused too much on tool and technique development without overall methodology and realistic focus. Real world, large scale design problems were required so these approaches could be extensively tested, evaluated, improved and developed. Only in this way these approaches will prove their potential and will they achieve sufficient robustness so they can be implemented in an industrial environment. Hence the recent application of CE and MDO to challenging realistic design problems such as the High Speed



Civil Transport Aircraft [4] [14] [15], Joint Strike Fighter Aircraft [16] and Large Civil Transport Aircraft (A3XX) [17] [18][19] [20] [21]. It is in this context that this thesis is situated (see thesis objectives, section 1.2).

Whilst the problems related to CE, IPPD and MDO are clearly complex, recent research has given rise to encouraging results and very promising steps have been made towards a full MDO system. However, we are still far from the ultimate goal...

## 1.2 Thesis objectives and general conclusions

Over the past years many Multi-Disciplinary Design and Optimisation methodologies have been developed, tested and published. Only recently MDO has been applied to realistic large scale design problems (see section 2.3 on page 28).

The thesis investigates the application of MDO for the design of a large civil airliner, i.e. an aircraft the size of the future Airbus A3XX. To reduce the complexity of the problem, only the aircraft's wing was considered. The research has focused on three main areas:

- Structural optimisation of the wing.
- Development of a manufacturing cost estimation capability for MDO and its integration into an MDO environment.
- Development of an MDO system which performs combined structural, aerodynamic and manufacturing cost studies.

For the design and cost studies, six overall (i.e. geometric) design parameters were selected which consist of sweep angle, aspect ratio, planform area, spar position, wing depth and wing twist. In addition two internal layout parameters, stringer pitch and number of ribs were also considered.

A large part of the research has been performed in a distributed MDO environment, interacting with people from several European aerospace companies and research institutes as part of a European MDO project. This did not only improve the validity of the results presented, but also lead to an appreciation of other MDO aspects such as exchange and generation of results and data, control of the overall design process, company organisational aspects etc.

As a member of the European MDO project team a lot of information which was mentioned in this thesis, such as the MDO software tools, definition of the reference design problem, design and analysis assumptions, etc.



has been the result of a team effort. However in parallel the author has been exclusively responsible for the following contributions:

- Development of recurrent manufacturing cost models for metal and composite wing structures, using a feature based approach.
- Integration of manufacturing cost information into an MDO problem.
- Development of a multilevel system which, for the first time, allows for the simultaneous analysis of manufacturing cost, mass and drag.
- Structural optimisation of a large realistic aircraft wing using STARS and MSC/NASTRAN

It are these contributions which formed the novelty of the performed research.

The objectives of the thesis were successfully achieved. Convergence of the structural optimisation proved to be very sensitive to design model definition and specific solution parameters. However, once the optimisation process was under control, successive design studies were performed without any problems.

Two recurrent cost models, a prototype metal cost model and composite cost model were developed. The validation process showed that both models are of a high level nature and are suitable for preliminary design tasks. Nevertheless, the models allowed the investigation of important MDO wing design problems such as stringer pitch and rib pitch optimisation. Successful integration in the MDO environment proved their potential for future more detailed applications. It allows a designer to evaluate cost changes with respect to specific design changes.

An MDO paradigm which allowed for combined mass, drag and recurrent manufacturing cost optimisation was developed and tested. It allowed for stringer pitch optimisation, combining minimum cost and weight, and it was demonstrated that manufacturing cost can be dealt with at a sub-level and does not have to be considered at the same level as mass and drag.

### 1.3 Thesis structure

The structure of the thesis aims at giving a chronological discourse of the performed research.

*Chapter 2* gives a general overview of Multi-Disciplinary Design and Optimisation. It describes different aspects of an MDO process, MDO methodologies and how MDO can be implemented. The chapter concludes with a summary of current MDO applications.

Having discussed MDO in general, *Chapter 3* focuses on the specific MDO implementation on which this thesis is based: a European large scale aircraft design problem. An overview of the European MDO project, the definition of the design problem and a discussion of the tools which were developed and applied is given.

*Chapter 4* further investigates one specific aspect of this MDO problem: the structural optimisation of the large aircraft wing box. It mainly reports the structural optimisation using MSC/NASTRAN and STARS. First of all the optimisation problem is described, followed by a theoretical overview of the optimisation algorithms which were applied and a selection of results. Problems which were encountered during the optimisation are also reported.

With the structural optimisation process under control, *Chapter 5* looks at the development of a manufacturing cost model which allows for integration into an MDO environment. After a general introduction on cost analysis and related aspects, the chapter explains two manufacturing cost models: a prototype cost model for metal wings and a composite cost model. Results for both models are shown and cost analysis on a combined metal/composite wing is performed.

*Chapter 6* describes the integration of the manufacturing cost models into an MDO environment and investigates an MDO methodology for performing combined aerodynamic, structural and cost analysis. Apart from an outline of the overall methodology and description of the general results, a more detailed investigation has been carried out on the optimisation of stringer pitch.

The discussion is completed with conclusions and recommendations for further research in *Chapter 7*.



# Chapter 2

## Architecture of MDO process

This chapter will give a general outline of Multidisciplinary design analysis and optimisation. It will describe what is required and must be taken into account when developing an MDO methodology. First a general discussion of different MDO aspects is given, followed by an overview of MDO methodologies which have been developed and examples of MDO applications.

### 2.1 Aspects of MDO

Design is an optimisation process which employs large quantities of physics data from engineering disciplines in order to develop a manufacturable and supportable product. This product has to meet the customer's performance and life cycle cost objectives. [22] The increasing emphasis on affordability in aerospace vehicle design necessitates reduced design cost and design cycle times, with integration of the system cost estimates into the aircraft design process. In order to achieve this, several aspects are required from a Multidisciplinary Design, Analysis and Optimisation process:

- Reduction of the design and analysis cycle time in all 3 phases of the design process: conceptual, preliminary and detailed design.
- Increase the number of disciplines (manufacturing, supportability, cost, stability and control, etc.) considered in the optimisation process .
- Increase the fidelity of the different design and analysis models ( structural, aerodynamic, dynamic, etc.) earlier in the design process, in order to achieve more accurate estimates of parameters such as weight, drag and cost for conceptual design trades.



- Integrate conceptual and preliminary trades with design to cost analysis to define the designs with either "best value" (optimal balance of cost and performance) or minimum cost (with performance compromised).
- Tools which allow for MDO and multi-discipline trade off analysis.
- Change in organisational architecture of company, recognising that MDO does not only encompasses processes and tools, but also people.

Several of these topics will be addressed in greater detail in the next paragraphs.

### 2.1.1 Increase number of disciplines

Traditionally, many MDO problems have mostly studied minimising weight and drag only. For a realistic aircraft design problem many more disciplines such as stability and control, propulsion, manufacturing, supportability and cost need to be involved. This, however heavily increases the complexity of the MDO problem. It also strongly influences the MDO architecture and raises issues such as coupling of disciplines and problem decomposition. Increasing the amount of disciplines also increases the amount of possible design variables to be studied in the MDO process. The paragraphs below will discuss these issues in further detail.

#### 2.1.1.1 Problem decomposition

Aircraft design is a complex process which contains many interactions between all of the participating aerospace disciplines. It is practical impossible to describe such a design and optimisation problem by a single analysis model. Problem decomposition aims to reduce a complex system into a set of multiple subsystems. A lot of research on problem decomposition has been performed by Sobieski [23] [24] and Renaud [7]. The following definitions are obtained from Koch [25]:

- **Complex System**

*A system which is composed of a number of subsystems, where each subsystem is embodied by a particular set of components or subsystems. Each component has its own work principle. The system, subsystems, and components involve the interactions of multiple disciplines. A complex system is multidisciplinary in nature. (e.g.: aircraft)*



- **Subsystem**

*A part of a system which is considered a system itself (comprised of multiple components or component groupings - sub-subsystems - itself; e.g.: an aircraft wing.)*

- **Component**

*A discrete physical entity that constitutes one part of a subsystem or system (a constituent element; e.g.: a rib of the aircraft wing.)*

Complex products are often represented through a hierarchical structure of information and these structures drive the processes by which these products are designed and developed. Although the above stated definitions are for *complex products*, they also apply to *complex problems*. This, because of the multidisciplinary nature of complex systems in engineering design, where the decomposition of a system can follow either the physical structure of the system (subsystem/component definitions) or the disciplines involved in designing the system.

The goal of decomposition is to break the system or problem along lines (disciplinary or physical) that make the subsystems or sub-tasks as independent and self-contained as possible, and to minimise the number of couplings which complicate the system level problem. Two basic classes of decomposition methods exist [26]: formal methods and intuitive or heuristics methods. For formal methods, the decomposition structure is derived from a mathematical definition of the problem. Intuitive or heuristic approaches are based on the natural functional or physical partitioning of a system into its sub systems.

#### **2.1.1.2 Hierarchic & non-hierarchic systems**

When looking at system decomposition, the issue of system hierarchy appears. A hierarchic system, is a system where there are one or more levels of "parent-child" relationships. In a hierarchic system, the information flows from parent (i.e.  $P_1$ ) to child (i.e.  $P_2, P_3$ ). This is shown in figure 2.1 A non-hierarchical system, however, is characterised by information flowing between all the boxes; i.e. parent and children. [24] [27] [28] Many problems can not be transformed into a simple hierarchic decomposition scheme. An example of such a problem is the optimal design of an active controlled , flexible wing (see figure 2.2). For this type of problem, a lateral link between aerodynamics and structures is required. Hence there is no simple link between parent and daughter, as information must be also transferred between the daughters. A top-down hierarchic structure is not anymore possible.

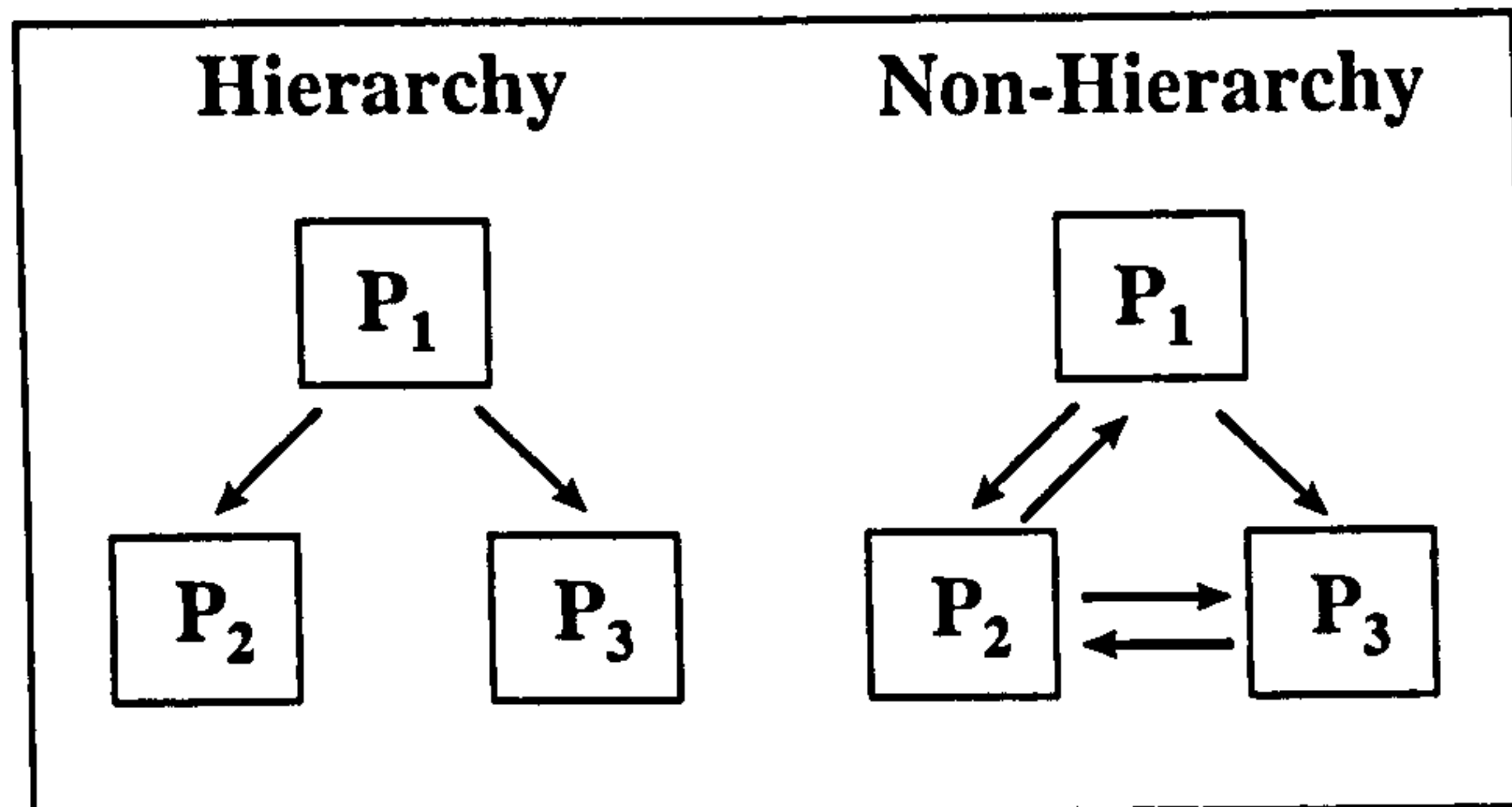


Figure 2.1: Hierarchic and non-hierarchic system.

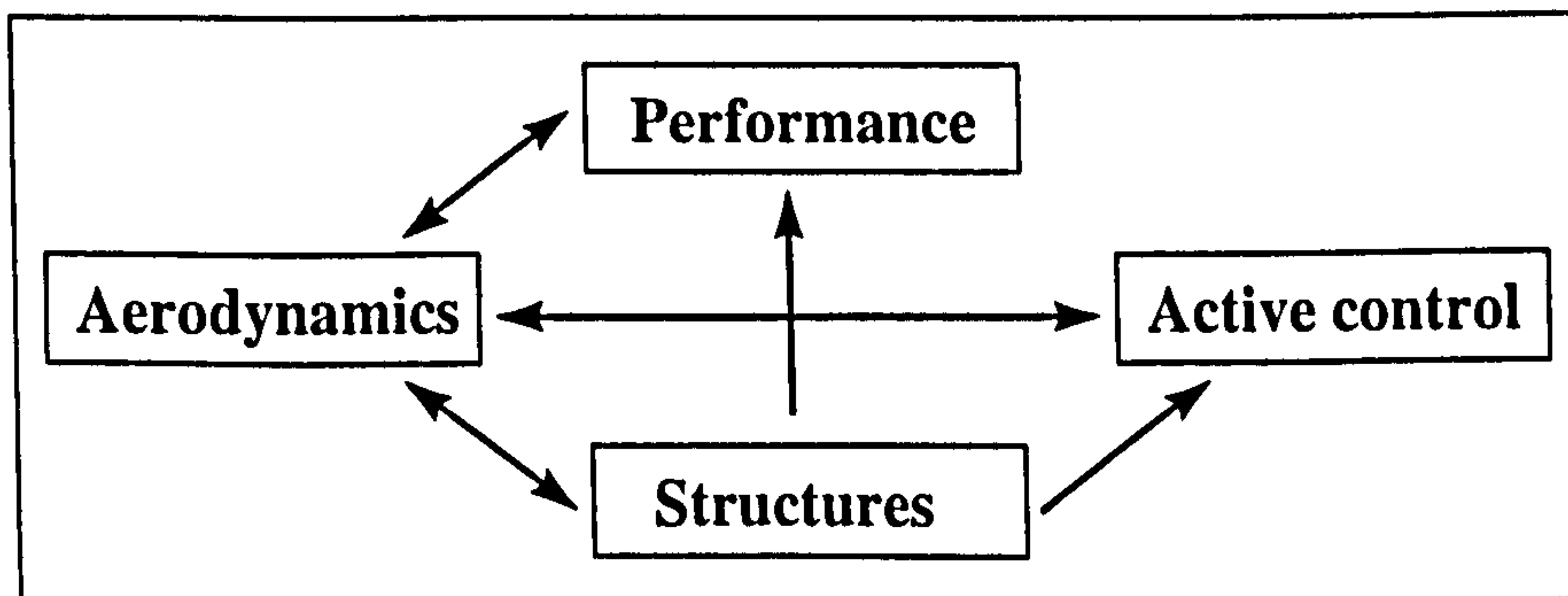


Figure 2.2: Problem decomposition for flexible wing.



### 2.1.1.3 Practical implication of decomposition for a real MDO problem

In real MDO problems the strict definition of hierarchy and formal decomposition methods, where the analysis of a hierarchical problem proceeds sequentially from master (parent) problem to one or more sub-problems does not often apply. It is more common to have problems where both vertical (between different levels) and lateral (at same level) interactions occur. In this case problems cannot be solved sequentially, as each problem requires information from the other, and iteration must occur. [25]

Complex, realistic MDO problems do not easily allow for decomposition using formal (mathematical ) decomposition, as it is not always clear how and to which extent certain disciplines are coupled. Manufacturing is a good example. What is the relation of manufacturing cost with respect to other disciplines such as structures (weight) and aerodynamics (drag)? One could argue that there is a tight coupling between structures and manufacturing and only a small or even no coupling between aerodynamics. However, if a certain manufacturing procedure has been chosen, because it is the cheapest from manufacturing cost point of view, a higher weight might be associated with this choice which will not only affect structures but also the aerodynamic behaviour.

Clearly, there exists a tight coupling between structural analysis (weight) and the calculation of recurrent manufacturing cost as optimised weight is an input to calculate the material cost in a recurrent manufacturing cost model. But it is not only weight which has an influence on both disciplines, as discussed further in this thesis also geometry parameters and specific manufacturing rules play an important role. It is also not straightforward if there exists a hierarchy between structural and recurrent cost analysis or if these 2 disciplines can be dealt with at the same level. The problem becomes even more complicated when one decides to expand the manufacturing cost to include non-recurrent cost, maintenance cost, life-cycle cost etc.

A main part of the work presented in this thesis addresses the issues mentioned above. It was chosen to use a method which requires a multi-level process in which manufacturing cost are taken into account, primarily at a sub-level. By proceeding in this way there is no requirement to incorporate manufacturing costs into a top level direct operating cost. The drawback of this approach is that cost now plays no role in the final assessment of direct operating cost and this could be seen as a weakness. Chapter 6 will discuss this in further detail.

#### 2.1.1.4 Variables for MDO analysis

With the increase in disciplines, involved in an MDO process, the number of design parameters which have a possible influence on the design increases as well. However, not all design variables have the same degree of influence on the design process. Some variables are only related to one discipline, other variables are shared between different disciplines. An important step, at the start of an MDO process is the selection of the driving design parameters. A screening process of all the variables is necessary to decide which variables will drive the multidisciplinary design process. This decision can be based on company experience, or by using screening experiments as described by Koch [25]. The purpose of screening experiments is to identify the most important parameters in a design problem and to reduce the initial set of total parameters. Screening is not always a straightforward process and often a compromise needs to be made between accuracy (more parameters than desirable but higher accuracy) and manageability.

#### 2.1.2 DOC selection

The objective function for the multidisciplinary design and optimisation of transport aircraft is often the minimisation of direct operating cost (DOC). Using DOC as an objective function is difficult as it is not always clear which parameters to select and how these parameters influence the DOC. For a commercial aircraft, a measure of "best" can be the minimisation of a DOC which includes the cost of ownership, maintenance, fuel, fees and crew cost. [29] But how can we bring in items such as material and manufacturing cost? Often this was done, by relating weight to manufacturing cost, but clearly this can be misleading since actual fabrication costs depend also on the number and complexity of parts and operations. To avoid many of the complex problems which are associated with using DOC as an objective function, the maximum take-off gross weight (MTOGW) is often used as simplified objective function. This objective function gives equal weighting to fuel, structural weight and payload. Although it does not accurately reflect DOC, as it is only vaguely related to it through fuel and structural weight, it seems to produce results that are satisfactory. Several types of DOC formulations can be used and it needs to be decided whether one wants to find the designs with either "best value" (optimal balance of cost and performance) or minimum cost (with performance compromised).



### 2.1.3 Increase fidelity of design and analysis models

Decisions made during the conceptual and preliminary design process will control the major cost and performance characteristics of an aircraft. Poor decisions will constrain the development of the aircraft for the next 30 years, while good decisions open up many options and opportunities. The dilemma facing the aircraft industry is that good decisions require high-fidelity design information to direct them. But it is just such information which has traditionally been time consuming and costly to generate. In particular it has been impractical to update this detailed information as the top-level aircraft specification and technical definition evolves. [17]

For a multidisciplinary process, one would like to have as much detailed information as possible, for example it would be preferable to use full structural optimisation. With the current computing facilities the solving time of high fidelity models has improved a lot. However, when introducing high fidelity models one has to closely monitor two issues:

- Unacceptable slowing down of the design cycle.
- Increase in the complex interaction of design variables and increase in the number of design variables.

[22]

The introduction of high fidelity models and tools into the MDO process also requires thorough understanding of these models. Traditionally these models would have been used by an expert. Introducing these models forward into the design process one cannot require from the project engineer to be an expert in each of these tools. Preferably the designer would like to have these tools integrated as 'black boxes' which have been set up by an expert and can be 'plugged and played' by one press on a button. However this exposes risks such as:

- Misinterpretation of the analysis results.
- Lack of visibility of the limitations by the analysis.
- Progressive loss of understanding of the subtlety in the detail of the analysis.

To reduce these risks, direct involvement of the specialist, working alongside the project engineers is necessary. In this way specialists can monitor and advance the analysis standards and could quickly develop and integrate new ideas from the project engineers into new analysis methodologies. [30] [31]



Allwright [17] also highlighted another problem related to introducing high fidelity models in the multi-disciplinary design process: a major bottleneck was identified in the updating of the different analysis models. It was possible to quickly generate superficial models automatically, or detailed models interactively given time, but there was no intermediate capability to generate or update analysis models which were appropriate for comparative preliminary design within the time scales and resources demanded by the project programmes. This problem is specifically related to tool development for MDO, which will be discussed in the next paragraphs.

#### 2.1.4 Tools for MDO

This section will describe what is needed to implement MDO. Many of the comments and items discussed were obtained from Allwright, MDO integration and recommendations [30]. The interdisciplinary coupling inherent to MDO tends to present additional challenges beyond those encountered in a single-discipline optimisation. It increases computational burden, increases complexity and creates organisational challenges for implementing the necessary coupling in software systems. [32] From organisational point of view, the analysis codes of each discipline have to be made to interact with one another for the purpose of system analysis and system optimisation. The complexity of the interaction is determined by the MDO formulation. Decisions on the choice of design variables and on whether to use single-level optimisation or multilevel optimisation have serious effects on the coordination and data transfer between analysis codes, the optimisation code and on the degree of human interactions required.

A successful MDO process strongly relies on the implementation and development of a specific set of tools, such as:

- A common electronic database.
- Computer aided design tools.
- Analysis tools.
- Integrated process management.
- Integrated control system [2].

As many disciplines, persons, departments and other companies or organisations can be part of the multidisciplinary design effort, it is necessary to have a common electronic data base, which is not only able to store the

initial data, but which allows for storage and comparison of results from different disciplines or design cycles. The high-fidelity simulations that underlie MDO are inherently computationally expensive and require adequate number crunching and data storage facilities (computational power = interaction = learning = design and process improvements). A strong set of tools is needed which can create in short time new, updated design models for the different disciplines and their analysis tools. To coordinate and steer the design process a robust control system is needed which is placed at the top of the design system and which drives the optimisation process. Because of the large number of people and disciplines involved, it is necessary to monitor and log the state and version of all the data and results generated. For this task, a process management system needs to be integrated to ensure that everybody works with the most recent data and result set. It must also be able to record the design history as this forms the design audit trail (Who did what, when?). The system also has to monitor the progress of the different disciplinary analysis and has to monitor the information flow, so that a new process can only start when all necessary input data has been received from the previous processes. It also has to be capable of managing distributed processes, located on different computers or at geographically different locations. Privileges also need to be introduced which allow for example the project team to have control of the whole design process, the design parameters and data, while people from the individual disciplines have only full control on their own discipline and restricted access to those of others.

The MDO process is an evolutionary process by its very nature. Hence the MDO infrastructure must be developed in such a way that it allows for easy further development and evolution of current processes. Inter-operability - plug-and-play are important factors which need to be included. An MDO process has to allow the developer to easily introduce, test and evaluate new or updated modules. The system has to be flexible so different aircraft projects and different stages of a single aircraft project which may require different modelling and analysis standards can be implemented.

An MDO infrastructure must make it easier for engineers to do a better job and must be accessible to them. Company requirements, strategies, policies and systems will all play a critical role in determining success or failure in the deployment of an MDO infrastructure within an organisation. If all these conditions are right, the deployment of MDO infrastructure and methodology in the company can be 'self-propelling', as it will automatically grow and propagate. Two possible scenarios for failure in MDO deployment can occur and must be avoided by all cost:

- An MDO system which has full company backing and meets the user



requirements, fails to deploy because it fails to make it easier for an engineer to do a better or quicker job. Therefore, it does not get used when an engineer is under pressure on a project.

- An MDO system which has the required functionality to help engineers to do a better job fails to deploy, because the impact of its deployment on other hardware and software systems is not anticipated and planned for at company level and the deployment is inadequately resourced and integrated.

### 2.1.5 Change in organisational architecture of company.

MDO does not only encompass processes and tools, but also people. The interaction between the different modules in the MDO software system on one side and the multitude of users organised in disciplinary groups on the other side may be complicated by departmental divisions in the organisation that performs the MDO. [32] Successful implementation of MDO depends on the recognition that people are one of the most important factors in this design methodology. For MDO to succeed, an organisational change in the company is needed. MDO requires people from different disciplines and background which are highly motivated to work as part of a team on a common goal: obtain the best, most cost effective design. Specialists and project engineers must be enabled to work together as a team. To make this 'teaming' happen, it is required that the concurrent engineering approach is initiated throughout the departments and companies involved in the MDO process. Good communication skills within the team are vital: engineers must not only understand their own role in detail, but must also be able to explain their specialist roles to their colleague engineers, often hiding the complexity of what they do so that more time can be spent on evolving the links to establish a well understood and highly effective multi-discipline design and simulation concurrent engineering process. Increased understanding of design issues and of other discipline skills must not be misinterpreted as a dilution of core specialist skills, but as an enhancement of skill. [30] [31]

## 2.2 MDO methodologies

In order to describe and control the complex problem of aerospace design, several multidisciplinary design and optimisation methodologies have been proposed and developed over the past years. A lot of work in this area has



been carried out by NASA's Langley Research Centre by Sobieski et al, the university of Stanford by Kroo et al. and by Renaud et al. at the university of Notre Dame. This section will give a brief overview of the main MDO methodologies which have been discussed in literature. Reviews of MDO problem formulations have been presented by Sobieski [32], Balling [33], Cramer [34], Kroo [35, 36], Braun [37], Renaud [7] and Koch [25]. Figure 2.3 gives an overview of the different approaches that have been considered. As can be seen from the figure, the MDO methodologies can be classified in several ways.

One way to classify them is to group them in single and multi-level optimisation approaches. The single optimisation approaches employ a single optimiser to solve the design problem. The multi-level optimisation approaches divide the design problem into sub-problems, which are optimised independently by separate optimisers, and which are controlled and driven by a system-level optimiser.

Another way to classify the MDO methodologies is group them into All-at-Once (AAO), Multi-Disciplinary-Feasible (MDF) and Individual Discipline Feasible (IDF) approaches. In this case the classification is made based on the distribution of the sub-problem analysis, feasibility evaluation and optimisation.

The latter classification is used to describe the different methodologies. As first approach, the standard classical formulation is discussed, however distributed architectures such as the AAO, MDF and IDF have numerous organisational and computational advantages over the standard approach [38] [34] such as:

- Providing a more natural fit to the current disciplinary expertise structure found in most design organisations.
- Allowing disciplinary experts to influence the design process (through subspace optimisation)
- Flexibility to efficiently alter a part of the design process, without having to repose the complete problem.
- Reduction in the integration and communication requirements.
- Use of a parallel optimisation architecture which can be used on a suite of platforms.
- Removal of iteration loops.

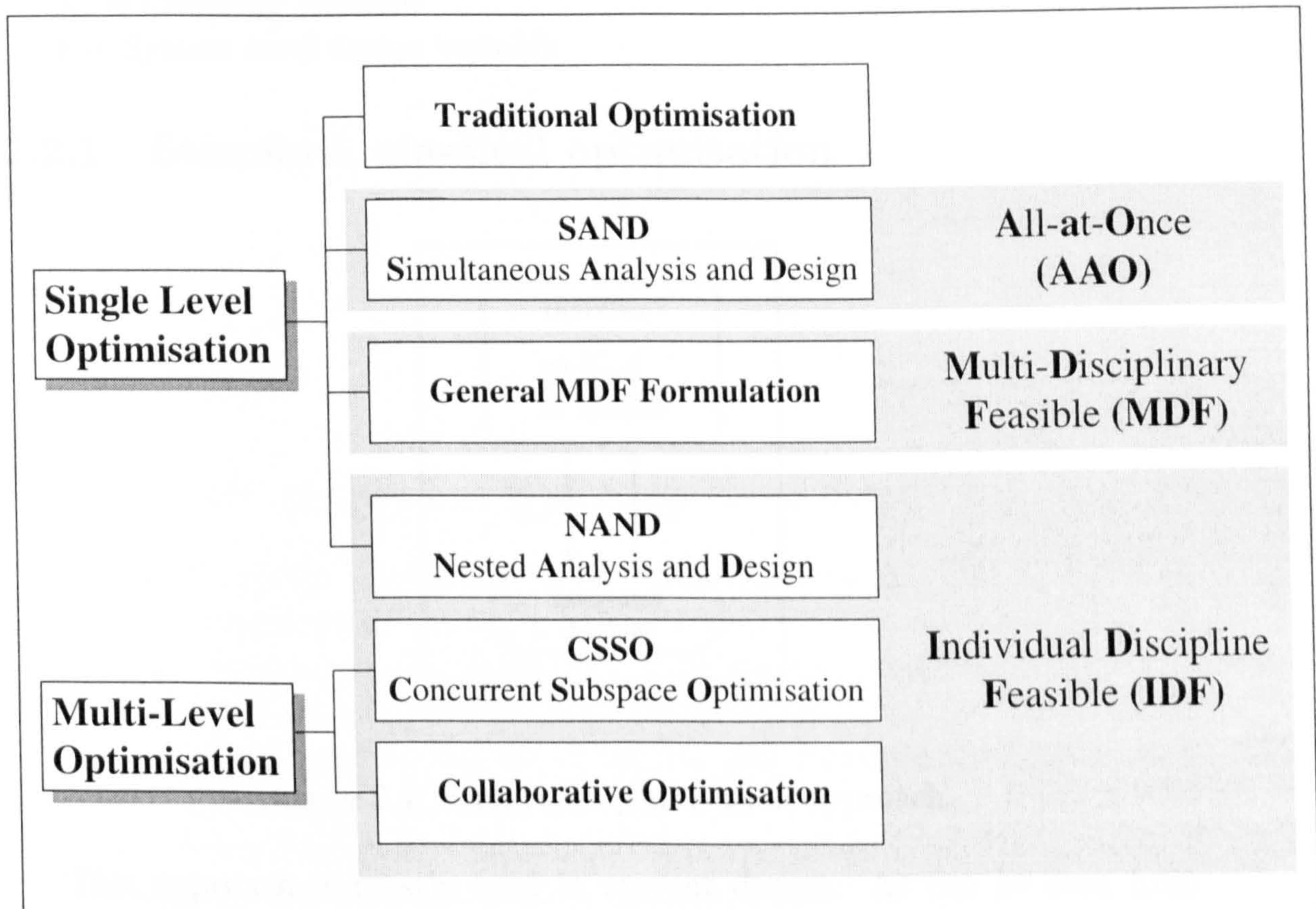


Figure 2.3: Classification of MDO methodologies.



The following symbols are employed in the discussion below:

- $c_i$  = Local analysis constraint
- $F$  = Objective function
- $g_{ij}$  = Compatibility constraint
- $s$  = Slack variable
- $\bar{x}_i$  = Domain specific design variable
- $x_i$  = Interdisciplinary coupling variable
- $y_{ij}$  = Coupling variable
- $z$  = System level design variable

### 2.2.1 Standard, classical optimisation

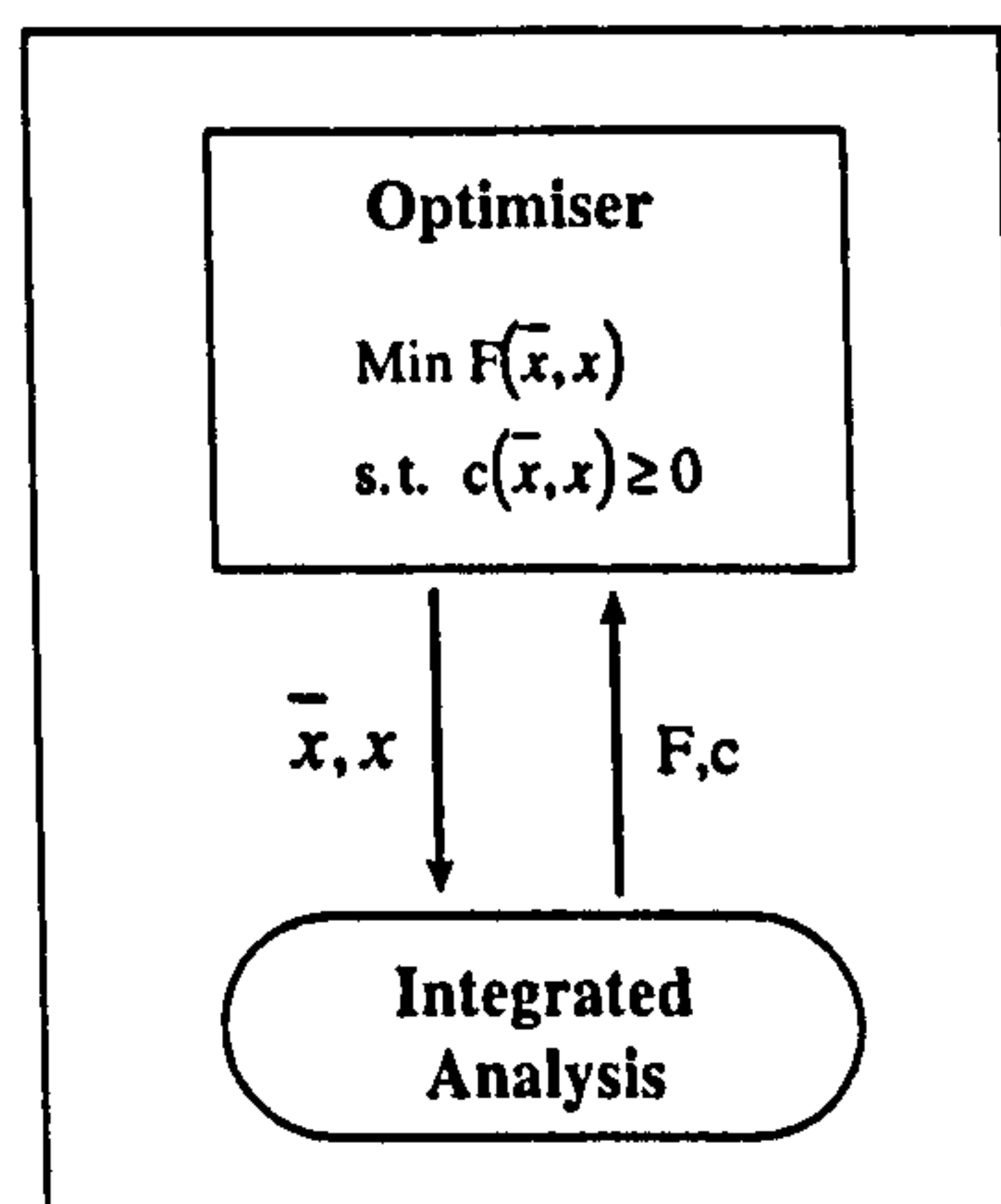


Figure 2.4: Standard optimisation approach.

This approach is mostly used in current design. As can be seen from figure 2.4, this approach requires an integrated set of analysis models, such that for a given set of design variables ( $\mathbf{x} = [\bar{x}_i; x_i]$ , i.e. domain specific variables and interdisciplinary coupling variables) the analysis returns the values of the constraints  $\mathbf{c}$  and the objective function  $F$ . Basically, the MDO analysis is treated as a single black box and the interdisciplinary coupling is accommodated within the integrated set of analysis. To achieve a design that is feasible, this approach requires satisfaction of an iterative process during each complete analysis cycle. In the case of combined weight and drag optimisation, for example, an optimiser passes the values of the design variables to the integrated structural and aerodynamic analysis. These analysis are iteratively evaluated until they converge to a consistent (i.e. multidisciplinary feasible) solution. The drag and constraint values are then used by the op-



timiser to modify the design variables in order to improve the design. This method is very computationally expensive and not so efficient. [38] [36]

## 2.2.2 All-at-Once (AAO) approaches

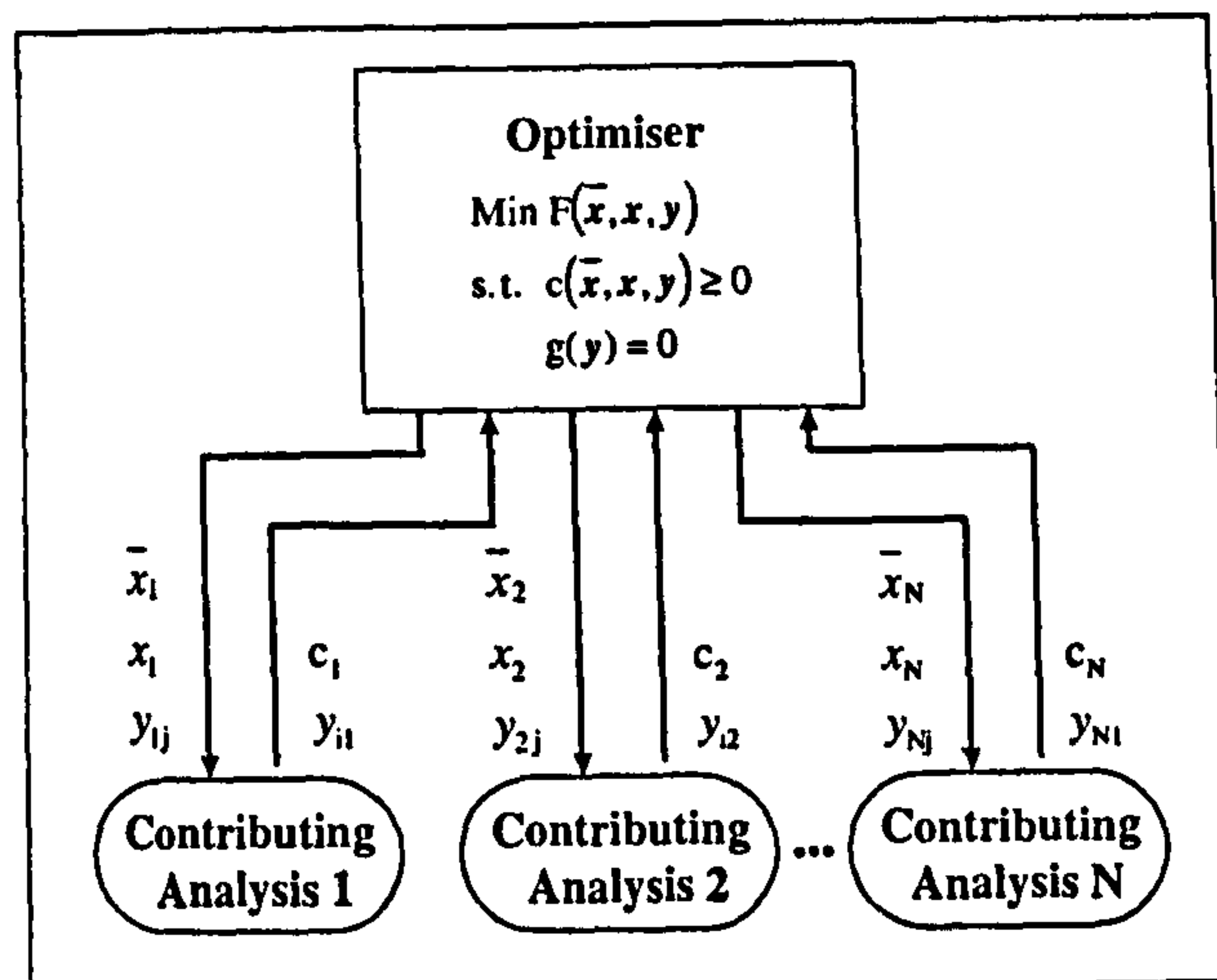


Figure 2.5: Simultaneous Analysis and Design (SAND) approach.

Figure 2.5 shows a schematic of the Simultaneous Analysis and Design (SAND) approach, which belongs to the All-at-Once classification. Using decomposition techniques, the traditional optimisation problem is restructured into several disciplinary analysis. A single optimiser is still used to evaluate the results of all analysis, but all of the analysis variables are optimisation variables and all of the analysis constraints are optimisation constraints. [34] To ensure compatibility between the different contributing analysis results, a set of compatibility constraints  $g_{ij}$  (equality constraints) and coupling variables  $y_{ij}$  are added. When satisfied, the equality constraints  $g_{ij}$  require that the value of a variable computed in analysis block  $j$  matches the value of the equivalent variable input into analysis block  $i$ . These coupling variables  $y_{ij}$  are included in the design variable vector along with both the disciplinary,  $\bar{x}_i$ , and interdisciplinary inputs,  $x_i$ . The main difference with the traditional approach is that the analysis are distributed and can be executed separately and in parallel. In the AAO approach, disciplinary or inter-disciplinary feasibility of the analysis is not required at each iteration until optimisation convergence is reached (i.e. where  $g_{ij} = 0$ ).

Benefits of the AAO approaches is that they often show a smoother design space than the standard approaches and are computationally more efficient. [37] The drawback is that for practical problems these approaches generally

involve a very large number of constraints. In addition, for large scale applications with thousands of design variables and constraints, the use of a single optimiser may lead to numerous problems, such as significant increase in the communication requirements between the contributing analysis and the optimiser. For tightly coupled problems this approach might need a large increase in the number of auxiliary design variables and constraints since all the design decisions are made by the single optimiser. Another negative aspect is that the analysis groups act only as function evaluators and don't participate in the decision process (i.e. do not perform sub-problem optimisation or constraint evaluation). [38] Balling [39] has shown that, based on the number of function evaluation calls, the AAO approach is more efficient than the other MDO approaches tested as fewer calls are needed. However, because the AAO has the largest optimisation problem that needs solving, the actual execution time is almost the same as for the other approaches.

### 2.2.3 Multi-Disciplinary Feasible (MDF) approaches

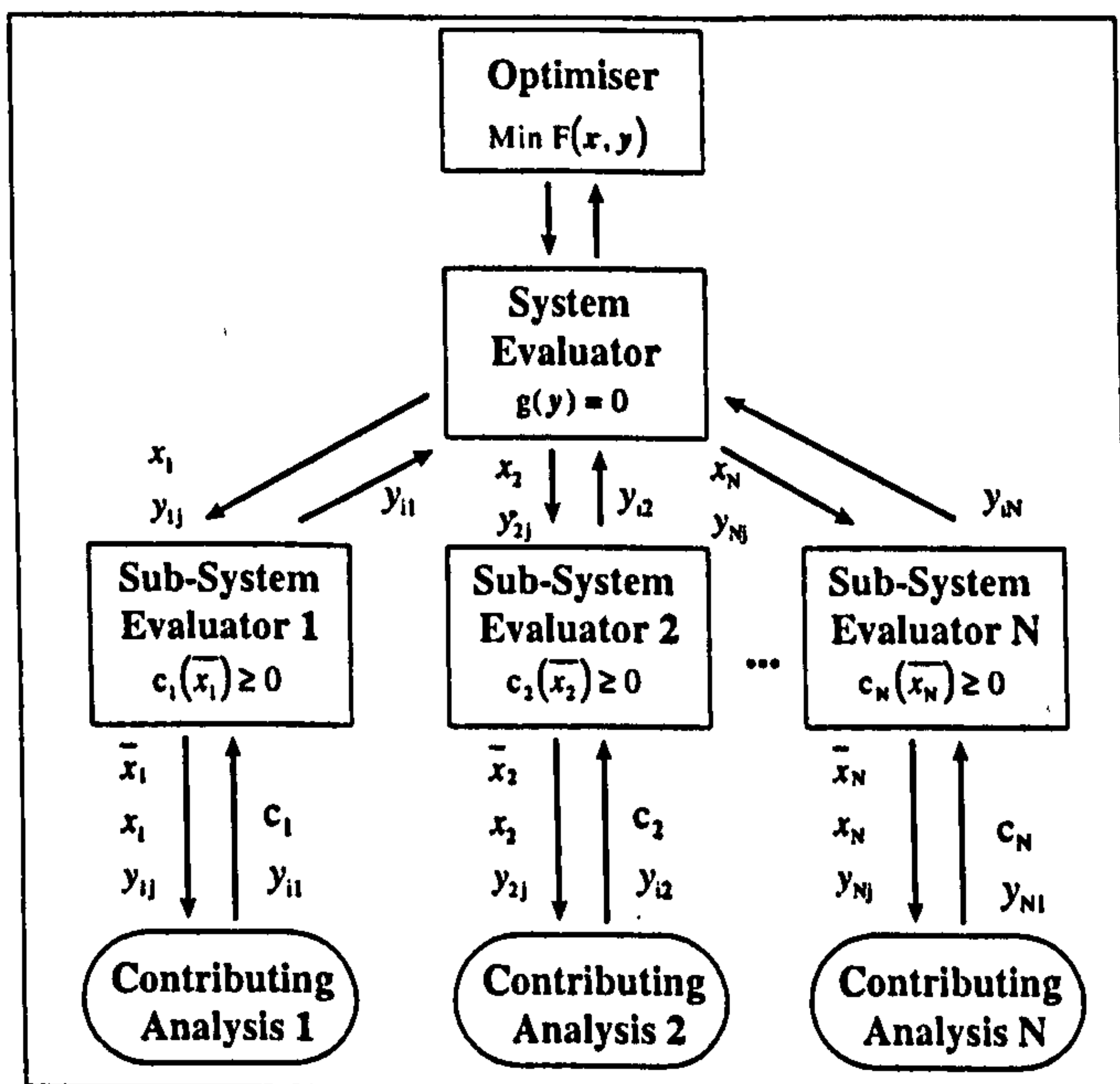


Figure 2.6: Multi-Disciplinary Feasible (MDF) approach.

The Multi-Disciplinary Feasible approaches were introduced in order to remove the main drawbacks of the All-At-Once approaches, i.e. the lack of involvement of the disciplines in the overall design process decisions and the high optimisation time. MDF methods are currently the most common way



of posing MDO problems and they are a complete opposite to the AAO approaches, as they allow for greater design authority among the distributed problems and are less sensitive to increase in problem size. [38] Both disciplinary feasibility (i.e. at individual discipline level) and multi-disciplinary feasibility (i.e. at global optimisation level) are required. As can be seen from figure 2.6 two additional levels have been added to the Simultaneous Analysis and Design formulation of figure 2.5 on page 21. Each discipline or contributing analysis is interacting with its specific 'sub-system evaluator'. These evaluators independently assess the feasibility of each sub-problem and this for a set of constraints which are specific for that sub-problem. The analysis and optimisation does not proceed until feasibility is achieved for all the sub-problems. Once this feasibility is achieved, a system evaluator is called which assesses the global or multi-disciplinary feasibility (or compatibility) of all the sub-problems. Only when this feasibility is achieved, the system level optimiser is activated. [25] This optimiser is used to ensure interdisciplinary compatibility at the overall solution and to minimise the objective function.

Because of the organisation of this approach, it is clear that it is more flexible and allows easier modification of sub-levels. For large problems, the structure of this approach makes the problem easier to comprehend. [34] With the introduction of the additional function evaluation levels, the actual optimisation problem has reduced in size and the optimisation itself has become less computational demanding. However, because of the added evaluation levels, the number of iterative loops has increased, which leads to performance drawbacks. Balling and Wilkinson [40] have shown that MDF approaches need significantly more function evaluation calls to reach convergence than the AAO approach and fail sometimes to converge.

#### **2.2.4 Individual Discipline Feasible (IDF) approaches**

Individual Discipline Feasible approaches can be classified as an in-between approach in comparison to the AAO and MDF formulations which are extremes. For AAO, no feasibility is enforced at each optimisation iteration, whereas for MDF complete multidisciplinary feasibility is required. The IDF approach maintains individual discipline feasibility while allowing the optimiser to drive the individual disciplines towards multi-disciplinary feasibility and optimality through control of the interdisciplinary data. [34] Three IDF approaches are discussed in this section.



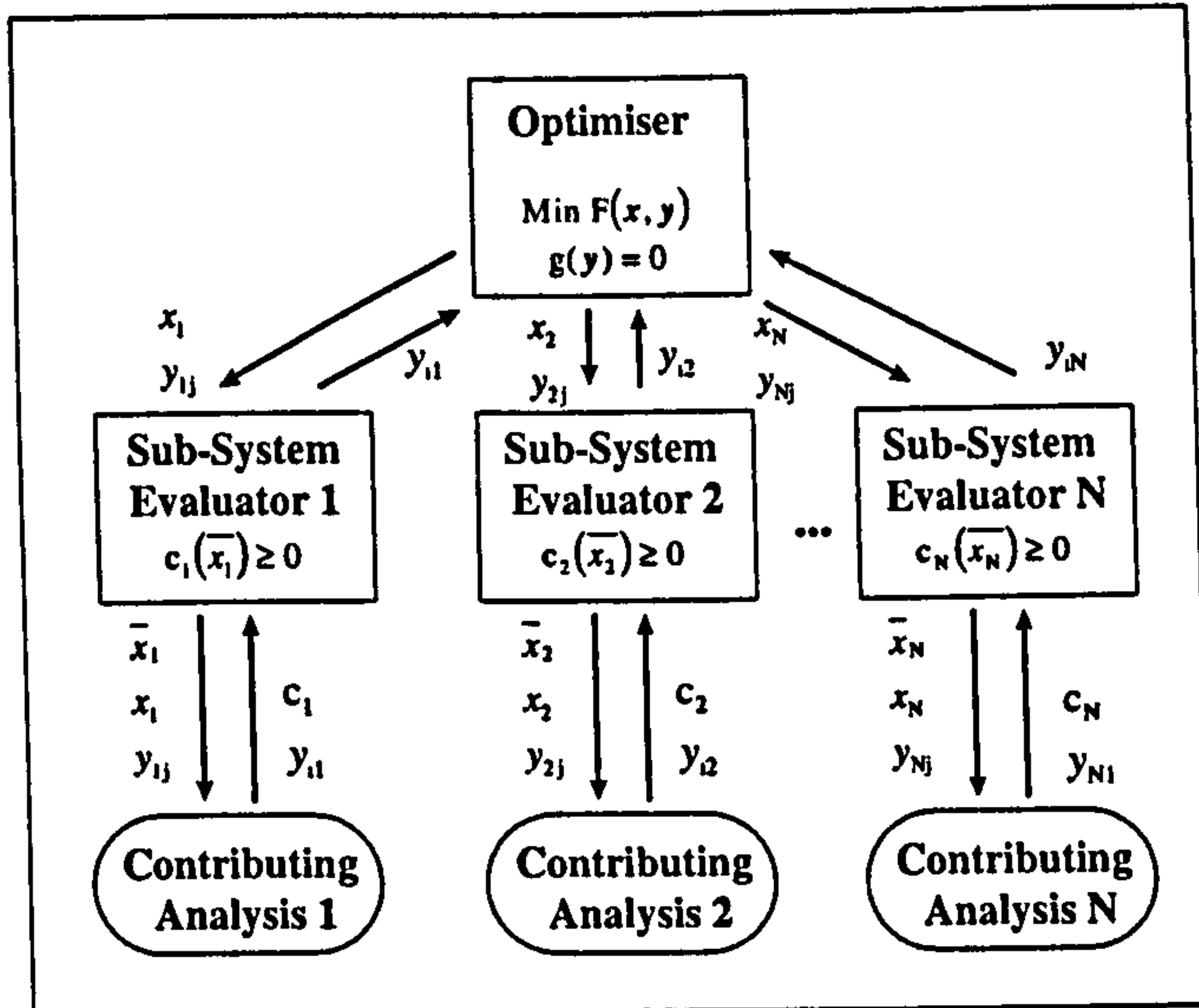


Figure 2.7: Nested Analysis and Design (NAND) approach.

#### 2.2.4.1 Nested Analysis and Design (NAND)

This approach is the last one which is based on a single level optimiser (as shown in figure 2.3 of page 19). Figure 2.7 gives an overview of the Nested Analysis and Design formulation. It is basically the same as the MDF approach, but the system evaluation level has been removed. Discipline feasibility is still required, before the system optimiser can assess the design and propose a new design change. As the system evaluation level is not present anymore, Balling and Wilkinson [40] have shown that fewer evaluation calls are required in comparison with MDF approaches.

#### 2.2.4.2 Concurrent Subspace Optimisation (CSSO)

Concurrent Subspace Optimisation is the first approach which uses multiple optimisers in the search of the multi-disciplinary optimum. As can be seen from figure 2.8 multiple sub-space optimisation problems are driven by a system level coordinator that controls the overall process. [38] CSSO has first been introduced by Sobieski [24]. The basic principle is that each subgroup tries to minimise the system objective, while trying to satisfy its own constraints and not preventing other groups from satisfying their constraints. Each group uses a linear approximation of the other group's constraints and a representation of the objective [36]. The system coordinator is used to evaluate the compatibility constraints in order to ensure compatibility of the sub-problem solutions. Drawbacks of the CSSO method are that the

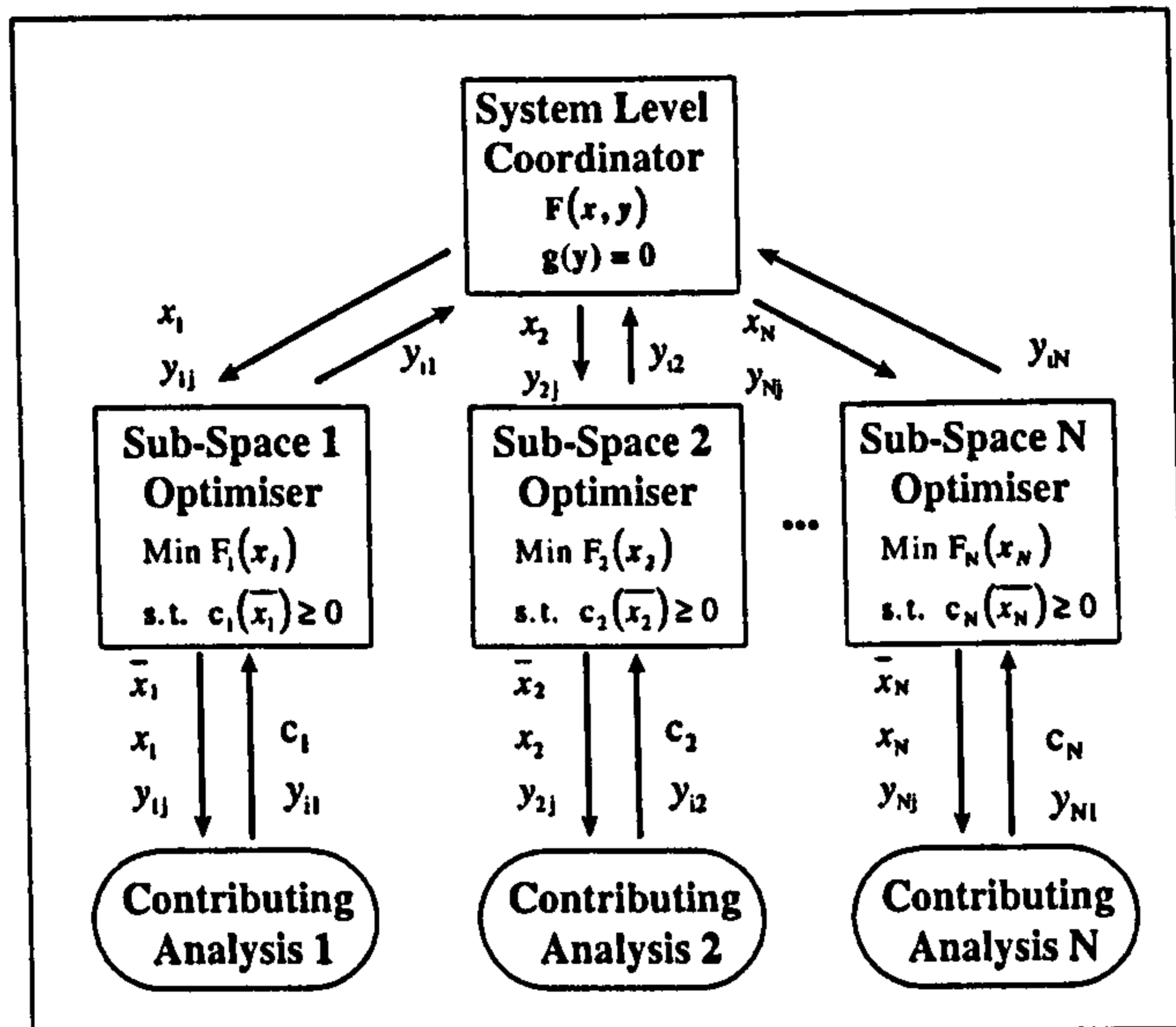


Figure 2.8: Concurrent Subspace Optimisation (CSSO) approach.

approach relies on linear approximations and the fact that sensitivities of the constraints of each sub-level need to be passed to each other group. For cases with strong sub-space coupling problems have been reported and test problems failed to converge [40]. However, other authors have reported significant analysis time reduction [41].

### 2.2.4.3 Collaborative Optimisation

The collaborative optimisation method is designed to have disciplinary autonomy, while achieving interdisciplinary compatibility. As can be seen in figure 2.9, the design problem is decomposed along analysis boundaries. Sub-space optimisers are integrated with each analysis block. Each sub-space optimisation group has control over its own set of local design variables and has to satisfy its own domain specific constraints. Explicit knowledge of the design variables and constraints from the other groups is not required. Each sub-group does not communicate directly with the other disciplines and hence maintains a degree of autonomy. The objective of each sub-optimisation problem is to obtain satisfactory values for the interdisciplinary design variables. A system level optimiser coordinates the design process and sets targets for the interdisciplinary design variables so the overall objective is minimised. The system optimiser also ensures that the constraints are satisfied at convergence. The advantage of Collaborative optimisation in comparison to CSSO is that linear approximations are avoided and that the



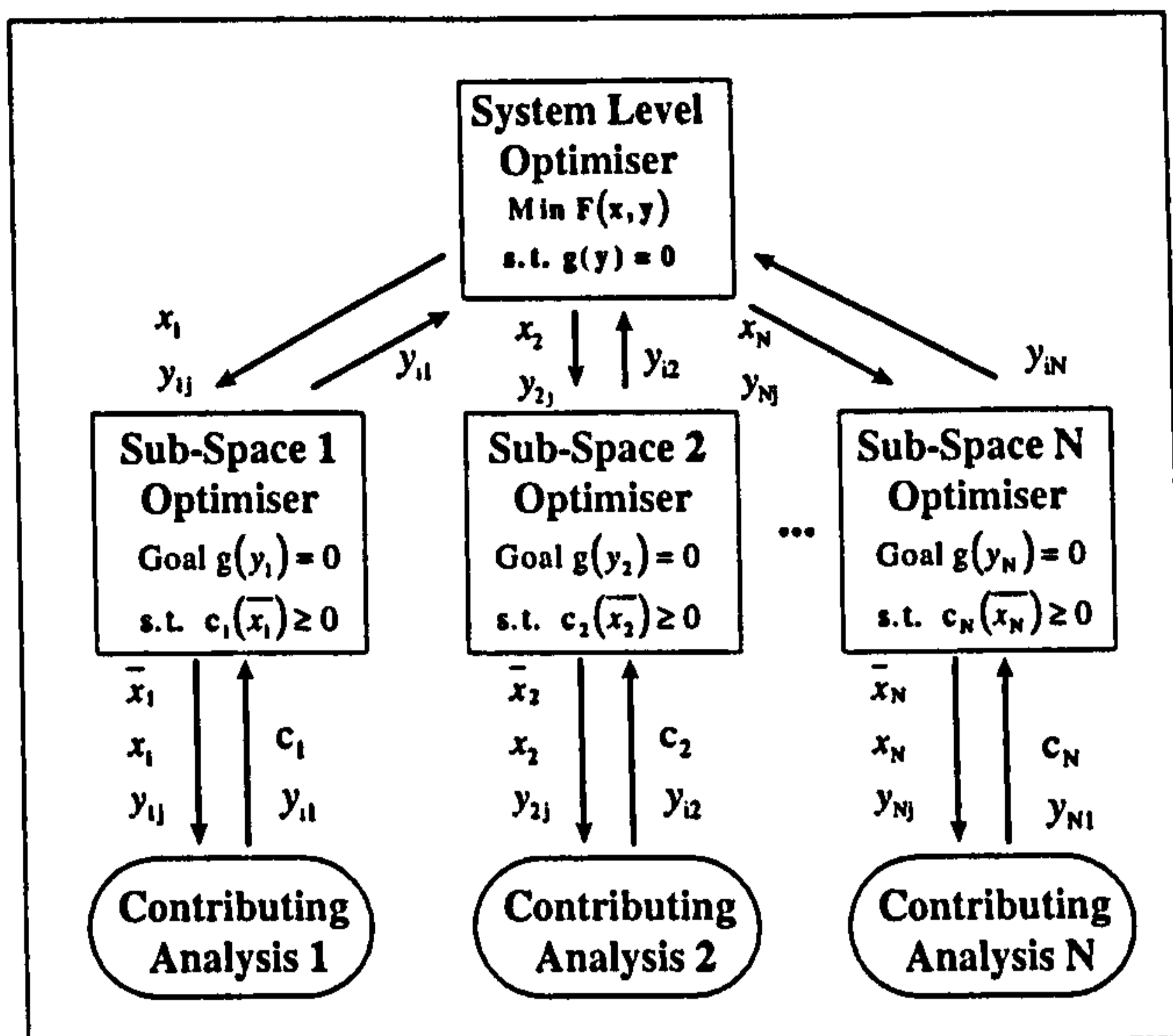


Figure 2.9: Collaborative optimisation approach.

number of required gradient information which needs to be passed between the sub-groups is reduced. [36] [38] Test problems carried out by Balling and Wilkinson have shown that the number of evaluation counts (iterations) is an order of magnitude greater than all the other MDO formulations and that excessive computation time is spent on attempting to satisfy the compatibility constraints. [40]

### 2.2.5 Conclusion MDO approaches

The Multi-Disciplinary Feasible (MDF) and Individual Discipline Feasible (IDF) approaches have the advantage of using, with moderate or no modification, existing single discipline analysis codes. An additional advantage of the IDF approaches compared to the MDF is that they avoid the cost of achieving full multi-disciplinary feasibility at each optimisation iteration, whereas this feasibility is required by MDF at each iteration.

A negative point is that IDF approaches requires the calculation of additional sensitivities which correspond to the variables (interdisciplinary variables) which are communicated between the disciplines. [34]

Although the Individual Discipline Feasible approaches use still some of the advantages of the Multi-Disciplinary Feasible approaches such as organisational advantages and flexibility for modifications, they are more computational expensive than the All-At-Once approaches. A significant increase in analysis calls and computation time is noticed, when employing multi-level

optimisation approaches (CSSO and Collaborative optimisation) instead of single level optimisation approaches. Of the IDF approaches, the NAND approach appears to be the most efficient. [40] [39]

Table 2.1 gives a comparison of predicted performance between AAO, IDF and MDF approaches. This has been based on test problems which were carried out by Cramer [34], Balling and Wilkinson [40], and Braun [38]

|                               | MDF   | IDF      | AAO    |
|-------------------------------|-------|----------|--------|
| Integration cost              | Low   | Low      | High   |
| Modification cost             | Low   | Medium   | High   |
| Number of optimisers          | One   | Multiple | One    |
| Size optimisation problem     | Small | Medium   | Large  |
| Sparsity optimisation problem | Dense | Moderate | Sparse |
| Number of design variables    | Low   | High     | High   |
| Computational cost            | High  | Medium   | Low    |
| Overall performance           | Slow  | Medium   | Fast   |
| Robustness                    | Low   | Medium   | High   |

Table 2.1: Comparison multi-disciplinary optimisation architectures.

The All-At-Once approach remains theoretically attractive because it will be the least expensive computationally. Unfortunately, AAO requires a higher degree of software integration than is likely to be achieved for large future realistic applications and AAO has large communication requirements. AAO also leaves the sub-disciplines with the task of function evaluation only. [34] [38]

By maintaining several of the advantages of the All-At-Once approach, while incorporating subspace optimisation to increase the role of the disciplinary expert, the Individual Discipline Feasible approaches have the most to offer to the engineering community when solving large-scale practical design problems. [38] Also, because of the large size and computational cost of solving MDO problems, the use of parallel and distributed computing will probably be required. The IDF approach is particularly well suited for integration in a distributed computing environment where analysis can be performed on different platforms which are suitable for each individual discipline. As with the IDF and MDF approaches the disciplinary analysis codes are kept intact, parallel processing capabilities which were developed for a discipline can be maintained.

No matter what approach is chosen, the efficient calculation of sensitivities will be critical for success. [34]



## 2.3 Examples of MDO applications

A considerable amount of MDO research has been carried out both on conceptual and preliminary design level. On the conceptual design level, a lot of MDO work has been performed in Europe and the USA. As this thesis focuses on MDO at preliminary design level, this section gives a review of preliminary design applications of MDO.

Since the 1980's, a lot of MDO research and MDO methodologies have been developed by J. Sobieski, J-F. Barthelemey et al. at NASA's Langley Research Centre. Lot of work at this MDO-branch has been carried out on the use of optimal sensitivities, global sensitivity evaluation and concurrent subspace optimisation. Because of the limited availability of structural optimisation codes, complete lack of high fidelity aerodynamic optimisation codes and low computing capabilities, the initial work concentrated on simple case studies and conceptual design issues. Over the past five years, however, the various necessary single-discipline design optimisation tools have appeared, also with a massive increase in computing capabilities. True multidisciplinary work became now possible and began to appear at the AIAA MDO conferences in Panama City Beach (1994) and Bellevue (1996). [30]

A survey of recent MDO developments by Sobieski [32] gave an overview of the past MDO research, focusing on the organisational challenges (i.e. the interaction of different disciplinary analysis codes with one another for the purpose of system analysis and optimisation). From this point of view previous research was divided into three categories:

- The first category classifies problems with two or three interacting disciplines, where a single analyst can acquire all the required expertise. Most of the papers in this category represent a single group of researchers or practitioners working with a single computer program, so that organisational challenges were minimised.
- The second category includes research where MDO of an entire system is carried out at the conceptual level by employing simple analysis tools. Because of the simplicity of the analysis tools it is possible to integrate the various disciplinary analysis in a single, usually modular computer program and avoid large computational burdens. Some of these codes are beginning to face some of the organisational challenges encountered when MDO is practised at a more advanced stage of the design process.
- The third category of MDO research includes work that focuses on the organisational and computational challenges, and on the development



of techniques that address these challenges. These include decomposition methods, global sensitivity techniques, development of tools to facilitate efficient organisation of modules and organisation of data transfer, and finally the use of approximation techniques to address the computational burden.

As mentioned by Sobieski [32] and Allwright [30], many papers have been presenting results of multi-discipline optimisation of aerospace applications using selected simulation tools. These studies have mainly focused on conceptual design or on combined structure -aeroelastic design at the preliminary design level.

MDO-studies on conceptual design level, of which some examples are mentioned here, have been performed on design of launch vehicles and new aircraft concepts such as the High Speed Civil Transport aircraft (HSCT).

Braun (NASA Langley) and Kroo (Stanford University) applied the collaborative optimisation architecture to the multidisciplinary design of a single-stage-to-orbit launch vehicle. Vehicle design, trajectory and cost issues were modelled. The design process used 95 design variables and 16 constraints. [37]

At the Georgia Institute of Technology, Schrage, Mavris and Marx have performed research on life cycle costing and developed a framework for the integrated product and process design (IPPD) of an aircraft system taking into account performance and economic perspectives. Results of this IPPD study were presented for wing concepts of the HSCT. Cost versus performance studies indicated that an aircraft with a hybrid wing structural concept, though more expensive to manufacture than some homogeneous concepts, can have lower direct operating costs due to a lower take-off gross weight and because it requires less mission fuel. [4] [14] [42]

Van der Velden (DASA Bremen) performed also multidisciplinary design optimisation of a supersonic civil transport aircraft. Simultaneous optimisation as function of Mach number, payload and range was performed using more than twenty variables representing the aircraft and its flight-profile. It was identified that by applying the developed methodology, significant reduction of the number of follow-on detailed design cycles can be obtained, especially for non-conventional designs. [43]

Many examples of MDO studies at preliminary design level, involving two or three disciplines and mainly concentrating on structure-aeroelastic design can be found. High fidelity structural optimisation tools (NASTRAN, ASTROS, STARS, etc.) were used and combined with low fidelity aerodynamic models to include the effect of structural deformation due to aerodynamic forces. These tools allow to obtain an optimum structure of minimum weight



while simultaneously satisfying multiple constraints related to aeroelastic effects, strength, maximum displacements, failure criteria and stability requirements. They also allow the design of composite structures, subjected to the previous mentioned requirements and buckling criteria. Studies have been performed on small and large transport aircraft wings and the High Speed Civil Transport aircraft (HSCT).

Striz and Lee from the University of Oklahoma have used the multidisciplinary optimisation tool ASTROS to perform optimisation of weight and roll effectiveness of a small transport wing, subject to stress, material and aeroelastic constraints. A metal and various composite wings were investigated. [44]

Borland (Boeing) and Barthelemy (NASA Langley) have demonstrated how high-level analysis codes, in this case structural analysis using the finite element method and aerodynamic analysis using computational fluid dynamics analysis, can be coupled with a mathematical optimiser in a 'simulated' distributed computing environment. The study used a simplified structural and aerodynamic model of a typical twin engine transport aircraft wing. The research demonstrated that aerodynamic and structural shape optimisation can be performed simultaneously to optimise a chosen objective function such as minimum weight, minimum drag or maximised range. [1]

Multidisciplinary optimisation of a large transport aircraft wing was performed by the McDonnell Douglas Corporation. The wing was optimised for minimum weight subject to simultaneous stress and flutter constraints. An in-house developed aeroelastic design optimisation tool (ADOP) was used for the analysis. This study was performed to validate the capabilities of ADOP as an advanced structural design tool. [45]

Another study at McDonnell Douglas looked at a practical application of MDO to a HSCT aircraft. The disciplines that Baker and Giesing included were structures and aerodynamics. By changing the wing twist, the MDO process successfully decreased the takeoff gross weight of a baseline aircraft which was already optimised for aerodynamics and structures separately. Their work discusses the classical trade-off between aerodynamics and structures. It was mentioned that the most challenging issue in MDO is the calculation of sensitivities of sub-discipline optima to global design variables. Finite difference sensitivities were used for the current study and new procedure to estimate optimum sensitivities was presented. [15]

Multilevel optimisation for the preliminary wing design of the HSCT was also performed by Rohl from the Georgia Institute of Technology. The wing design was decomposed into three levels, with a top level generating preliminary weights, mission requirement and performance information, an FE-based structural optimisation on the second level and detailed buckling opti-

misation of individual skin cover panels on a third level. ASTROS was used for the structural optimisation and the wing was sized subject to strength, buckling and aeroelastic constraints. The work presented in this paper describes mainly the integration of the panel buckling procedure into the structural optimisation system and initial results. [46]

Detailed design of composite panels taking into account aeroelastic effects has been studied by Bartholomew from UK Defence Evaluation and Research Agency (DERA) and Wellen from DASA Airbus. The authors describe how detailed design considerations can be taken into account into the overall aircraft structural design, by using a multilevel approach. The research was part of a collaborative programme conducted within GARTEUR (Group for Aeronautic Research and Technology in Europe). For the analysis, DERA's structural optimiser STARS was used and interfaced with DASA's panel design program. A three spar wing was used as benchmark problem by all GARTEUR partners. [6]

A nice overview of current multidisciplinary issues is given by Venkayya and Tischer from Wright-Patterson Laboratory and Bharatram from Wright State University. They have conducted a wing design optimisation study of a five spar aluminium wing to determine the effect of the configuration variable sweep on the minimum weight, subjected to strength and flutter constraints. ASTROS was used for this study. The paper presents the results and methodology but also gives an interesting introduction on the history of MDO and its future directions. [47]

The review of the papers published on MDO applications between 1990 and 1995 revealed two facts:

- Most papers and research originate from the USA.
- Most papers present the results of a multi-discipline optimisation, but there is very little critical appraisal of the validity of the solution process, the value of the resultant design in contributing to the top-level product design and development process [30], and the practical and organisational aspects of the investigated problems.

Research which specifically addresses the development and validation of MDO methodologies in aircraft design has emerged since 1996 from activities such as a European MDO project, an MDO Test Suit developed by NASA Langley Research Center and GARTEUR.

- **European MDO Project**

Form 1996 until the end of 1997, a two year European Union funded



project (BE95-2056) attempted to create a distributed Multi-disciplinary Design and Optimisation system. It explored the application of MDO methodologies to the design of large-scale civil airliner components. The MDO system incorporated weight, drag and manufacturing costs and included other factors such as flutter speed considerations and stability as constraints. The project involved a cooperation between most of the European airframe manufacturers. Partners in the project were British Aerospace (Project Leader), Aerospatiale, NLR, DASA, CASA, SAAB, Dassault, ALENIA, Aermacchi, HAI, DERA, ONERA, the University of Delft and the Structures and Materials group of Cranfield College of Aeronautics. As the work presented in this thesis was mainly part of this project, its scope and methodology will be discussed in detail in the next chapter.

- **NASA Langley MDO Test Suite**

Since 1995, the AIAA MDO Technical Committee and NASA Langley have been active in creating a library of test problems to act as a show-case for the methods of MDO. The MDO-Branch at NASA Langley has introduced the MDO Test Suite on the web [48]. This Test Suite has the following purposes:

- Provide the MDO researchers with a set of problems for the development of new optimisation methodologies.
- Establish a 'standard' set of problems for comparing relative advantages of MDO approaches and formulations.
- Provide the applied mathematics community with MDO problems representative of various engineering areas.

The problems are divided into classes by complexity from small demonstration problems, over problems with engineering content to complex problems which include several engineering disciplines. Each problem is equipped with description and source codes for stand-alone subroutines for function and constraint evaluation. If available, the source for MDO solutions and derivatives is provided as well. [30] [48] [49]

- **GARTEUR**

The GARTEUR Structures and Materials GoR has supported research activities on structural optimisation since 1990. A lot of work was performed on codes for the buckling design of composite panels. Between the partners, different codes were compared and their integration into the overall structural optimisation process was investigated. Based on

the promising results from the previous research, a GARTEUR activity started in 1995, which looked at the multidisciplinary optimisation of structures and aeroelastics. A turboprop commuter aircraft was used as baseline for the study. Since this project, and under impulse of the European MDO project, GARTEUR has taken measures to encourage interdisciplinary activities. Currently, there is a proposal for work on the modelling and simulation of a flexible aircraft. Although this is strongly focused on flight mechanics issues as the aim is to investigate the effect of low frequency flexible aircraft modes on primary flight control systems, it would be possible to study the impact of active control systems on airframe design. For this activity, MDO could be used, providing that the appropriate analysis techniques for control/structures interaction are developed. [30]

## 2.4 Conclusions

This chapter has given an overview of different MDO aspects. Many of the issues presented will now be discussed from a more practical point of view in the next chapter, where a detailed outline is given of the European MDO project.



## Chapter 3

# The European MDO project; A driving scenario for MDO implementation.

The previous chapter has given a general outline of MDO methods. This chapter will now explain in more detail how a recent European project has attempted to put the previously mentioned MDO methodologies into practice on a realistic large scale design problem: the wing of a large civil airliner. An overview of the applied methodology, problem definition, specific assumptions and developed tools will be given. It will also outline how the work, presented in this thesis heavily interacted with other project modules and how it forms an integrated part of this MDO project.

### 3.1 Project motivation

Under the impulse of current findings and trends in the field of MDO and concurrent engineering, a large European initiative started, financially supported by the European Commission. For the first time a joint consortium of 14 European partners from the European aerospace industry, research institutions and two universities investigated the potential for new and effective multi-discipline design methodologies. The MDO project had the overall objective of strengthening competitiveness of the European aircraft industry and of providing a European design capability for future aircraft. In support of this development, the project partners defined a set of objectives, which stated that the project must:

- Develop and demonstrate the viability of MDO methodologies and validate them for a simplified but realistic aircraft preliminary design task.

- Investigate and demonstrate a common implementation architecture for MDO.
- Develop standards for data exchange and the solution process in order to facilitate and demonstrate industrial exploitation.
- Investigate the issues relating to the control of data exchange, implementation of computationally cheap algorithms for the calculation of derivatives, the control of numerically intensive calculations, the overall MDO process control and the end user-visualisation.
- Perform multi-disciplinary analysis and optimisation combining aerodynamics, structures and manufacturing disciplines. This to include the derivation of aerodynamic and structural sensitivities, the study of design variable influence, integration of aeroelasticity, effects of the control system on the loads, structural modelling and manufacturing constraints. The MDO project should also allow a comparison of the different CFD and structural optimisation tools which are being used by all the partners involved.

## 3.2 Practical implications

The specific nature of the MDO project, i.e. being a European Commission funded initiative, has raised some specific practical implications which were never encountered by previous MDO research. For the European Commission, the motivation for funding research projects has several reasons:

- To initiate and improve cooperation between the specific industry and research institutes of all the EU member states and to facilitate collaboration on commercial and research projects.
- To disseminate information and promote technology transfer between the different member states, especially from the more advanced countries to those who are still developing or modernising their industry.
- To focus European industry on current challenging research issues and to ensure that Europe can play a leading role.
- To make the European industry more competitive on the international market.

Because of its high technological and competitive character, the European aerospace industry is one of the main areas targeted by the European



Commission. Airbus is an example to what successful collaboration between different aerospace companies can lead to. It is from this vision that the MDO project found European support and this also clarifies its specific distributed character. From the start, the project was faced with huge organisational challenges:

- Large variety of commercial and in-house tools used by the different partners.
- Large variety in computing infrastructure
- Different company cultures and approaches to design and problem solving.
- Overcome language barriers.
- Disciplinary 'jargon': different disciplines would have different terminology for the same parameters or phenomena. (e.g. wing thickness (aerodynamics) = wing depth (structures))

[18]

Excellent project management and the successful development of adequate software for data handling, exchange of results and generation of analysis models (see section 3.5) has overcome many of these barriers and was the foundation for the successful completion of the project.

Because of the distributed nature of the project, the MDO methodology applied was also distributed in nature. A multi-level approach was adopted. It was also decided to allow each partner to use their own analysis tools. This complicated the generation of the project data and exchange of results, but permitted the investigation and comparison of different in-house and commercial analysis codes.

As mentioned in the previous chapter, multi-discipline design is only possible when human factors associated with teamwork; cross-discipline interaction and cross-discipline understanding are addressed. Hence, from the start of the project, high importance was put on using teams with members from different disciplines and organising multi-discipline meetings and workshops.

### 3.3 Project organisation

During a two year period the project followed a divergence-convergence strategy (see figure 3.1). It evolved from an initial, simplified multidisciplinary analysis of a reference MDO problem, through a refined MDO process which



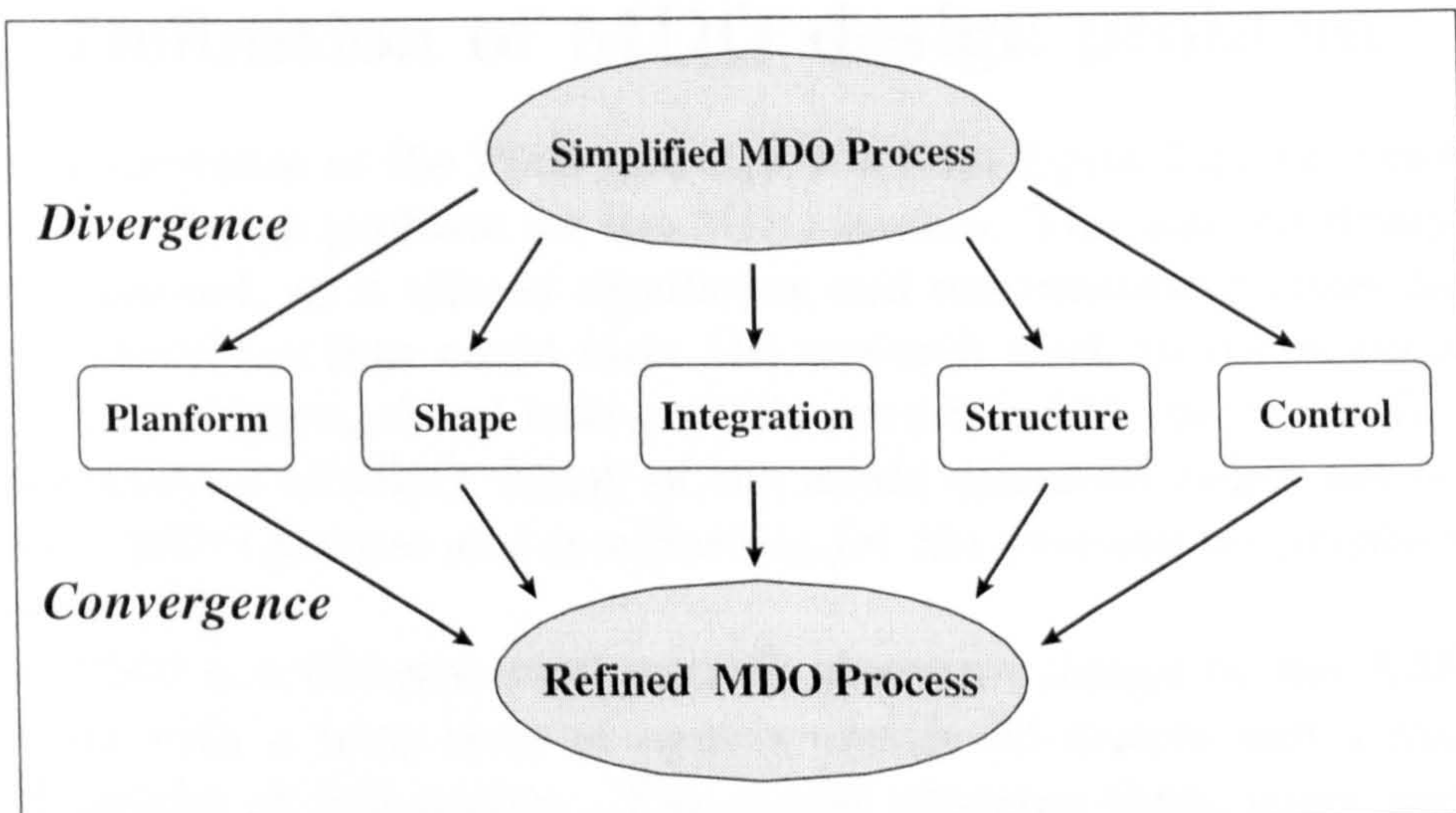


Figure 3.1: Divergence-convergence strategy adopted by the MDO project.

integrated the conclusions and recommendations of the various parallel research themes. As can be seen from figure 3.1 the project was divided in three main phases:

- Baseline phase. Implementation and validation of a simplified MDO process. It included the establishment of common tools which allowed for parameterised aircraft design and generation of high-fidelity analysis models. After this was established, the analysis models and tools were validated by investigating a number of primary design variables and deriving multidiscipline performance sensitivities for them.
- Divergence and investigation phase. Groups of partners worked together in small teams to investigate and address various scientific and computational challenges in the areas of planform optimisation, surface shape optimisation, structural design and optimisation (including manufacturing cost), control system optimisation and MDO software frameworks.
- Convergence and integration phase. This phase looked at the integration of results and recommendations developed in the divergence phase and on the formulation of general conclusions and recommendations on how MDO should be implemented.

[30]



### 3.4 Definition of MDO design problem

The wing structure of the P500 (see figure 3.2 and figure 3.3) was selected as the common design problem for the MDO studies. This aircraft design problem was selected, as it offered significant and representative cross-discipline design interactions that could steer the research work to the relevant aeronautical and computational issues which are critical to the successful industrial exploitation of MDO. Many of the issues discussed below are obtained from the '*MDO process and specification for the primary sensitivities study*' document. [50]

The P500 is a 650-seat civil aircraft precursor design to the A3XX (see figure 3.4) with a wing span of approximately 80 meters and a maximum take-off weight of 550 tonnes. The model contains three spars and has a crank at 35% of the overall wing span. Table 3.1 shows the main aircraft characteristics.

|                     |                   |
|---------------------|-------------------|
| Wing Area           | 725m <sup>2</sup> |
| Wing Aspect-Ratio   | 8.2               |
| Wing Span           | 77.1m             |
| Wing Sweep          | 33°               |
| Wing root chord     | 16.5m             |
| Wing crank chord    | 10.075m           |
| Wing tip chord      | 3.647m            |
| Fuselage length     | 70.4m             |
| Max take-off weight | 551T              |
| Engines             | 4@310kN           |
| Passenger capacity  | 650 mixed class   |
| Mach No cruise      | 0.85              |

Table 3.1: Reference aircraft specifications

Analysis models were developed for both an aluminium wing and an aluminium inboard and composite outboard wing. Although the project mainly concentrated on the wing, information on the overall aircraft configuration was needed to perform adequate structural, aerodynamic, aeroelastic and control studies. More information on the overall aircraft configuration can be found in '*MDO Process and Specification for the Primary Sensitivity Study*'. The next paragraphs will discuss the structural and aerodynamic models which were developed and used.

From a preliminary aircraft operational study, carried out by British

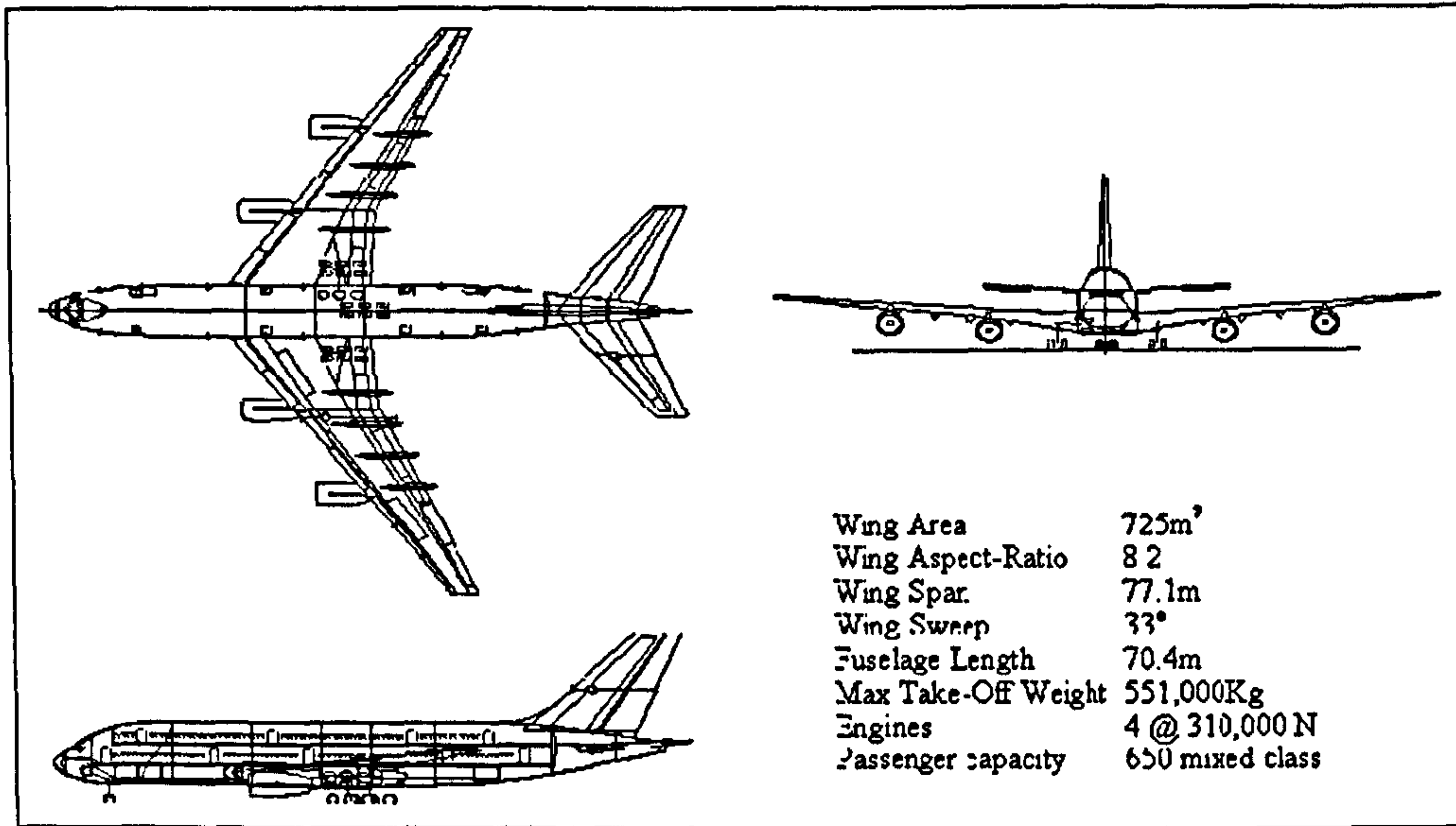


Figure 3.2: P500 reference aircraft.

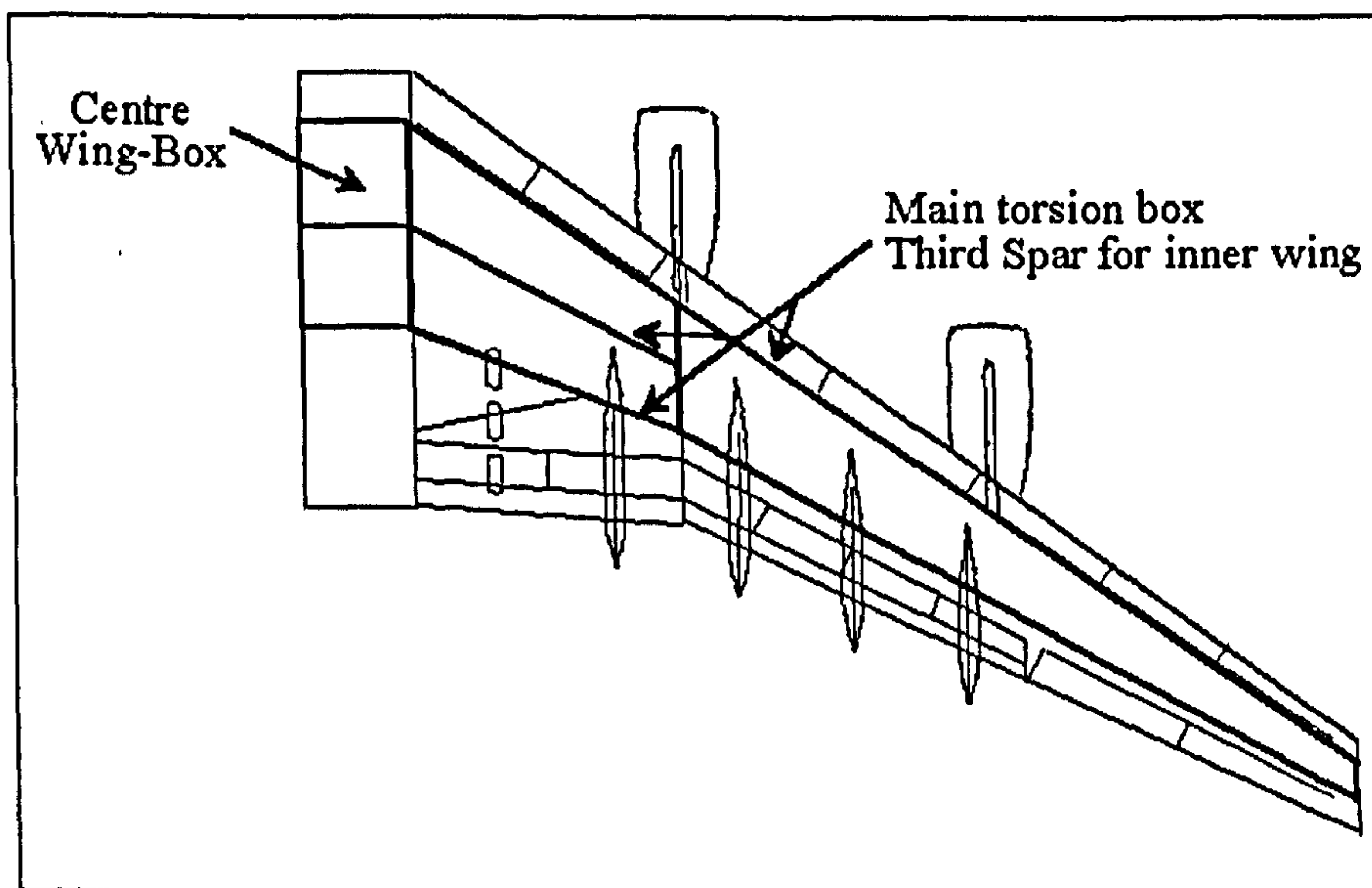


Figure 3.3: Wing layout, P500 aircraft.





Figure 3.4: A3XX, Airbus's future large airliner.

Aerospace, seven design operations were extracted. They were selected because of their significance from an integrity or efficiency (Direct Operating Cost) perspective:

- **Heavy cruise.** Important operation from economics point of view. It relates to the start of the design long range cruise when the aircraft is full of fuel and passengers and/or payload.
- **Economic cruise.** This operation relates to the case where the airline is flying a shorter route and when the aircraft is filled with passengers for about 75% of its capacity. This case is significant for aircraft range and fuel burn economics.
- **Light cruise.** This flight condition is important for flutter (dynamic instability of the aircraft) and is characteristic for the last part of a flight, i.e. the end of cruise and start of descent when the aircraft has minimum fuel in the wing to damp dynamic instabilities.
- **Diversion.** A case which occurs when the aircraft is diverted to an alternative airfield. Important from fuel burn economics and aircraft range points of view.



- **Pull-up.** This case is important from the point of static strength. It relates to the 2.5g pull-up manoeuvre which is required in the case of an aborted landing at maximum take-off weight and with the flaps retracted. This induces a high combined upward bend and shear load in the inboard wing structure.
- **Push-down.** An operation important for static strength and relates to the -1g traffic avoidance manoeuvre at maximum weight which induces a high combined downward bend and shear load in the inboard wing structure.
- **roll-case** This operation occurs for a pilot induced traffic avoidance manoeuvre with high payload weight, fuel weight and a high equivalent airspeed which induces a high combined bending and torque load in the wing. A flight case important from static strength and roll-control point of view

### 3.4.1 Structural definition

The structural definition of the wing involves the main torsion box structure, pylon attachment fittings and ailerons. The wing tip, fixed leading edge with slat support structure and the support structure for the main landing gear are ignored. The structure also has a number of cut-out features, such as stringer cut-outs, rib lightening holes and skin manholes. All these cut-out features were not taken into account. The main torsion box contains three spars of which the centre spar extends as far as the inboard engine which is located at the crank (see figure 3.3). The ribs are oriented perpendicular to the rear spar outboard of the crank and inboard they stay parallel as far as possible with the ribs outboard of the crank. For the reference wing model the rib pitch is 600 mm and the stringer pitch is 195 mm. This results in a total of 69 ribs and 34 stringers for the top and for the bottom skin. An overview of the rib layout can be seen in figure 3.5

#### 3.4.1.1 Finite element model of wing

Figure 3.6 shows the current finite element model of the wing box. Engine pylons and pylon attachments are included. The structural idealisation models skins, stringers, spars, ribs, control surfaces and fittings only. A description of how these structural components have been modelled follows below. The elements which were used for the finite element model were: rod, beam and quad4 membrane and shear elements. From reference [50] and [51, 52] we obtain the following definitions:



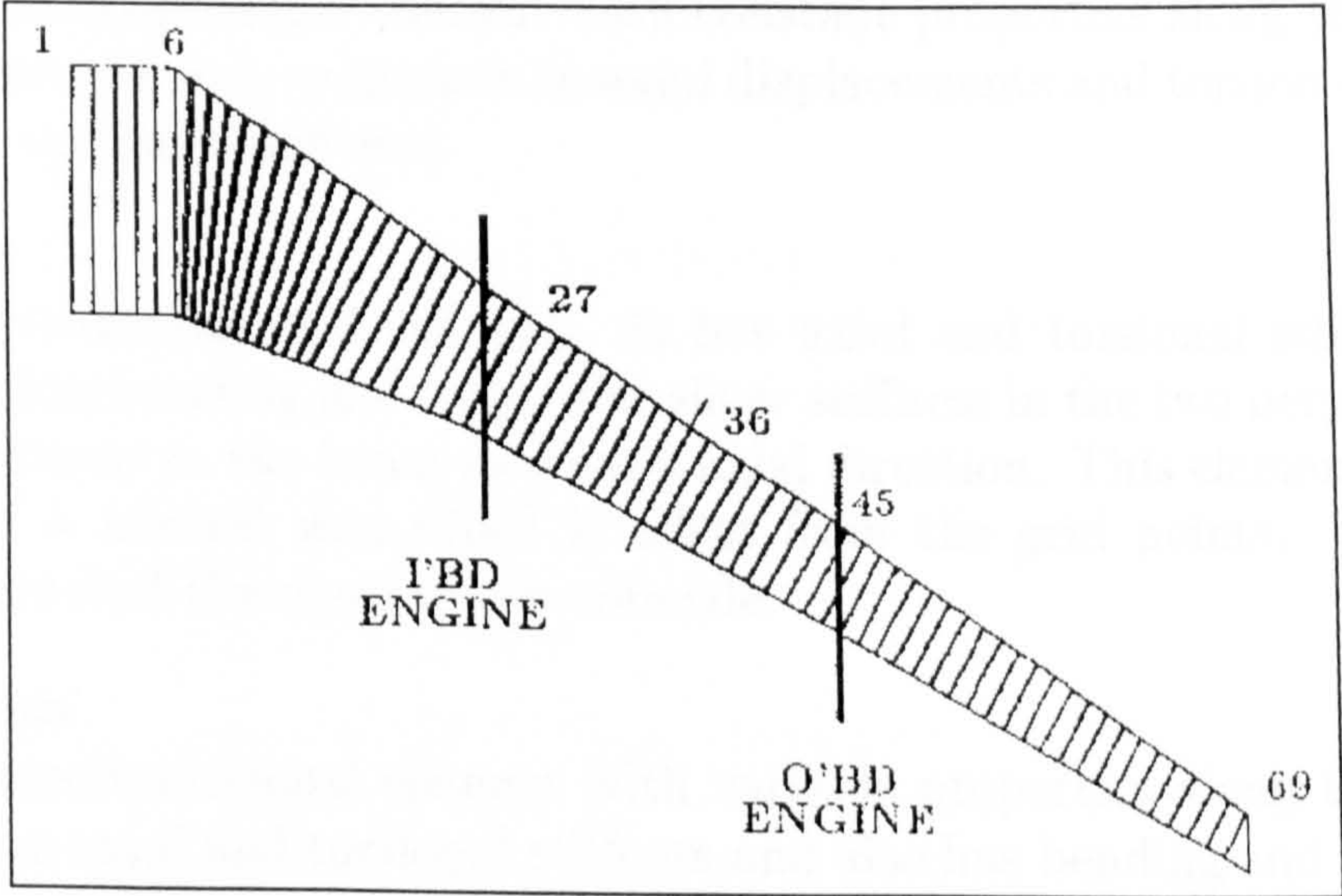


Figure 3.5: Rib layout.

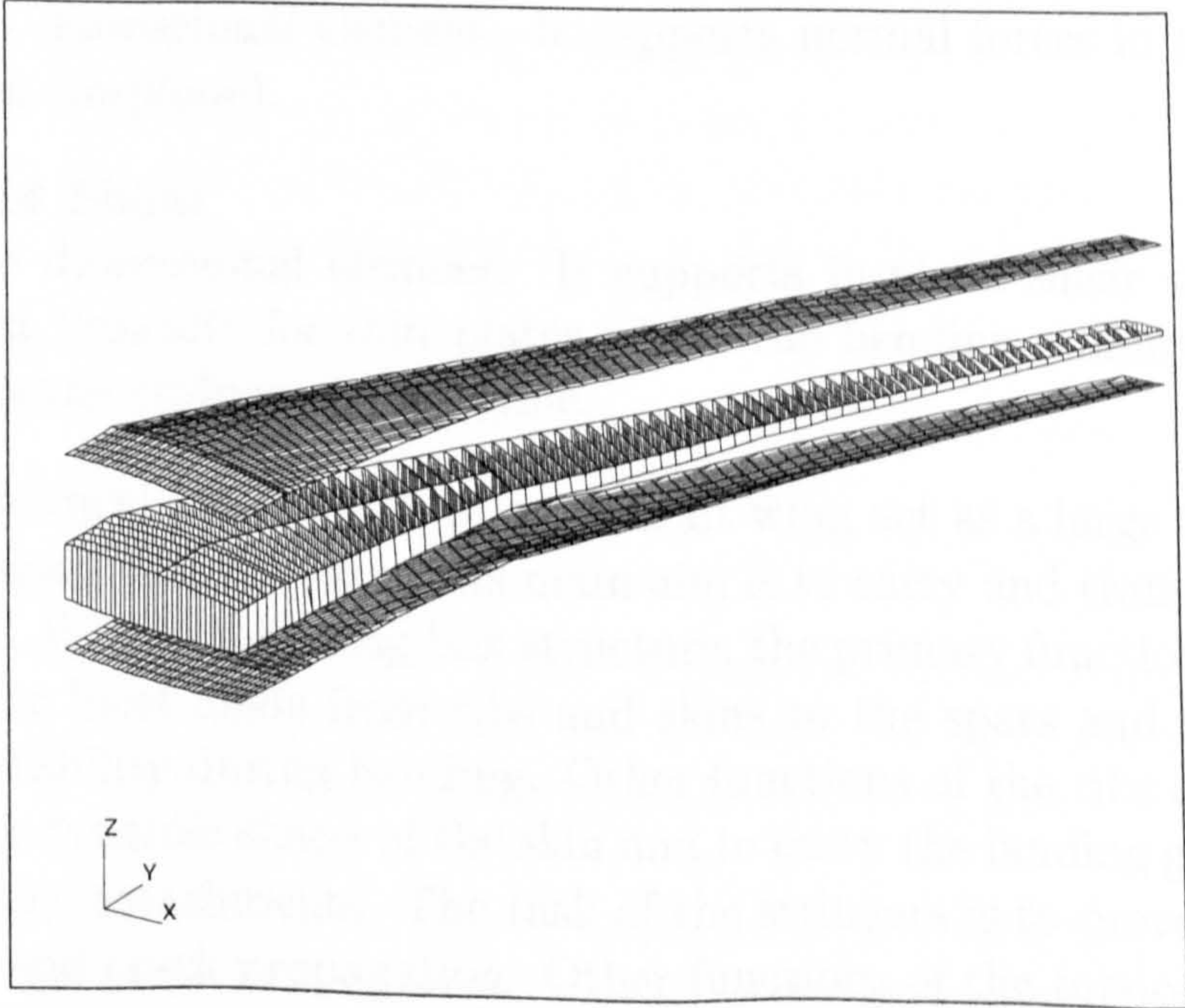


Figure 3.6: Wing finite element model.

- **Rod**  
A one-dimensional element with constant properties along the length. It provides only resistance to axial displacements and torsion (i.e. axial and torsional stiffness).
- **Bar**  
A one-dimensional element. It has axial and torsional stiffness and also has bending and transverse shear stiffness in the two perpendicular directions to the beam element's axial direction. This element can also have a neutral axis which is offset from the grid points. The shear centre and the neutral axis coincide.
- **Beam**  
A one-dimensional element with variable properties along the length. It has axial and torsional stiffness and also has bending and transverse shear stiffness in the two perpendicular directions to the beam element's axial direction. The principal axis of inertia need not to coincide with the local element axes and similarly the neutral axis and shear axis do not need to coincide. (e.g.: open sections) This element can also have a neutral axis which is offset from the grid points.
- **Quad4 Membrane**  
A two dimensional element. It supports normal forces in x and y direction (in-plane).
- **Quad4 Shear**  
A two dimensional element. It supports in-plane shear stresses and is used typically for thin plates where the bending stiffness and axial membrane stiffness is negligible.

The structural components of an aircraft wing act as a large torsion box, formed by spars and the skins. Its main aim is to carry and transfer bending and torsion. Within this wing box structure, the primary function of the ribs is to transfer local loads from ribs and skins to the spars and to maintain structural stability during bending. Other functions of the ribs are to maintain the aerodynamic shape of the skin and to carry the landing gear, engines and secondary attachments. The task of the stringers is to prevent buckling of the skin and crack propagation. Other functions of the torsion box are to create an efficient aerodynamic surface and to provide fuel storage.

Based on the structural function of the wing box components and looking at the definition of the finite elements types, each structural element was modelled in the following way:



- **Skin**  
The upper and lower skins are modelled as membranes with the thicknesses as the design variables.
- **Stringer**  
The stringers are modelled as pure tension-compression elements, without torsional rigidity. The stringer cross-sections are taken as the design variables.
- **Spars** Spar caps are idealised as pure tension-compression elements without torsional rigidity. Its cross section will be the design variable for the optimisation process.  
  
The spar web is made of shear elements with no axial stiffness. To be able to deal with bending, vertical stiffeners which are tension-compression elements and which have no torsional rigidity are added. The thicknesses are taken as the design variables.
- **Ribs**  
Rib caps are modelled as tension-compression elements, without torsional rigidity. Two types of ribs are present in the model: lightly and heavily loaded ribs. The heavily loaded ribs, such as the pylon ribs (ribs 27 and 45, see figure 3.5) and the rib at the wing-fuselage intersection (rib 6, see figure 3.5) are integrally machined. All the other ribs are assumed lightly loaded and are formed. The rib webs of the formed ribs are represented by shear elements without axial stiffness. The webs of the machined ribs are modelled by membrane elements.
- **Control surfaces**  
Two ailerons have been modelled for the control optimisation. The aileron was modelled using 8 bar elements.
- **Fittings**  
The fittings are elements which introduce and distribute the applied loads to the structure. They are modelled by rigid elements, RBE3, which do not add additional stiffness to the structure.

Table 3.2 summarises the structural members of the wing box and the type of elements by which they are represented in the finite element model.

As mentioned earlier, an outboard wing made of composite material was also studied. For the reference wing, it was decided to start the outboard composite wing between the two engines, at rib 36. The properties of the composite material are shown in table 3.3.

| Structural member        | Element type                          |
|--------------------------|---------------------------------------|
| Skin panels              | QUAD4, TRIA3<br>(work as membrane)    |
| Stringers                | ROD                                   |
| Spar webs                | SHEAR, ROD                            |
| Spar caps                | ROD                                   |
| Formed Rib webs          | QUAD4, TRIA3<br>(work as shear panel) |
| Machined Rib webs        | QUAD4, TRIA3<br>(work as membrane)    |
| Rib caps                 | ROD                                   |
| Spars and rib stiffeners | ROD                                   |
| Control surfaces         | BEAM                                  |
| Fittings                 | RBE3                                  |

Table 3.2: Overview structural idealisation.

| Laminate property  | CFC wing element |           |           |           |
|--------------------|------------------|-----------|-----------|-----------|
|                    | Skin Panels      | Stringers | Spar webs | Spar caps |
| $E_{xx}(N/mm^2)$   | 71140            | 87450     | 32220     | 87450     |
| $E_{yy}(N/mm^2)$   | 32500            | 29660     | 32220     | 29660     |
| $G_{xy}(N/mm^2)$   | 17550            | 13250     | 27840     | 13250     |
| $\mu_{xy}(N/mm^2)$ | 0.44             | 0.37      | 0.55      | 0.37      |

Table 3.3: Laminate properties of CFC wing elements.

To design composite components properly one has to take into account the number of plies, the stacking sequence and the ply orientation as additional variables. This would complicate very much the optimisation process, which was already a large problem. In addition, not all commercial optimisation codes are able to perform such a detailed optimisation. Hence it was decided to use composites with fixed percentages of laminates for each laminate orientation, based on company experience. Table 3.4 shows the percentages of laminates considered for each element. For this the 'black metal' approach, orthotropic material cards (MAT8) were used to model the composite skin material properties. Only shells can be modelled as orthotropic material. For the stringers however, as they were modelled by rods which only have axial stiffness the isotropic material card was maintained and has values for composite material. Its elasticity modules E has been replaced by



| Laminate orientation | CFC wing element |           |           |           |
|----------------------|------------------|-----------|-----------|-----------|
|                      | Skin Panels      | Stringers | Spar webs | Spar caps |
| 0°                   | 45%              | 60%       | 10%       | 60%       |
| ±45°                 | 45%              | 30%       | 80%       | 30%       |
| 90°                  | 10%              | 10%       | 10%       | 10%       |

Table 3.4: Percentage of laminates for CFC wing elements.

the composite elasticity modulus  $E_{xx}$ . This can be justified as the actual stringer lay-up has for 60% 0° plies, which gives it mainly axial stiffness. Spars and ribs are still made of aluminium.

### 3.4.1.2 Loads

Loading information was provided by the project partners working on the aeroelastic simulation of the reference aircraft. The aeroelastic analysis requires that the operational in-flight mass distribution of the wing is modelled accurately. These loads are described in greater detail in section 3.4.3.

Static loads for two principal structural design cases (+2.5g and -1g) are specified as bending, shear and torque distributions acting on the aeroelastic reference axis. The aeroelastic reference axis has been defined as a straight line through the wing box and aligned at a constant fraction of the outboard wing box chord (see figure 3.7). Structural reference nodes have been chosen on this axis. Using RBE3 elements the loads are then distributed from the aeroelastic reference axis onto those structural reference nodes. Figure 3.8 shows how these loads are mapped.

These loads for the pull-up and push-down case have been differentiated to obtain forces and moments which can be applied to discrete points along the reference axis. A set of wing loads was provided by BAe. Figures 3.9 and 3.10 show the forces and moments for the pull-up and push-down cases.

During the project however, some unrealistic thickness distributions were observed. After structural optimisation it was found, for example, that the skin and spar thicknesses near the root rib were at their minimum gauges. This could be explained as the wing is behaving as a one-dimensional beam where during optimisation all the bending material is moved to places which are most effective from structural efficiency point of view: places near the greatest depth of the wing box. However, other load cases and strength requirements, would in reality prevent the occurrence of minimum gauges near the wing root. Hence other load cases were suggested which could make the optimised design more realistic:

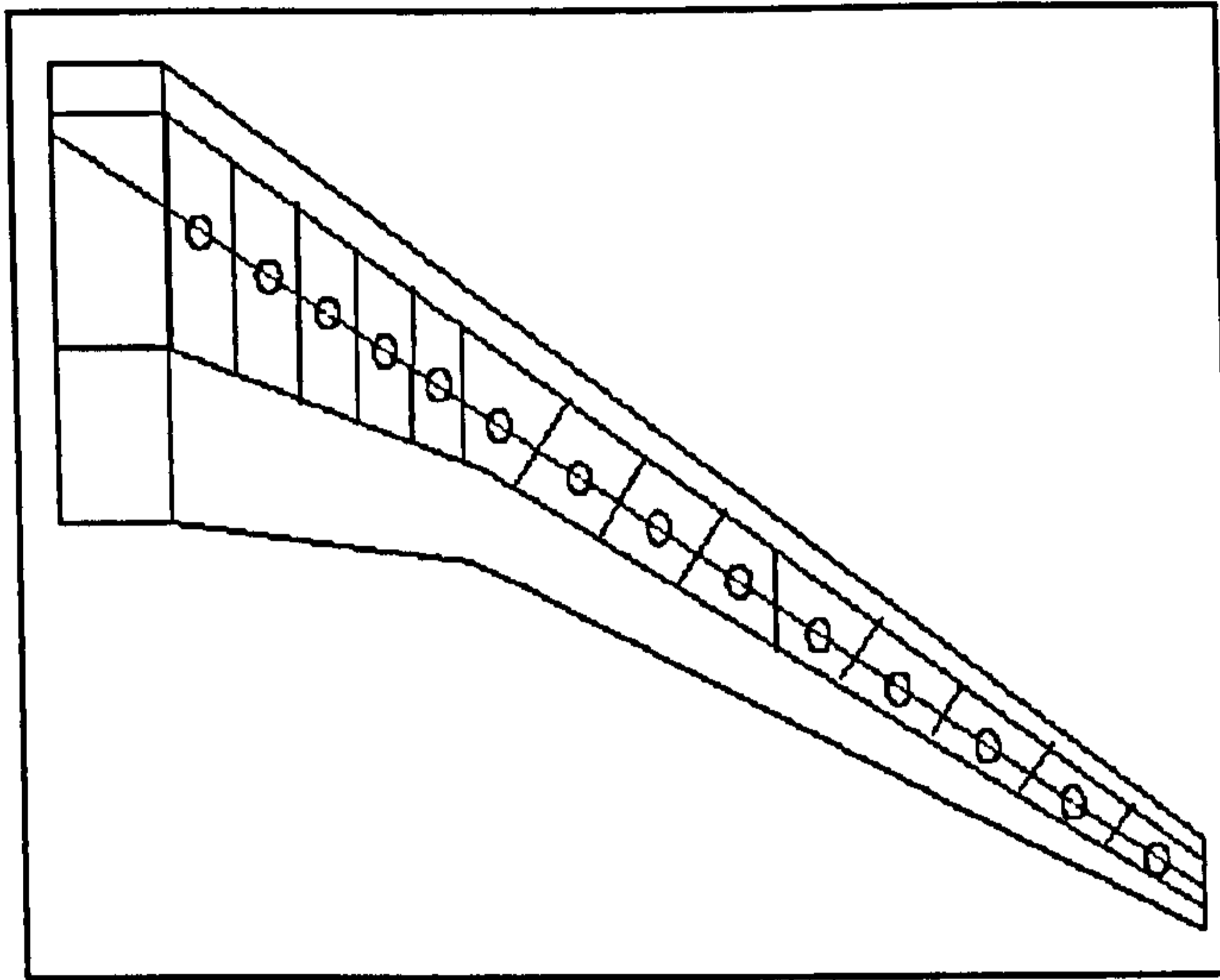


Figure 3.7: Aerodynamic reference axis

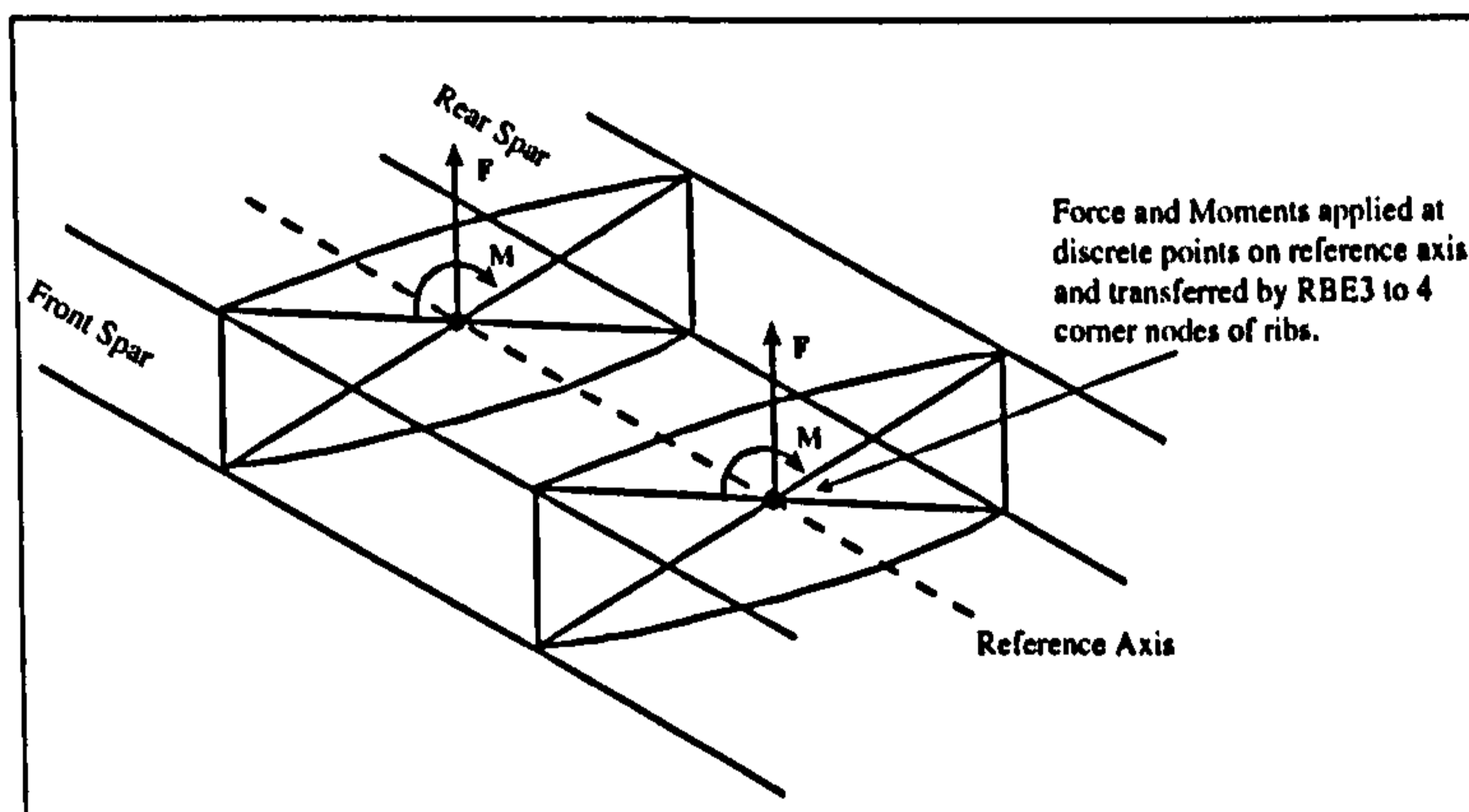


Figure 3.8: Application and distribution of loads to the finite element model.



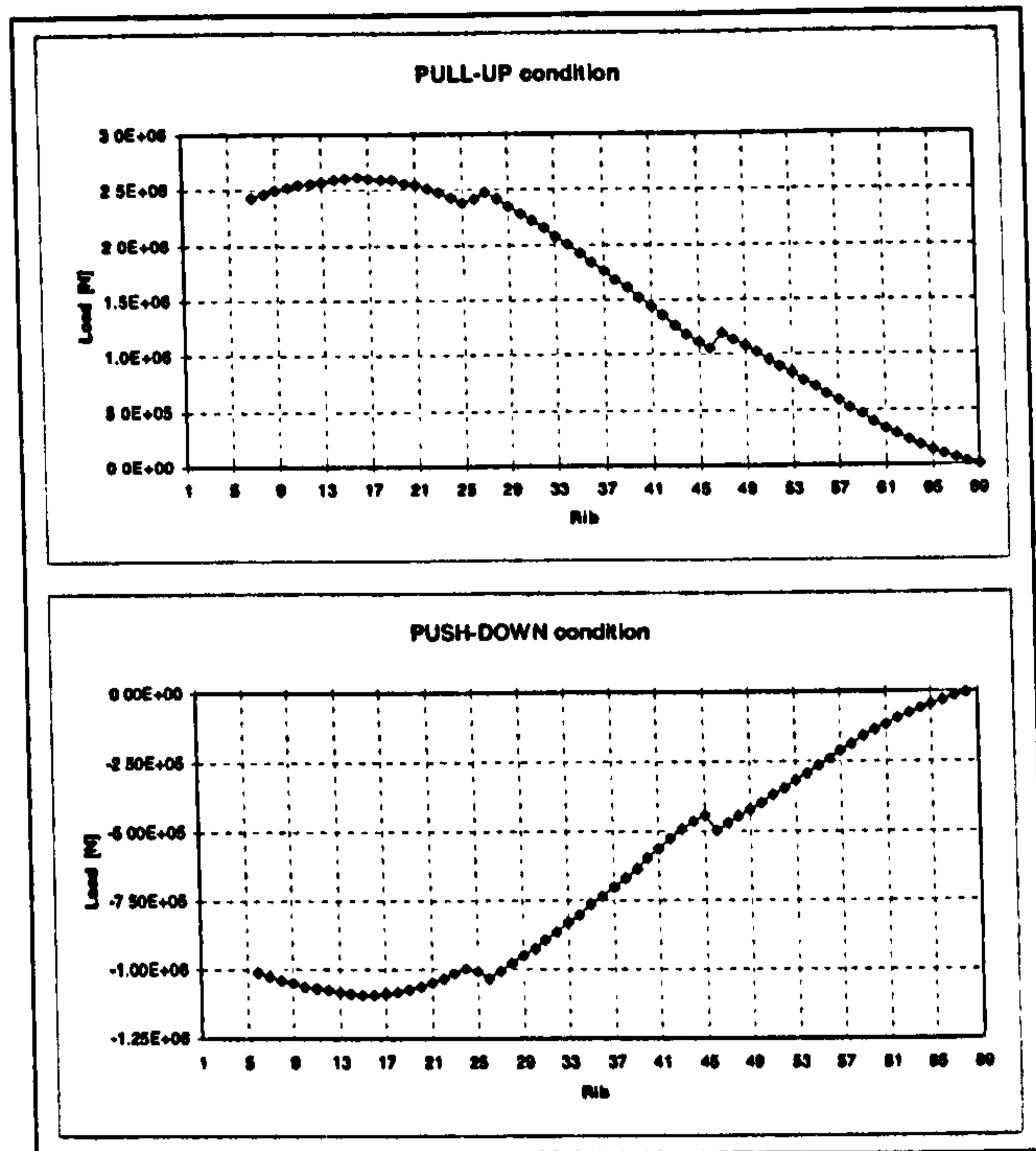


Figure 3.9: Gross shear force on reference axis, pull-up and push-down case.

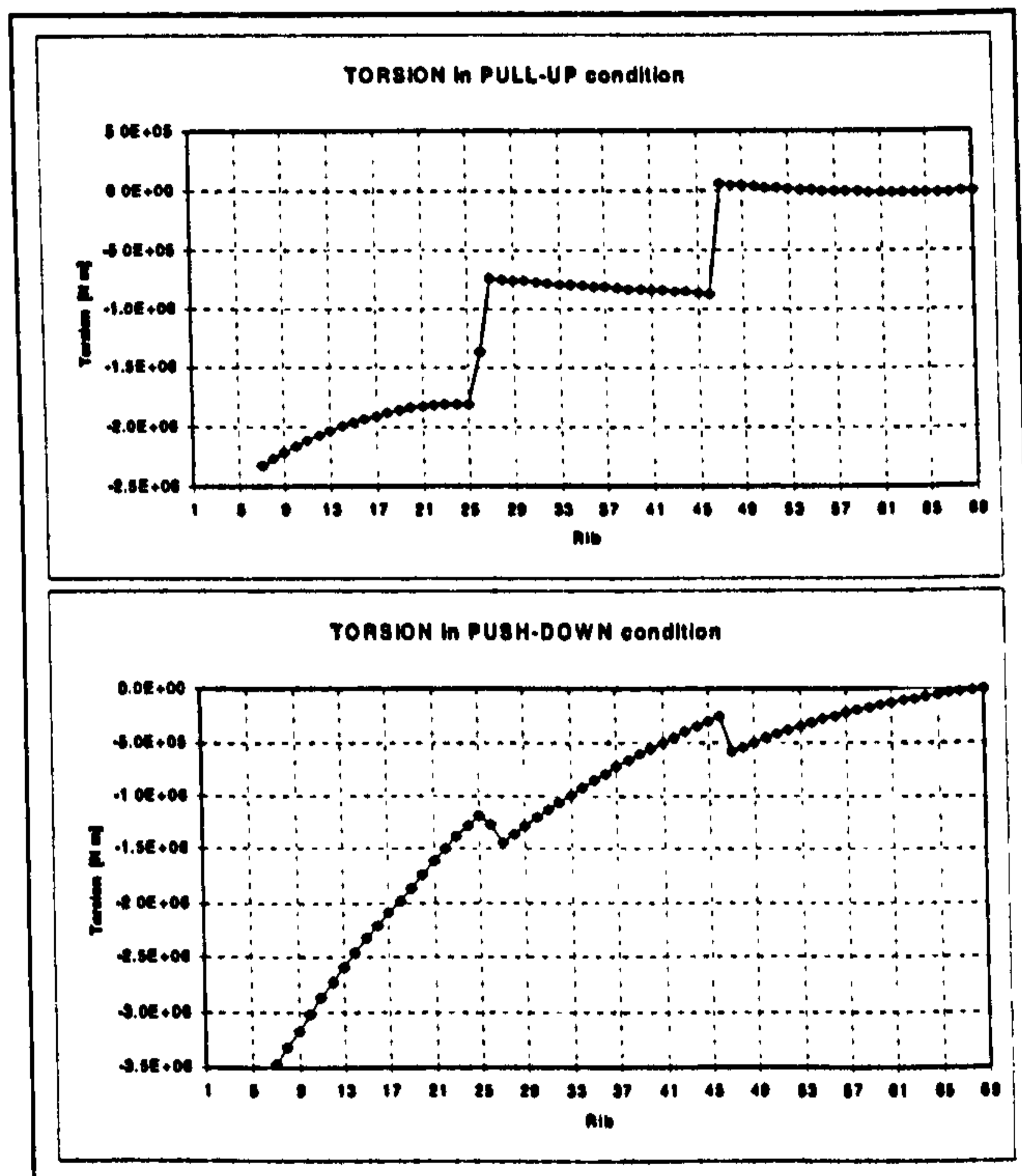


Figure 3.10: Moments on reference axis, pull-up and push-down case.

- Load cases at different airspeed, with same load factors.
- Load cases at same airspeed, with different load factors.
- Load cases with extended flaps.
- A crash condition with associated fuel loads acting on the front spar web.

The first two cases would have an effect on the pitching moment and possibly the overall torsion moment, resulting in greater shear loads and hence an increase in the thicknesses. However, the first option was not found practical as increasing the airspeed would move the aircraft outside the speed envelope, whilst reducing the speed would reduce the maximum obtainable load factor as the wing lift coefficient is on the stall boundary. Moving down along the stall boundary [53] would not move the centre of pressure and would result in reduced loads and stresses. The second option has been investigated, but not much difference in the optimum design was noticed. The last two options were not investigated due to the required work involved to adapt the models. [54] Landing cases have also not been included but should be considered in order to obtain a more realistic design.

### 3.4.1.3 Boundary conditions

The wing is constrained at the root (rib 1) and at the wing-fuselage intersection (rib 6, see figure 3.5). The purpose of the wing joint is to only transfer shear loads into the fuselage, while other loads are taken by the joint of the two wings. Basically, the aircraft's wings 'hang' onto the fuselage. This behaviour is modelled by imposing zero displacement at rib 1 in the spanwise direction (i.e.  $y$ ) and at rib 6 in the chordwise direction (i.e  $x$  and  $z$ ) with vertical displacement ( $z$ ) fixed for the spar nodes only. Rotations are also prevented around roll ( $xx$ ) and yaw ( $zz$ ) axis at rib 1. An overview of these constraints is seen in figure 3.11. In this figure the constraint numbers 1,2,3 represent the translation vectors, while 4,5,6 represent the three rotations vectors.

### 3.4.1.4 Finite Element codes used

A variety of in-house and commercial codes has been used by the project partners for structural analysis and optimisation. Tools included:

- MSC/NASTRAN



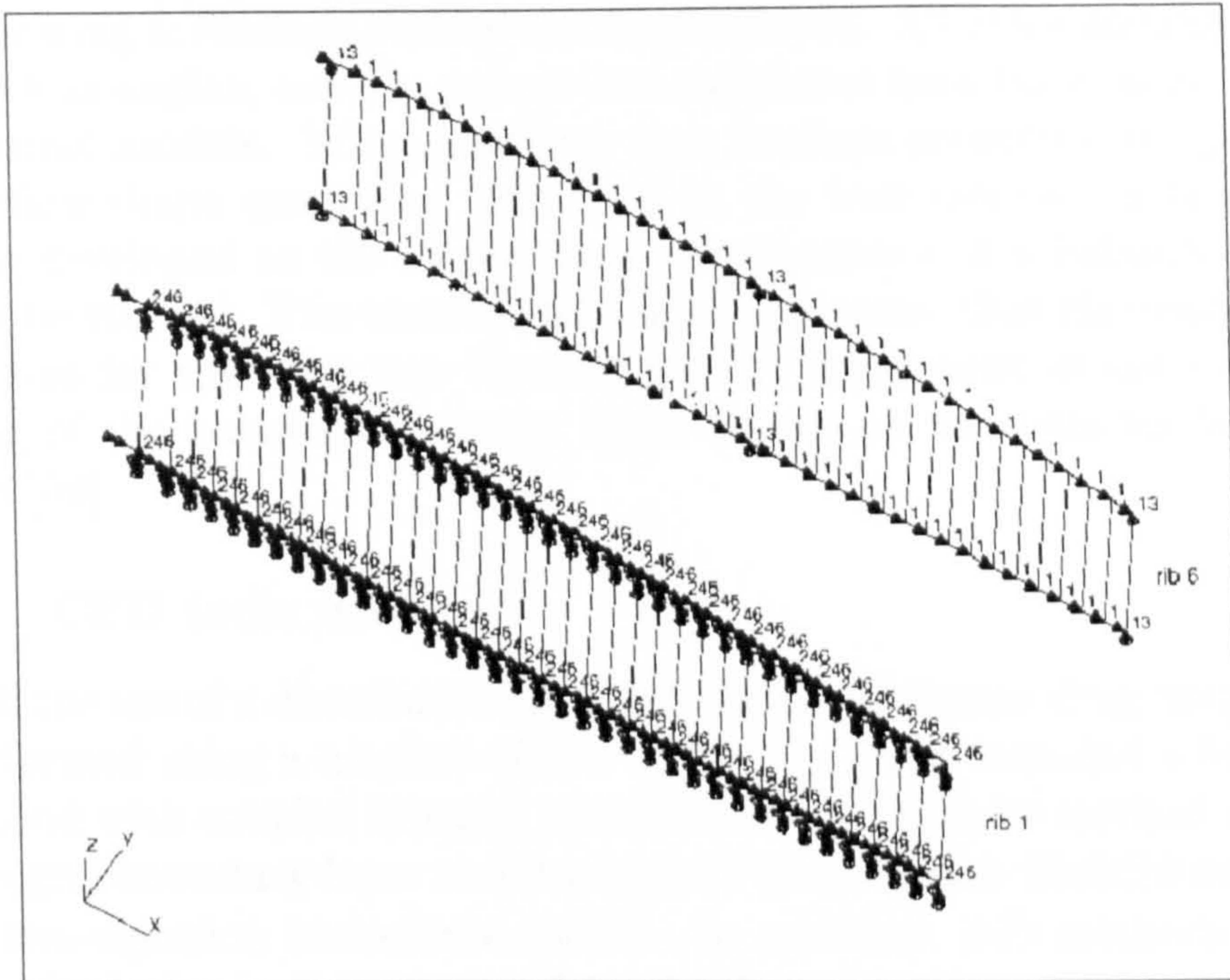


Figure 3.11: Applied boundary conditions.

- ELFINI (from Dassault)
- OPTSYS (from SAAB)
- STARS (from DERA)
- B2000 (from NLR)
- ALACA (from CASA)

The program employed by CASA is different to the other optimisers and analysis codes as it is based on optimising at the detailed stressing level, so that components are optimised on an individual basis. Results of the structural optimisation will be discussed in the next chapter.

### 3.4.2 Aerodynamic definition

The aircraft configuration for the aerodynamic simulation consists of the wing including a rounded wing tip and the fuselage, including the main wing/fuselage joint fairing. Geometric models for wing-alone, wing & centre



section or wing & fuselage section can be generated. All other aircraft components such as engine, nacelle, pylons and tailplanes have been omitted in the aerodynamic models. Wing planform and fuselage geometry are generated by a surface shape generator, described in the tool section. A trim model has been developed as the aerodynamic performance of a balanced aircraft needs to be studied. The trim model has to make sure that the aerodynamic calculations for the wing can be used for the assessment of the overall lift and drag of the balanced aircraft. More details on the trim model can be found in [50].

#### 3.4.2.1 CFD tools used

Through the use of a distributed approach the aerodynamic drag optimisation was performed using a number of 3-D methods. These included a full potential method with coupled integral boundary layer; an Euler method with coupled integral boundary layer and Averaged Navier-Stokes Multiblock method with a two-equation turbulence model. In addition, 2-D methods adapted for 3D calculations were also used to check the results and investigate the potential improvements in calculation speed. [55] [56] [57]

#### 3.4.3 Aeroelastic modelling issues

To ensure that no dynamic instability of the aircraft occurs in any part of the flight envelope, the aeroelastic behaviour had to be studied. Aeroelastic studies first identify the dynamic mode shapes of the aircraft and their degree of damping or excitation at any flight condition. Once this has been identified, the aircraft design must be constrained to ensure that all critical modes are damped. In addition to the static finite element model, other modelling aspects have to be integrated in order to be able to perform flutter analysis:

- Engine model.

Engines and engine pylons have been modelled by a stick model, combined with GENEL elements. The MSC/NASTRAN GENEL elements are used to represent the mass and stiffness of the engine. It is not an element in the same sense as for example a CQUAD4 or CBAR element. The GENEL element is used in cases when one wants to include a substructure in a model which is difficult to model using standard elements. The input data for the GENEL element can be derived from a hand calculation, another computer model or actual test data. However, when using GENEL elements, extreme care must be taken with respect to the fact that it contains rigid body modes, the stiffness matrix which has to be positive definite, the fact that no large differences



in matrix terms are allowed and the fact that the stiffness terms added should not be significantly different to the terms already present in the main stiffness matrix of the structure.

- Fuselage, fin and tail model.

The mass and stiffness of fuselage, fin and tail have been added. It has been modelled by beam elements and concentrated mass elements.

- Wing non-structural mass model.

Non-structural mass (e.g. hydraulic systems), fuel mass and structural mass (e.g. flap tracks) which is not represented in the finite element model, such as the weight of the engines, is added to the wing finite element model. These masses are represented by inertia matrices, given at points along the wing aeroelastic reference axis. To transfer this additional mass information from the aeroelastic reference axis on the finite element model RBE3 elements are used in the same way as is described in figure 3.8 on page 47.

As the total fuel mass and the fuel mass distribution are flight case dependent, the inertia matrices are different for each flight case. The project software, described in [50] is able to adapt this distribution for each case.

- Aerodynamic panel model.

By using a lifting surface model for unsteady aerodynamics, an interface is created between structures and aerodynamics. Through this model, structural displacements can be mapped onto the aerodynamics model and the aerodynamic loads can be mapped back onto the structural model. To model the interface, aerodynamic panelling of wing, engines, fin, tail, elevators, ailerons and rudder are generated.

## 3.5 MDO tools

In the previous chapter, section 2.1.4, on page 15 the development of the critical tools required by a successful MDO implementation was described. A major effort was spent on the development and evaluation of tools to support data storage, exchange and comparison of information, process management and process control. As mentioned earlier one of the objectives of the MDO project was to allow the various partners to employ their own in-house and commercial design tools. This required additional flexibility from the MDO software.



### 3.5.1 TDMB

The Technical Data Modeller and Browser is a database environment which contains all the technical data but also offers the capabilities of an editor, graphical browser, a programming interface and network integration. Originally developed by British Aerospace, the capabilities of TDMB have been expanded and modified for the MDO project partners. Within the MDO project, TDMB was developed to support the integration of technical computing tools. It is used to store all the project data and allows the user to store, compare and plot results. TDMB has been designed to have the look and feel of an editor rather than that of a traditional database. This was done in order to support the evolutionary aspect of information and to make it as accessible as possible for the broadest range of engineers. A programming interface was developed which allows external applications in FORTRAN and C, or ICAD to attach easily to the database in order to allow direct data exchange. This is done via both FORTRAN and C subroutine libraries which enable the full range of technical computing tools, from model generation to results post-processing to communicate through a common database. To reduce the amount of data used during a TDMB session, it is possible to partition the database in individual database files, this capability also allows it to control or restrict the read/write access of users to the data. Figure 3.12 shows an example of the database. As can be seen, TDMB caters for object orientated data structures comprising a tree structured network of information nodes. Each node has a name and description associated with it and in its simplest form will have a single value corresponding to a primitive data type. These named information primitives can be grouped together to define higher level data structures, and in turn these can be built into more extensive multiple nested data structures, or may be used to supply an existing generic data structure with additional information. At any stage, an information structure may be defined as a new generic data type, such that when a new node is created of that generic type, an entire information-tree is created. Primitive data types at the lowest level can be character, integer or real. Arrays can be defined for both low and high level data types. A description by formulas can be used to define the value of a node.

### 3.5.2 MMG

The Multi Model Generator offers a unique capability for the rapid and systematic generation of analysis models for aerodynamic, structural, aeroelastic and manufacturing cost simulation of a large commercial transport aircraft wing. It coordinates the execution of a number of aircraft design, definition



| AC_Def_HCO    | AC_Def_HCO   | AC_Def_HCO | This Aircraft_DEFINITION_HCO   |
|---------------|--------------|------------|--------------------------------|
| Status        | Text         | *          | Status of aircraft definition  |
| Specification | *            | *          | This specifies the aircraft    |
| Definition    | *            | *          | This details the full geometry |
| Geometry      | *            | *          | Geometric definition of        |
| Fuselage      | FuselageGe   | *          | Definition of fuselage         |
| Wing          | WingGeom     | *          | Definition of wing             |
| Fin           | FinGeom      | *          | Definition of fin              |
| Tail          | TailGeom     | *          | Definition of tail             |
| Inboard       | EngineGeom   | *          | Definition of inboard          |
| Outboard      | EngineGeom   | *          | Definition of outboard         |
| Allevons      | Allevons     | *          | Allevons definitions           |
| Structure     | *            | *          | Structural definition of       |
| Operations    | Army         | *          | Detailed information about     |
| Manufacturer  | Manufacturer | *          | This details the manufacturing |
| ControlDat    | *            | *          | Control Data for Aerodynamics  |
| Results       | *            | *          | Analysis results representing  |
| Plots         | *            | *          | Plots of the reference data    |
| Assessment    | *            | *          | Overall assessment against the |
| Application   | Text         | *          | External application to model  |

Figure 3.12: Technical Data Modeller and Browser.

and model generation modules.

The main MMG modules are:

- **Definition general arrangement** Using top level specification parameters such as area, aspect-ratio etc., this module generates the detailed information on the general arrangement of the aircraft wing, fin and tailplane. The geometric definition includes the calculation of general arrangement dimensions, parametric definition of the wing surface and surface definition of fuselage, fin and tail.
- **Surface Shape Generator** This module generates the surface shape of the wing. The wing surface geometry is defined by three airfoils, i.e. one at root, crank and tip. A number of additional airfoils has been included to gain more control of the wing surface. (see figure 3.13) After positioning the airfoils in the wing planform, a spanwise interpolation between the airfoils is carried out to create the wing surface.
- **Structural Loads and Sizing** This module defines the structural sizes of the spars, ribs and stringers within the wing surface based on a lifting line assessment of the loads. Aspects of this preliminary sizing will be discussed in more detail in section 3.5.2.1.
- **Finite Element Model Generator** Creates the finite element models of the wing, i.e. geometry, element data and property data. It also generates and defines the optimisation model in terms of design variables and constraints. Both metal and composite models can be generated.

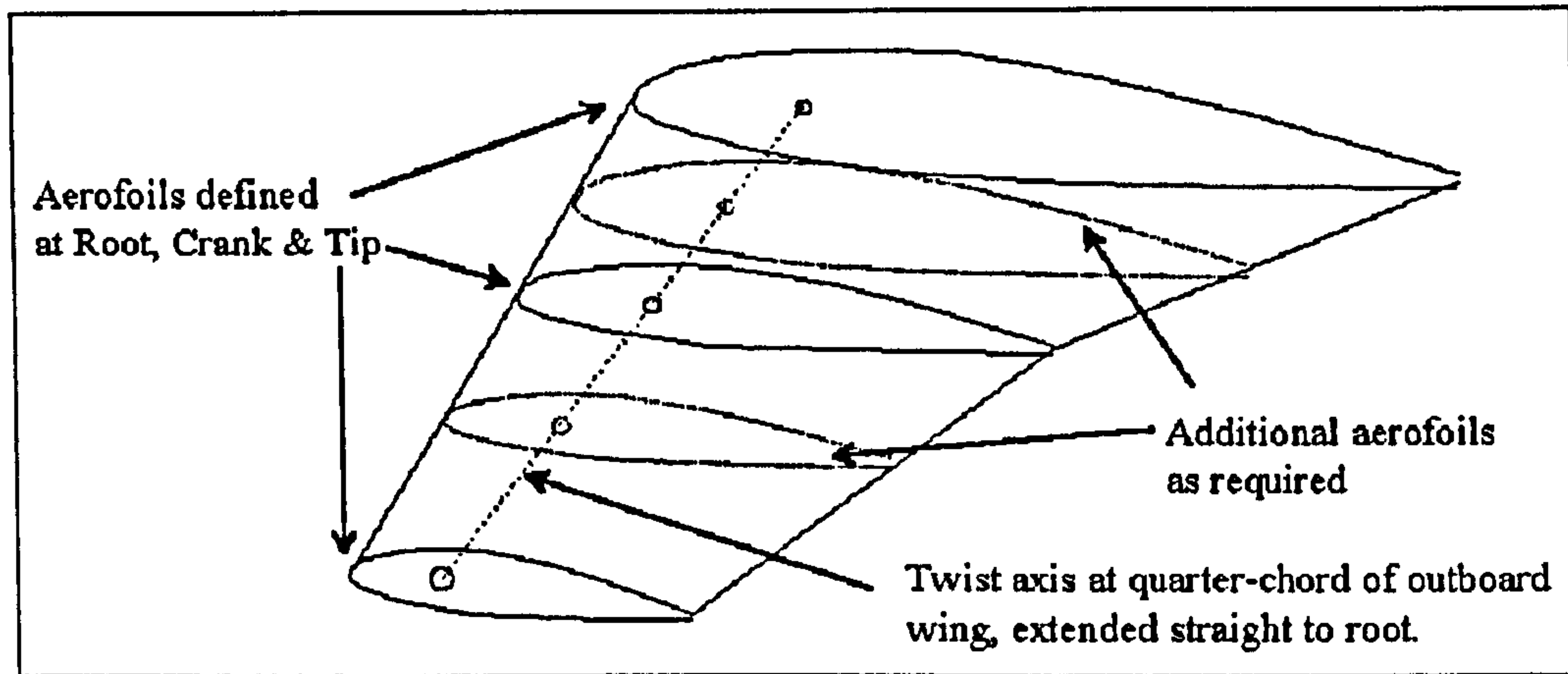


Figure 3.13: Aerofoil definition used by surface shape generator.

- **Aeroelastic Model Generator** This model generates the stick mass and stiffness models, the aerodynamic panelling and the control cards for aeroelastic analysis. Combined with the finite element model generated by the previous module, the aeroelastic model is obtained.
- **Cost Model** Based on TDMB data and previous modules, design information for the main structural components is generated. This module then proceeds with the estimation of recurrent manufacturing cost for the wing. Chapter 5 will give a detailed outline of the cost model.

### 3.5.2.1 Structural Layout Loads and Sizing

Detailed information on this module can be found in references [54] and [58]. This preliminary sizing module has been developed by TU-Delft and uses parts of their in-house developed Aircraft Design and Analysis System (ADAS). The module calculates the loads and sizes which are sensitive to wing design variables such as aspect ratio, wing area, etc. It also estimates the allowable stress levels at each wing rib station. The wing planform, defined by TDMB, is divided into a large number (approx. 100) of strips oriented in the flow direction. On each of these strips, the aerodynamic and inertia loads are calculated separately. After adding both contributions together, the resulting shear forces and pitching moments are integrated along the span. In a last stage, the shear forces and pitching moments are then interpolated to the rib positions and converted to the wing reference axis.

The aerodynamic loads, shear force and pitching moment, are obtained using a preliminary design method [59] for the calculation of the spanwise



lift distribution, the lift incidence and lift coefficient. A distribution for the inertia loads is obtained by breaking down the wing and fuel weight into discrete nodal masses. The forces induced by the inertia are obtained by multiplying masses by the acceleration (Newton's Law) and the moments are obtained in similar way, also taking into account the moment arm with respect to the wing reference axis. Engine mass and centres of gravity have also been included as nodal masses.

The preliminary sizing estimates allowable stress levels based on buckling and fixed stress levels were used to represent material failure and fatigue. The sizing calculation of skin panels and spar webs ( see [58] ) starts off by using a simple one-dimensional beam representation of the wing box. For each rib station all the load cases are checked to determine the maximum positive and negative bending moments, the absolute shear force and the maximum absolute torque.

### Skin and stringer sizing

The loading intensities (load per unit width of the skin panel) for a given bending moment  $M_x$  are calculated using the wing box cross-section at each rib station. For simplification, the wing box cross-sections are represented by a simple beam. To allow for the curvature of the skins and the eccentricity of the stringers with respect to the skin, an effective depth of 80% of the maximum box depth is assumed (See figure 3.14). The stringers were not

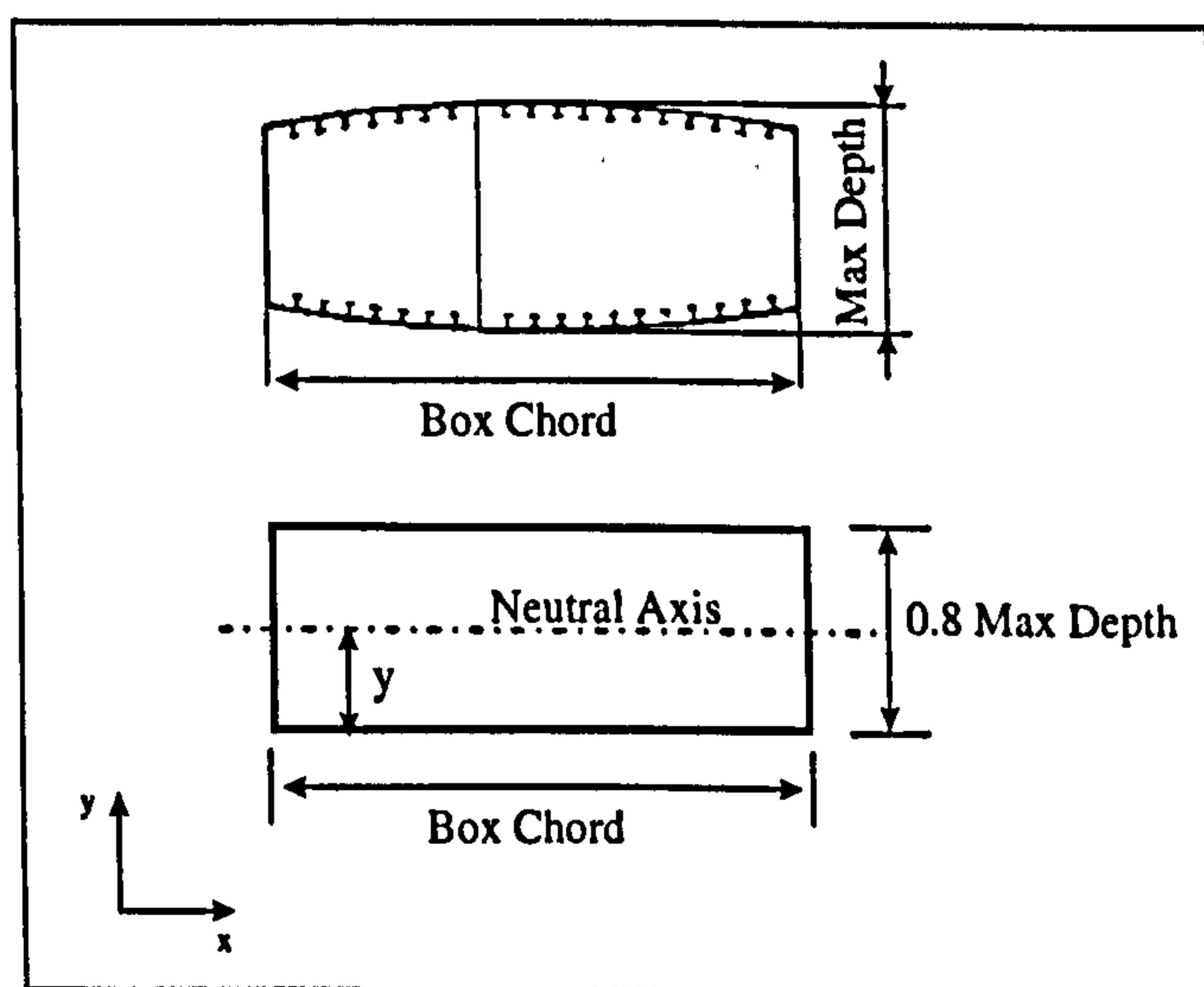


Figure 3.14: Simplified wing box cross-section.

considered separately, but were taken into account in the skin thickness. The

ratio of the smeared stringer thickness to the skin thickness varies from 0.5 to 1.0. A ratio of 0.5 was assumed. Using this representation we know that:

$$M_x = F \times 0.8 \times MaxDepth \quad (3.1)$$

For a given loading case, the loading intensity in the top skin is then obtained from:

$$p = \frac{M_x}{0.8 \times MaxDepth \times BoxChord} \quad (3.2)$$

The loading intensity in the lower skin is equal, but opposite in sign. For the sizing it was assumed that buckling and material failure determine the allowable stress level in the compression skin, while fatigue is the governing criterion for the tension skin. It was also assumed that neither skin buckling nor stringer buckling are allowed to occur below ultimate load, material yielding is not allowed below limit load and material failure is not allowed below ultimate load. The buckling stress allowable is based on simultaneous skin and stringer buckling modes which can be obtained from ESDU [60] data sheets or calculated by :

$$\sigma < \frac{\eta}{j} \sqrt{\frac{pE}{L}} \quad (3.3)$$

With:

$E$ = Young's Modulus, 72000 Mpa

$L$ = rib pitch

$\eta$ = efficiency of the compression panel, depending on the stringer type used it can have theoretical maximum of 0.9. A value of 0.8 was assumed.

Because all the loads in the Technical Data Modeler and Browser were defined at limit load, the ultimate load failure stresses had to be divided by the safety factor  $j=1.5$ .

For the compression skin material failure represents a more critical condition than yield. For Al 7057 the stress limit for material failure has been defined as:

$$\sigma < \frac{\sigma_{fail}}{j} = 387MPa \quad (3.4)$$

Material failure appeared to be the most critical condition for 80% of the wing (from centre to tip), while the buckling was the active criterion for the last 20% of the wing (wing tip). As mentioned earlier, fatigue is assumed to be the governing design criterion for the tension skin and a fatigue stress



level  $\sigma_{fatigue} = 300MPa$  at ultimate load has been selected. Converted to limit load this leads to:

$$\sigma < \frac{\sigma_{fatigue}}{j} = 200MPa \quad (3.5)$$

Using the smearing ratio, the skin thickness and stringer area then follows from:

$$t_{skin} = \frac{2}{3} \times \frac{p}{\sigma_{all}} A_{str} = \frac{1}{3} \times \frac{p}{\sigma_{all}} \text{ timesStringerPitch} \quad (3.6)$$

### Spar sizing

From the absolute shear force and absolute torque (twisting moment), the shear flow for the spars is estimated. It is assumed that the shear flow and spar web thickness is the same for all spars. Assuming that shear force and twisting moment act in the same direction, the shear flow is calculated as:

$$q = \frac{|T_z|}{0.8 \times MaxDepth \times NrSpars} + \frac{|M_y|}{2 \times 0.8 \times MaxDepth \times BoxChord} \quad (3.7)$$

To take into account buckling effects, an allowable shear stress of 80% of the material failure stress under shear was assumed:

$$\tau_{all} = \frac{0.8}{\sqrt{3}} \sigma_{fail} \quad (3.8)$$

For each spar at a given spanwise rib station, the required spar thickness is then obtained by:

$$t_{web} = \frac{q}{\tau_{all}} \quad (3.9)$$

### Allowable Improvements

Early in the project, it became clear that the absolute stress levels were too high for the upper skin and too low for the lower skin. Using BAe experience, a correction was made and the top skin material failure cut-off value was reduced from 387 MPa to 300 MPa and the bottom skin fatigue and damage tolerance was increased from 200 MPa to 240 MPa.

However, these fixed cut-off values are not sensitive to aircraft design changes. For this reason, TU-Delft developed an improved allowables method which included the modelling of the non-linear behaviour of material buckling and a fatigue and damage tolerance criterion. An interactive process

was introduced where the structure is sized to create a preliminary thickness distribution and the allowable stresses for this sized structure.

### Rib Sizing

From the start of the project, it was decided that ribs were not going to be

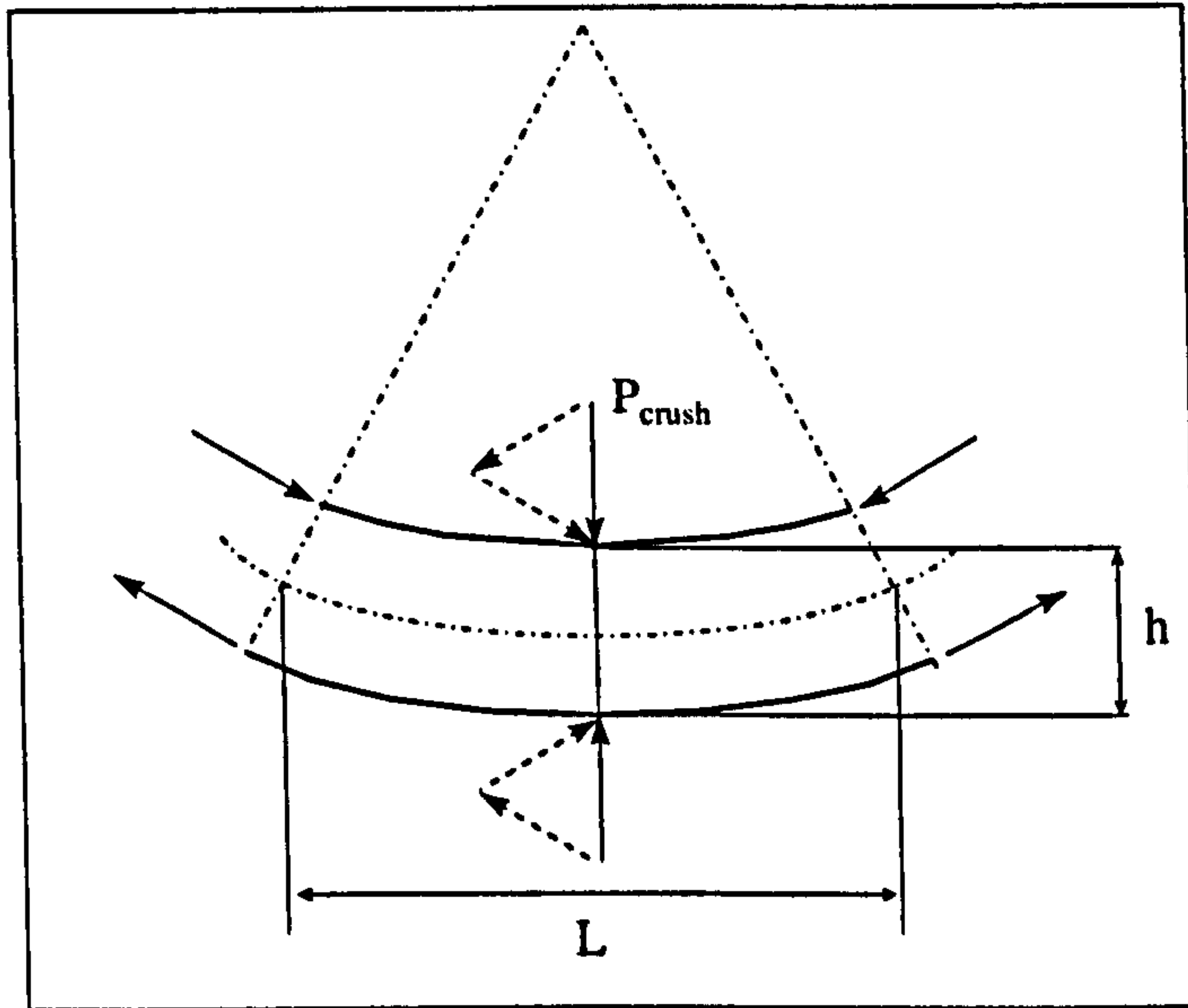


Figure 3.15: Beam subjected to bending.

optimised. Hence, ribs and rib stiffeners were assigned a fixed constant thickness and cross section. However, when applying geometric design variables such as aspect ratio, rear spar position, wing t/c the rib mass contribution changed by a large amount and changes in rib mass had a significant effect on the overall mass sensitivity of the wing box. Hence it was important to size the ribs realistically. For this reason, TU-Delft developed a rib sizing module which sized the rib and stiffeners as compression panels subjected to crushing loads induced by the bending deformation of the wing. Assuming a two-dimensional beam as shown in figure 3.15, the crushing load is caused by the vertical component of the axial forces in the booms of this beam. The crushing load has been calculated in the following way:

$$P_{crush} = \frac{M^2 L}{EIhw} \quad (3.10)$$

with:



M = bending moment  
L = rib pitch  
EI = bending stiffness  
h,w = height and width of torsion box

### 3.5.3 MDO framework

During the project, several frameworks for the implementation and control of MDO processes have been investigated. The specific implementation was different for different partners. Several objectives for the MDO framework were defined (see references [17] [61]):

- Enable aircraft design and analysis engineers to:
  - develop new multidisciplinary design processes.
  - configure design and analysis methods to perform that process.
  - control the execution of MDO processes.
- Facilitate collaboration between companies, for both aircraft design and research projects through the establishment of standards and mechanisms for data exchange.

A total of five commercial or in-house frameworks were assessed by the project: TOSCA (in-house, developed by BAe), SIFRAME (commercial), iSIGHT (commercial), SPINE (commercial), CHAINCE (in-house, developed by TU-Delft). Two frameworks, SPINE and TOSCA, will be briefly discussed below as they were used as part of this thesis. Apart from these five frameworks, individual project partners have created their own MDO methodologies. The main part of the work presented in chapter 6 uses such an in-house developed MDO approach. Although the MDO methodology described in this thesis proved its effectiveness as a benchmark test case, it lacks other capabilities required from an MDO framework, such as flexibility, process control and monitoring, multiple users, data management etc.

#### 3.5.3.1 TOSCA

The TDMB Optimisation Solution Control Agent (TOSCA) was developed by British Aerospace, as part of the MDO project [62]. It is a utility to coordinate the solution of a design optimisation problem. TOSCA acts as an "Agent" or "Assistant" to the design engineer. It executes as a background

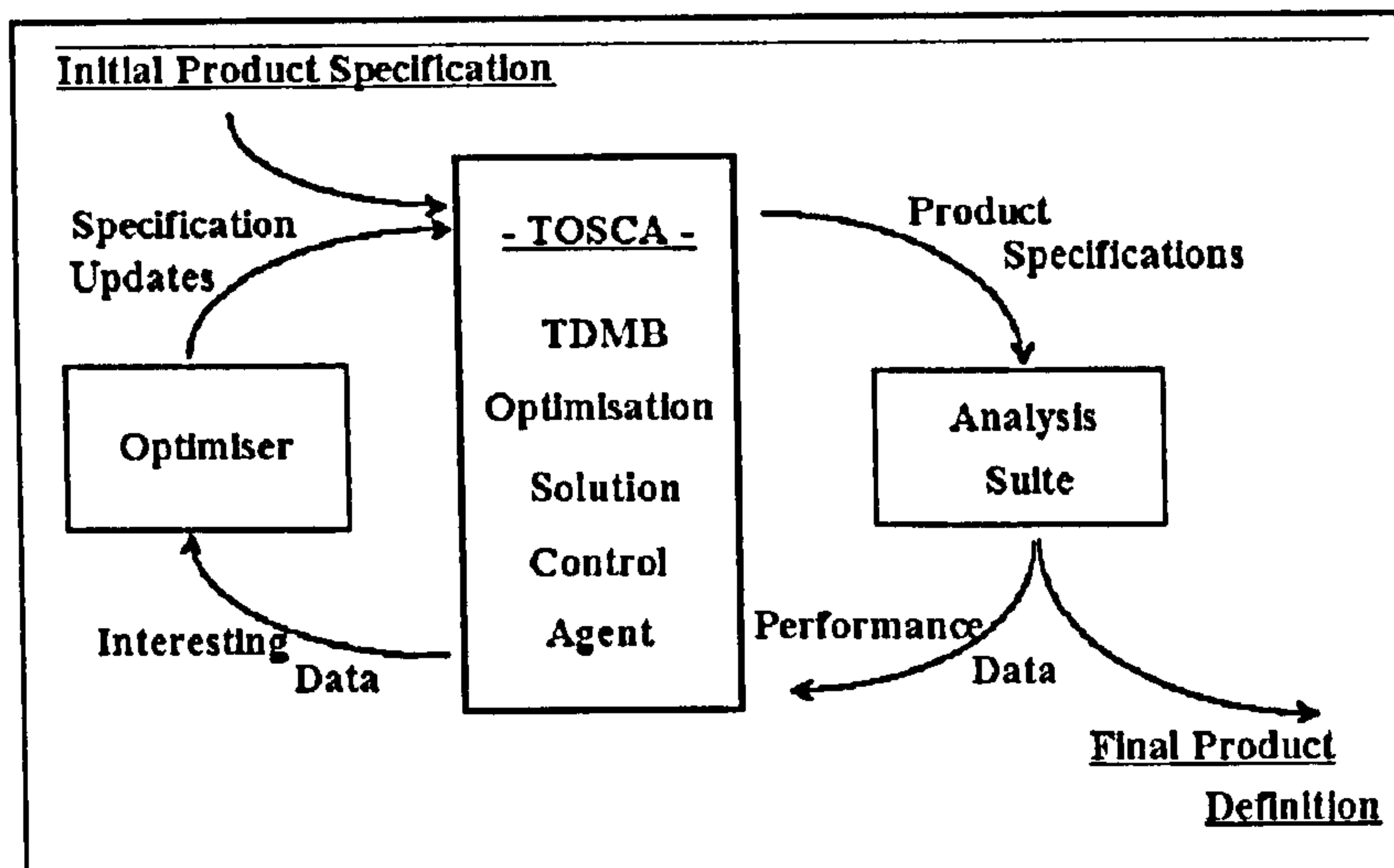


Figure 3.16: TOSCA, an MDO framework developed by British Aerospace.

process controlling the execution of product analysis tools to determine overall product performance and controlling the execution of optimisers to recommend new designs. In this way, having specified a design problem and solution strategy, the design engineer is freed from the mundane issues of job control and file management. Instead, the engineer can focus time and attention on the challenging issues of the engineering validity of the product design ideas being developed and the next issues to be investigated. Figure 3.16 gives an overview of the TOSCA architecture. As can be seen from this figure, the optimisers in TOSCA act as master for the MDO calculations and drive the process. It contains a set of optimisers which allow the designer to make use of several means to find the improved design. One has the choice to use:

- an indirect method by creating first a surface in space, using a matrix of data points, and applying a simple search algorithm onto it.
- a direct method where the optimum is searched from point to point.

For the search a zero order method or gradient method can be applied. A problem with TOSCA is that it does not allow the user to specify or change process structures for the MDO problem. More information about TOSCA can be found in references [62] [57].



### 3.5.3.2 SPINE

SPINE was developed by NLR (see references [61] [63] [64] ). The SPINE-based working environment provides end users with access to resources available from the network, as if these resources were located on the same "virtual" computer. Resources are accessed in a Windows type of environment. Tools can be started by dragging and dropping input files to it, opening then through the menu, etc. SPINE allows people from several disciplines to work together and to combine their knowledge or can be used by one individual on a network or stand alone. Within SPINE, the optimisation module is seen as a tool. As with TOSCA, SPINE does not allow easy change or definition of process structures for the MDO problem.

### 3.5.4 Online Support and Documentation

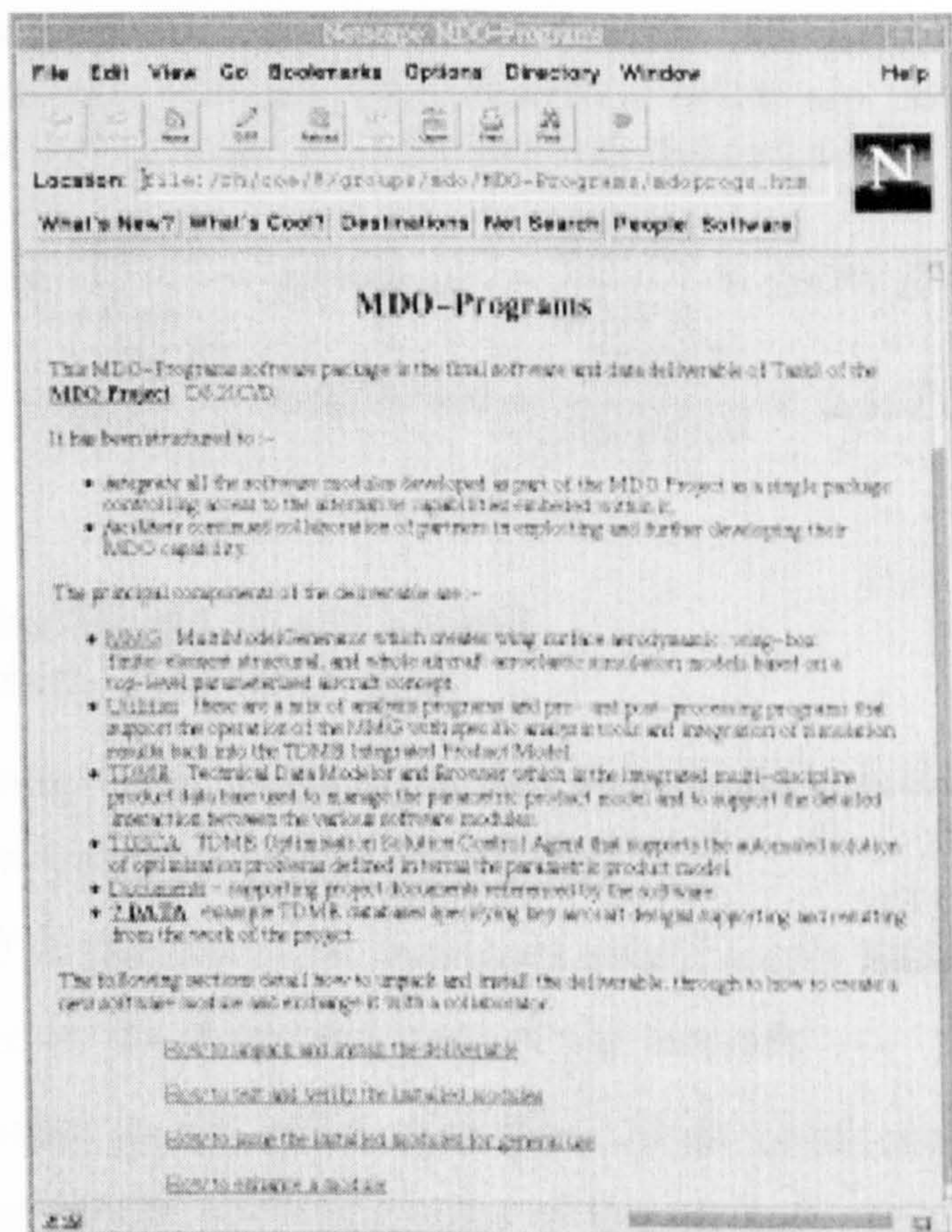


Figure 3.17: Online support documentation.

Online support and documentation has been implemented to assist de-



signers with the use of the MDO software. All explanatory on-line documentation has been generated in html format and Netscape is used for visualisation. Figure 3.17 shows how the on-line documentation is presented to the user. By clicking on the hyper links, the user can browse through the documentation.

## 3.6 Performance measure

### 3.6.1 DOC

The overall objective of the MDO project was to minimise the aircraft direct operating cost (DOC) by investigating changes of certain aircraft variants. The DOC formula used was based on mass and drag. The interaction of drag and mass in the preliminary design of an aircraft is very complex. Drag affects the fuel burn, which in its turn influences the take-off mass of the aircraft. This take-off mass in its turn influences the aircraft and power plant size requirements. The formulas described below are taken from [17], [50] and [31]. The initial objective function was defined as:

$$\Delta DOC = \frac{\delta DOC}{\delta Fuel} \times \Delta Fuel_{DOC} + \frac{\delta DOC}{\delta OWE} \times \Delta OWE + \frac{\delta DOC}{\delta Thrust} \times \Delta Thrust + \frac{\delta DOC}{\delta MTOW} \times \Delta MTOW \quad (3.11)$$

Where:

$OWE$  = Operating Weight Empty aircraft

$MTOW$  = Maximum Take Off Weight

$\frac{\delta DOC}{\delta Fuel} \times \Delta Fuel_{DOC}$  = the direct effect that increments in fuel burn and fuel costs have on the operating cost of the aircraft.

$\frac{\delta DOC}{\delta OWE} \times \Delta OWE$  = the effect that operators weight empty has on acquisition costs and thereby on the operating costs of the aircraft.

$\frac{\delta DOC}{\delta Thrust} \times \Delta Thrust$  = the effect of engine thrust requirement on engine costs and thereby on the operating costs of the aircraft. Engine thrust requirement is related to maximum take-off weight according to take-off field length performance.

$\frac{\delta DOC}{\delta MTOW} \times \Delta MTOW$  = the effect of maximum take-off weight on the op-



erational costs of the aircraft.

The increment in fuel burn, relevant for the economic missions can be written as:

$$\Delta Fuel_{DOC} = \frac{\delta Fuel_{DOC}}{\delta OWE} \times \Delta OWE + \sum_{\text{Economic-Cases}} \left[ \frac{\delta Fuel_{DOC}}{\delta Drag_i} \times \Delta Drag_i \right] \quad (3.12)$$

Where:

$\frac{\delta Fuel_{DOC}}{\delta OWE}$  = the partial derivative of fuel burn with respect to OWE.

$\Delta OWE$  = the increment in OWE between the variants.

$\frac{\delta Fuel_{DOC}}{\delta Drag_i}$  = the partial derivative of fuel burn with respect to the aircraft drag for each operational case.

$\Delta Drag_i$  = the increment in drag between the two aircraft calculated for each operational case relevant to aircraft economics.

The increment in thrust requirement between two aircraft variants can be derived from:

$$\Delta Thrust = \frac{\delta Thrust}{\delta \frac{MTOW^2}{Wing - Area}} \times \Delta \left( \frac{MTOW^2}{Wing - Area} \right) \quad (3.13)$$

With:

$\frac{\delta Thrust}{\delta \frac{MTOW^2}{Wing - Area}}$  = the partial derivative of the thrust requirement with respect to the take-off wing-loading.

$\Delta \left( \frac{MTOW^2}{Wing - Area} \right)$  = the increment in take-off wing-loading between the aircraft.

The increment in Maximum Take-Off Weight between two aircraft variants is derived by adding the change in structural mass and the change in fuel requirement, for the long range design mission:

$$\Delta MTOW = \Delta OWE + \frac{\delta Fuel}{\delta OWE} \times \Delta OWE + \sum_{\text{Range-Cases}} \left[ \frac{\delta Fuel_{range}}{\delta Drag_i} \right] \quad (3.14)$$

With:

$\Delta OWE$  = the increment in OWE between the aircraft.

$\frac{\delta Fuel}{\delta OWE}$  = the partial derivative of fuel requirement with respect to OWE.

$\frac{\delta Fuel_{range}}{\delta Drag_i}$  = the partial derivative of fuel requirement with respect to Drag for each of the operational cases relevant to the long range design mission.

Assuming the optimal mass of the aircraft is defined by  $m_{opt}$  and the drag for the reference wing is given by  $drag$ , then for an aircraft variant  $i$  where a parameter  $p_i$  has been varied, the optimal mass is given by  $m_{opt+i}$  and the drag by  $drag_i$ . The variation (increment) of the empty aircraft operating weight and the wing drag with respect to the parameter changes is given by:

$$\Delta OWE = \frac{(m_{opt} - m_{opt+i})}{\Delta p_i} \quad (3.15)$$

$$\Delta Drag = \frac{(drag - drag_i)}{\Delta p_i} \quad (3.16)$$

However, it soon became clear that a modified form of the DOC formula was required in order to provide a tractable problem for the set time frame of the project. For this reason, a simplified Direct Operating Cost formula was used (see 3.17). This takes into account changes of weight and drag, without direct coupling being employed. Most of the work of this thesis has been based on this formula.

$$\Delta DOC = \sum_{i=1}^n w_1 \times \frac{\Delta M}{\Delta p_i} \times \Delta p_i + \sum_{i=1}^n w_2 \times \frac{\Delta D}{\Delta p_i} \times \Delta p_i \quad (3.17)$$

With:

$i$  = aircraft variant

$p_i$  = parameter being varied

$w_i$  = weighting factor

$\Delta M = (m_{opt} - m_{opt+i})$

$\Delta D = (drag - drag_i)$

### 3.6.2 Simplified MDO process

In principle, an MDO process couples a wide range of design parameters. However, to reduce the complexity of the MDO process a simplified study was introduced which only looked at 6 aircraft variants. These six primary variants involve a mix of planform, surface shape, and structural parameters



|                         | Reference Aircraft Value | Variant Aircraft Value | Increment in Value  | Percent Increment |
|-------------------------|--------------------------|------------------------|---------------------|-------------------|
| Wing QC Sweep           | 33.0°                    | 33.99°                 | 0.99°               | 3%                |
| Wing Area               | 725m <sup>2</sup>        | 746.75m <sup>2</sup>   | 21.75m <sup>2</sup> | 3%                |
| Wing Aspect Ratio       | 8.2                      | 8.446                  | 0.246               | 3%                |
| Wing thickness          | 0.1                      | 0.103                  | 0.003               | 3%                |
| Wing Outboard Twist     | 0.0°                     | 0.25°                  | 0.25°               | n/a               |
| Wing Rear Spar Position | 0.65                     | 0.6695                 | 0.0195              | 3%                |

Table 3.5: Overview six primary variants.

and were selected to provide a common reference basis for the optimisation studies. Table 3.5 shows these variants.

These six primary variants were selected because they are major driving parameters in the design of an aircraft as they have an influence on several disciplines. They ensure that more than one discipline is involved in the problem at the same time (i.e. multi-disciplinary coupling). This multi-disciplinary coupling means that for a change in sweep, for example, both the planform and shape design parameters need to be updated employing the relevant redesign tools. The six variants influence the aircraft performance in the following ways:

- Wing sweep: major factor in controlling wave-drag characteristics, transonic cruise performance of aircraft and aeroelastic behaviour of the wing.  
Invokes changes in planform and shape.
- Wing area and aspect ratio: major factors in controlling the overall lift, mass and vortex drag characteristics of aircraft.  
Invokes changes in shape.
- Wing thickness: major factor in controlling the trade-off between overall wing structural mass and aerodynamic drag.  
Invokes shape and structural changes.
- Wing outboard twist: major factor in controlling the spanwise loading distribution of the wing and interaction between vortex drag and wing structural mass.  
Invokes planform, shape and structural changes.

- Wing rear spar position: major factor in controlling the chord, mass and torsional stiffness and effectiveness of aileron control system. Invokes changes of structures and control.

Figure 3.18 shows how a change of these six variants actually affects the wing layout.

The simplified MDO process requires evaluating the sensitivity of the aircraft wing to changes in these parameters. In case of the weight component, the procedure needs to calculate a minimum weight configuration using sizing parameters for a reference wing and then repeat this operation sequentially for the same wing with a small variation in one of the parameters. The numerical values computed through this process provided the inputs to the finite difference expressions given in equation 3.17. Initially, only a three percent change of these parameters was studied for the derivation of primary sensitivities and comparison of the analysis tools used within the project. However, it was soon found that the results from the three percent changes indicated that the optimisation process was stable and that the preliminary sizing proved to be accurate. Hence bigger parameter variations were applied and additional parameters were included in the optimisation studies. Table 3.6 on page 70 gives an overview of the parameters that have been used during the more detailed MDO studies. As can be seen, some of the parameters were originally planned to be taken into account in the study, but were later omitted because of lack of time.

### 3.7 Interaction Thesis - MDO Project

All the MDO-work which has been reported in this chapter was the result of intensive team work between all partners. The work reported in this thesis focuses on structural optimisation and cost analysis. However, certain parts of this work relied and interacted closely with work which was carried out by other members of the project team. This was mainly the case for the preliminary sizing and aerodynamics.

For example, the development of the cost models as part of this thesis showed the potential for rib and stringer pitch optimisation with respect to cost. However, it was also identified that proper sizing of ribs was required to allow for good cost predictions and to take into account crushing effects. Interesting cost modelling results from this thesis, encouraged TU-Delft to enhance their rib preliminary sizing module. In consequence, an intense interaction with regards to development, testing and exchange of results developed. This co-operation allowed for the successful development of



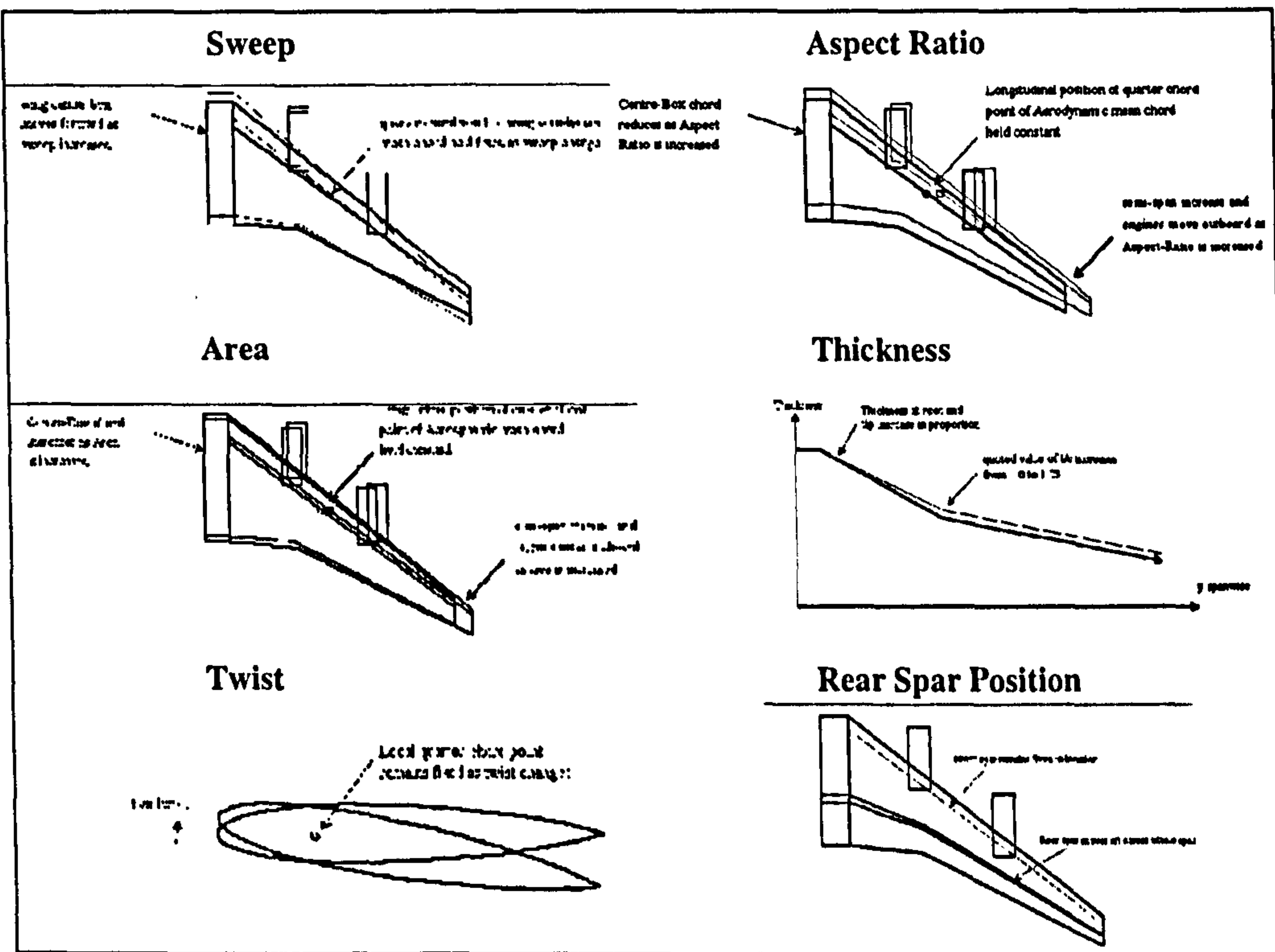


Figure 3.18: Six primary wing variants.

an improved set of cost models for this PhD work, of which the results are presented in further chapters.

To conclude, the involvement of the research as part of the MDO project made it certainly more challenging dynamic and realistic. This, not only from the technical point of view, but also from the aspect of personal experience and communication skills. Not everyone gets a chance to work for two years with technical experts from fourteen different aerospace companies, research institutes and universities located all over Europe...



|                      | Simplified Process | Planform Study | Shape Study | Structure Study | Control Study |
|----------------------|--------------------|----------------|-------------|-----------------|---------------|
| <b>PLANFORM</b>      |                    |                |             |                 |               |
| Aspect Ratio         | S                  | X              | X           | X               | c (X)         |
| Sweep                | S                  | X              | X           | X               | c (X)         |
| Wing Area            | S                  | X              | X           | X               | c (X)         |
| Taper Ratio          | c                  | X              | c           | c               | c             |
| Kink Location        | c                  | X              | c           | c               | c             |
| Engine Location      | c                  | X              | c           | c               | c             |
| <b>SHAPE</b>         |                    |                |             |                 |               |
| Overall Thickness    | S                  | X              | X           | c (X)           | X             |
| T/c Distributions    | c                  | c              | X           | c               | c             |
| Camber Distributions | c                  | c              | X           | c               | c             |
| Twist Distributions  | c                  | X              | X           | c               | c             |
| <b>STRUCTURE</b>     |                    |                |             |                 |               |
| Spar Position        | S                  | c (X)          | c (X)       | X               | X             |
| Stringer Pitch       | c                  | c              | c           | X               | c             |
| Rib Pitch            | c                  | c              | c           | X               | c             |
| Stringer Section     | c                  | c              | c           | X               | c             |
| Materials            | c                  | c              | c           | X               | c             |
| <b>CONTROLS</b>      |                    |                |             |                 |               |
| Aileron Size/Pos     | -                  | -              | -           | -               | c (X)         |
| Control Law          | -                  | -              | -           | -               | c (X)         |
| <b>RESPONSES</b>     |                    |                |             |                 |               |
| Drag                 | R                  | R              | R           | R               | R             |
| Structural Mass      | R                  | R              | -           | R               | R             |
| Flutter Damping      | R                  | R              | -           | -               | R             |
| Structural Cost      | -                  | -              | -           | R               | -             |
| Actuator Activity    | -                  | -              | -           | -               | R             |
| Dynamic Response     | -                  | -              | -           | -               | R             |

Table legend:

S, X : design variable for sensitivity/optimisation study

c : design variable is fixed

- : design variable is ignored

R : response used in objective function or constraints

Table 3.6: Overview key design variables

# Chapter 4

## Structural Optimisation

This chapter discusses the structural optimisation of the MDO wing. It describes the optimisation model, shows results for optimisation using MSC/NASTRAN and STARS and highlights some of the issues which are necessary to obtain good optimisation results.

### 4.1 Optimisation problem

For the structural optimisation problem, the wing was subdivided into fifteen zones, each containing several ribs. Figure 4.1 shows the position of each zone on the wing. Each zone contains ten or eleven size design variables depending on whether the centre spar is included or not. The top and bottom skin are subdivided in the chordwise direction into four panels. The individual size design variables are:

- Four size variables covering the thickness of the top skin, including the stringers and spar caps.
- Four size variables covering the thickness of the bottom skin, including the stringers and spar caps.
- One size variable for each spar web.

The size design variables are also shown in figure 4.1. As the centre spar only extends as far as the crank, the remainder of the wing box has two spars and, hence we have a design variable less. Stringers and skins are part of one design variable with skin thickness being proportional to stringer area. This design variable also includes the spar caps. Basically, the spar cap is assumed to be another stringer, but with a different proportional relationship between the skin and the spar cap. Originally, the spar caps were defined as separate



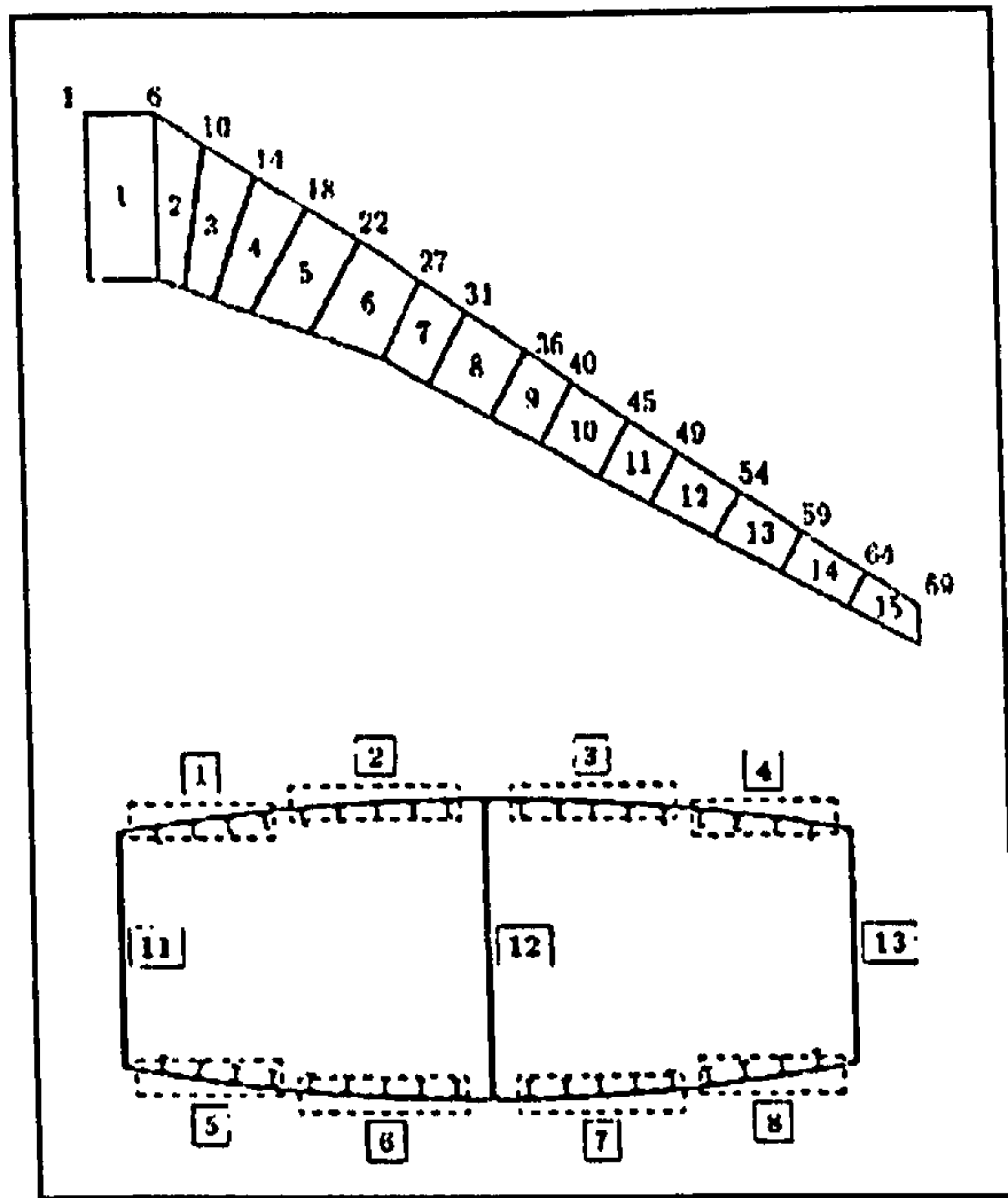


Figure 4.1: Design variable definition.

design variables, but for reasons of convergence (see section 4.6) they were omitted. In total 156 design variables were used. The rib webs were not defined as size variables in order to simplify the optimisation problem and, hence did not take part in the optimisation process.

The objective of the structural optimisation was to minimise the weight, subjected to size and stress constraints. For the membrane and bar elements of the upper and lower skin, the stress limits are on the axial stresses (stresses in spanwise direction, i.e.  $\sigma_x$  in element coordinate system), while the shear elements of the spar webs have limits on the average shear stress. As discussed in section 3.5.2.1 on page 3.5.2.1 the stress limits are based on material limits, local buckling of the skin, shear buckling, general buckling and fatigue life which have been estimated by the preliminary sizing routine. During optimisation, these stress limits are kept constant. Figure 4.2 shows the stress limits which were applied for the optimisation for skin & stringer panels and spar webs. These values apply to the reference aircraft with the reference stringer pitch of 0.195m and 69 ribs. As can be seen for the skin & stringer panels in compression, only approximately the outer 30% of the wing span is sized by buckling criteria while most of the wing is sized by the strength of the material (i.e. -300 MP). The allowable stress for the skin in tension is 240 MPa.



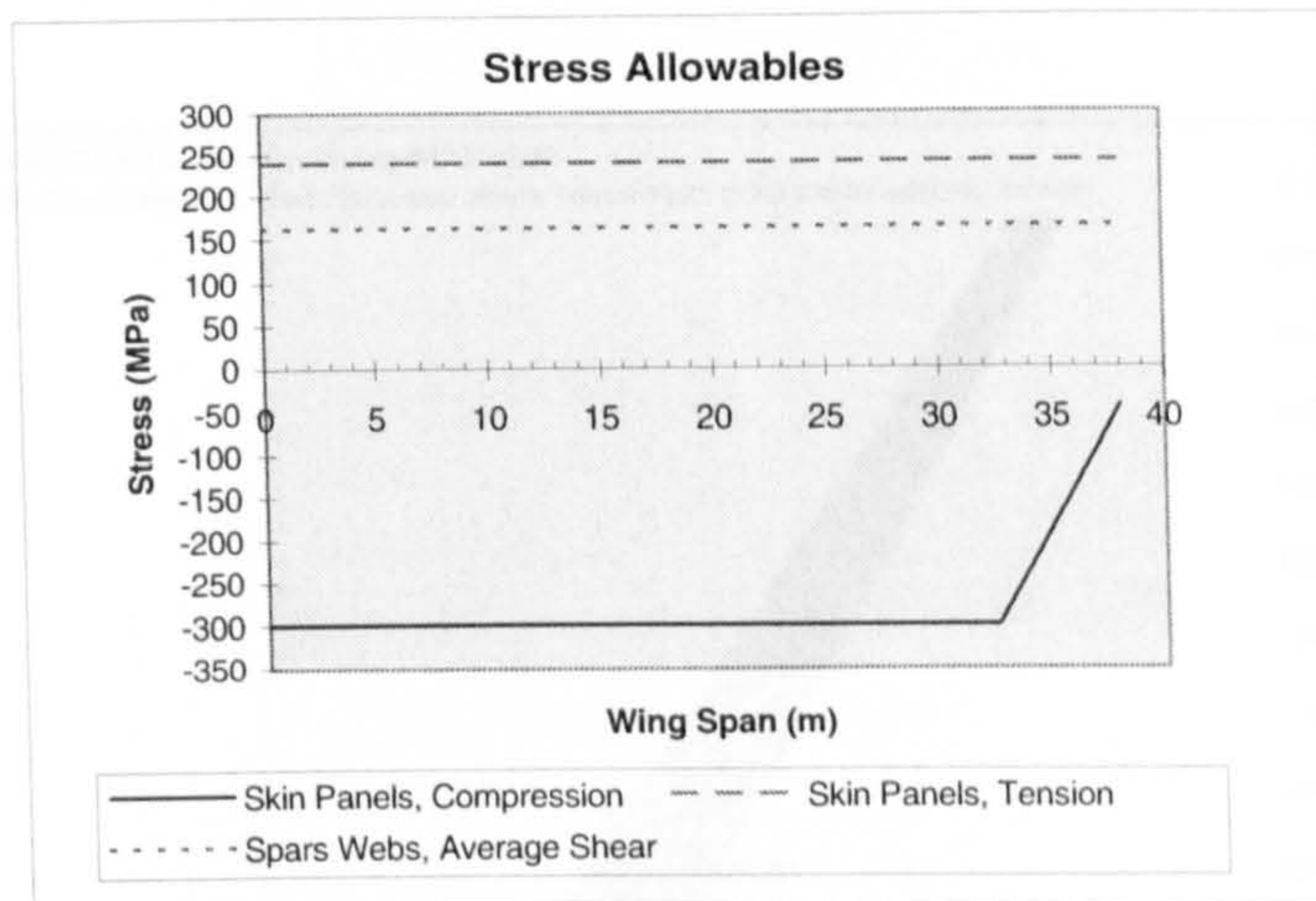


Figure 4.2: Allowable stresses top, bottom skin & stringer panels and spar webs.

## 4.2 Static Analysis

Static analysis, using MSC/NASTRAN was carried out on the Reference model. Based on the preliminary sizing, the initial estimated wingbox mass is around 30500 kg. Table 4.1 gives the initial weight breakdown of the FE model, based on values from preliminary sizing. All weight results shown in this table and in all the other weight results discussed in this thesis are for a wing box structure of a single wing. (i.e. semi-span)

Figure 4.3 shows the stress levels for the +2.5g pull-up condition in the top and bottom skin. Only the +2.5g case is presented, as the push-down manoeuvre was less critical. As can be seen from the figures, the highest stresses are concentrated between rib 6 (wing-fuselage connection) and rib 27 (inboard engine). A tip deflection of approximately 5.9 m was observed for the pull up condition and -2.4 m for the push down condition. The wing twist at the tip counted approximately  $-0.197^\circ$  (pull up case) and  $0.048^\circ$  (push down case)

## 4.3 Optimisation using MSC/NASTRAN

### 4.3.1 Algorithms and options

The MSC/NASTRAN Design Sensitivity and Optimisation software offers three different optimisation algorithms to the user: modified feasible directions (see section 4.3.1.2), sequential linear programming and sequential



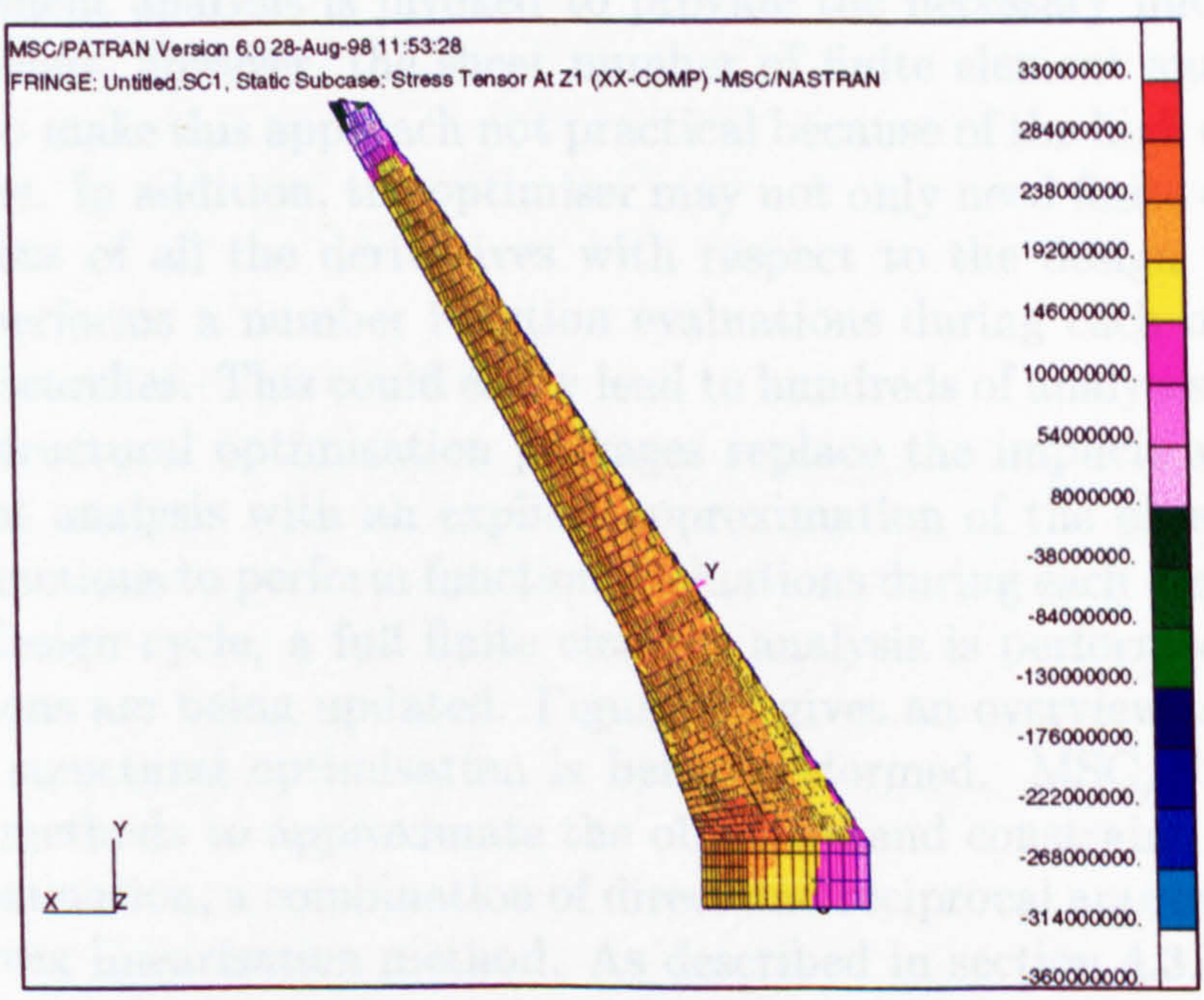
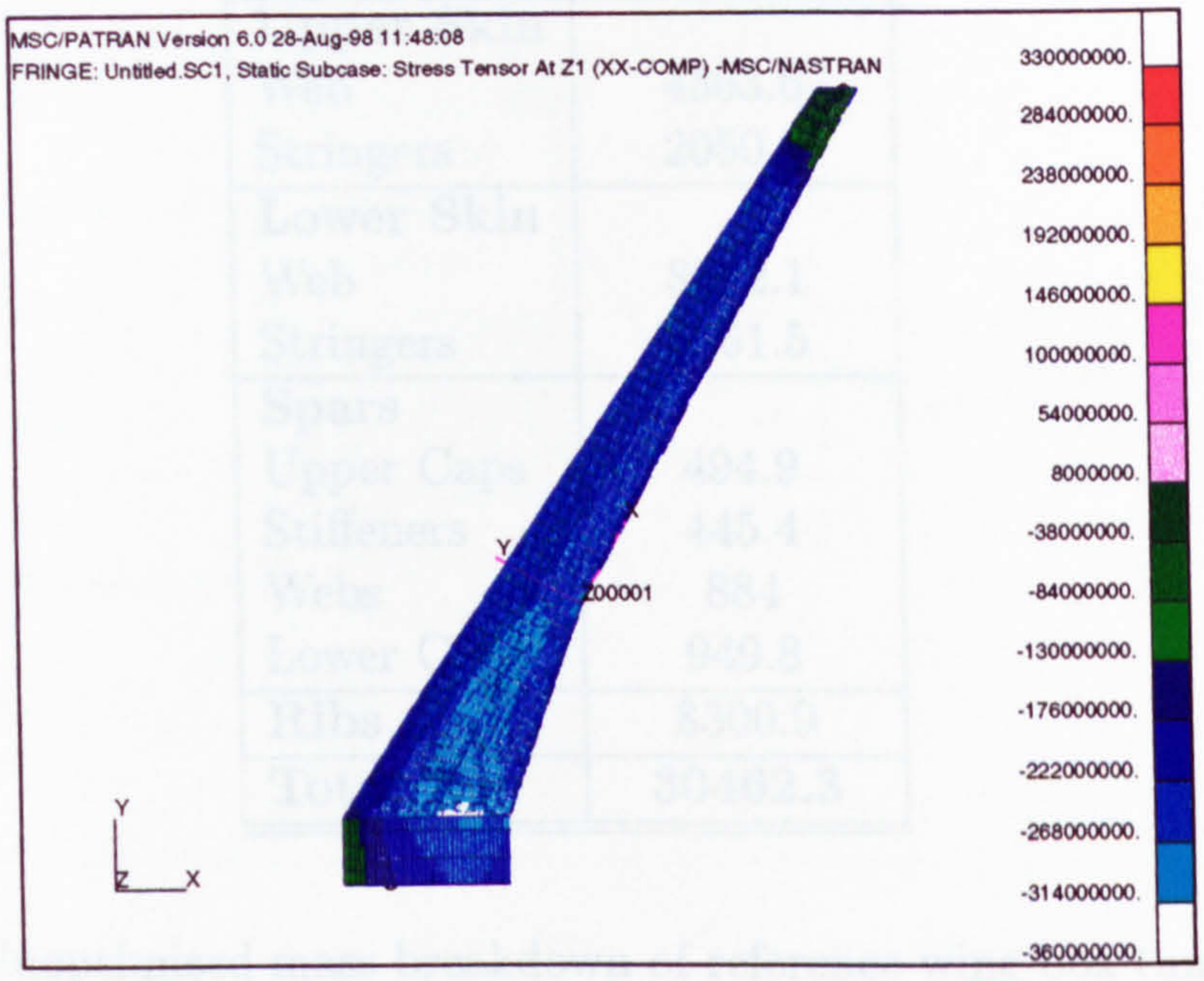


Figure 4.3: Initial stresses ( $\sigma_x$ ) in top skin and bottom skin, pull-up load case.



| Component         | Weight (kg)    |
|-------------------|----------------|
| <b>Upper Skin</b> |                |
| Web               | 4563.6         |
| Stringers         | 2050.1         |
| <b>Lower Skin</b> |                |
| Web               | 8812.1         |
| Stringers         | 3961.5         |
| <b>Spars</b>      |                |
| Upper Caps        | 494.9          |
| Stiffeners        | 445.4          |
| Webs              | 884            |
| Lower Caps        | 949.8          |
| <b>Ribs</b>       | 8300.9         |
| <b>Total</b>      | <b>30462.3</b> |

Table 4.1: Unoptimised mass breakdown of reference wing box (units in kg).

quadratic programming methods (see sections 4.3.1.3 and 4.3.1.4).

In general, whenever a structural optimiser needs a function evaluation, the finite element analysis is invoked to provide the necessary information. For big problems, however, the sheer number of finite element analyses required tend to make this approach not practical because of the high computational expense. In addition, the optimiser may not only need finite difference approximations of all the derivatives with respect to the design variables, but it also performs a number function evaluations during each of the one dimensional searches. This could easily lead to hundreds of analyses. For this reason, all structural optimisation packages replace the implicit and costly finite element analysis with an explicit approximation of the objective and constraint functions to perform function evaluations during each design cycle. After each design cycle, a full finite element analysis is performed and the approximations are being updated. Figure 4.4 gives an overview of how the NASTRAN structural optimisation is being performed. MSC/NASTRAN offers three methods to approximate the objective and constraint functions: direct approximation, a combination of direct and reciprocal approximations, and the convex linearisation method. As described in section 4.3.1.1 all approximations are based on first order information and use Taylor series expansions.

The NASTRAN optimisation results which are discussed in section 4.3.2 were obtained using the Sequential Quadratic Programming algorithm and



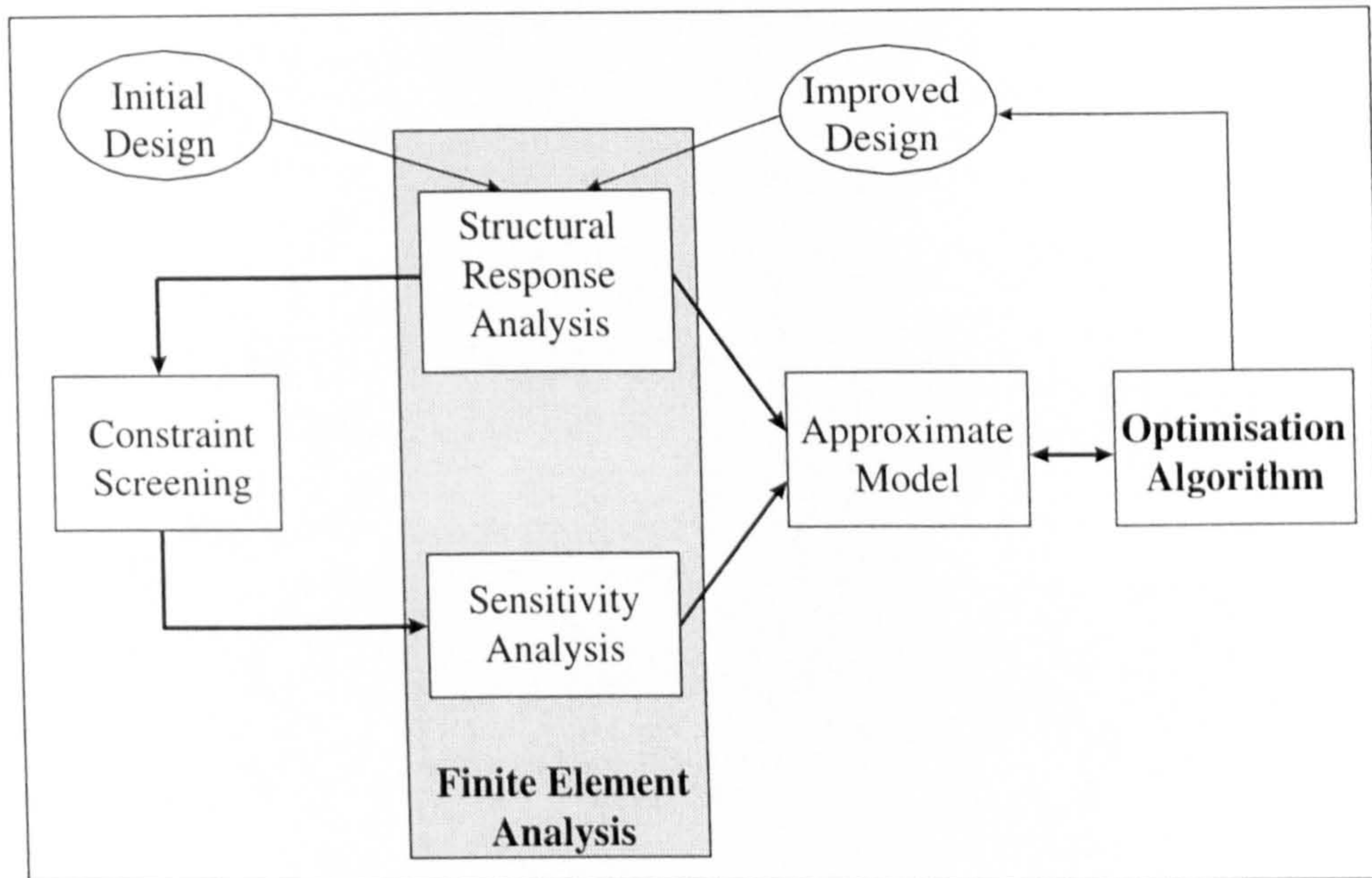


Figure 4.4: NASTRAN structural optimisation algorithm.

a combination of direct and reciprocal approximations. As is discussed in section 4.6, the selection of the optimisation algorithm and the function approximation method was not so straightforward. It is based on numerous strength optimisation runs using different combinations of the above mentioned algorithms and approximation methods. A lot of these investigations were performed by the University of Delft, as part of the MDO project.

#### 4.3.1.1 Approximation methods

As mentioned earlier, formal approximations have been introduced in order to reduce the number of costly finite element analyses. Using references [65] [66] [67] this section first explains the principle of using formal approximations.

Within MSC/NASTRAN the objective and constraints are approximating functions which are based on Taylor series expansions. Consider the function  $f(x)$  of a single variable, the Taylor expansion about the point  $x^*$  can be written as

$$f(x) = f(x^*) + \frac{df(x^*)}{dx}(x - x^*) + \frac{1}{2} \frac{d^2 f(x^*)}{dx^2}(x - x^*)^2 + \dots \quad (4.1)$$

With  $x - x^* = \Delta x$ , Taylor's expansion of equation 4.1 now becomes

$$f(x^* + \Delta x) = f(x^*) + \frac{df(x^*)}{dx} \Delta x + \frac{1}{2} \frac{d^2 f(x^*)}{dx^2} \Delta x^2 + \dots \quad (4.2)$$



In a more general case, when considering a function  $f(x_1, x_2, \dots, x_n)$  which contains multiple variables  $(x_1, x_2, \dots, x_n)$  the matrix notation of the Taylor expansion from equation 4.2 can be written as

$$f(\mathbf{x}^* + \Delta\mathbf{x}) = f(\mathbf{x}^*) + \nabla f^T(\mathbf{x}^*)\Delta\mathbf{x} + \frac{1}{2}\Delta\mathbf{x}^T\mathbf{H}\Delta\mathbf{x} + \dots \quad (4.3)$$

With  $\mathbf{x}$  and  $\mathbf{x}^*$  being  $n$  dimensional vectors,  $\nabla f(\mathbf{x}^*)$  being the gradient vector (i.e. a column vector of the form  $\partial f/\partial\mathbf{x}$ ) and  $\mathbf{H}$  being the Hessian (i.e. an  $n \times n$  matrix which can be written as:  $\frac{\partial^2 f}{\partial\mathbf{x}\partial\mathbf{x}}$ ).

As the approximations in MSC/NASTRAN are only of the first order, only the first derivative term of the Taylor series of equation 4.3 is used. The approximations of the objective and constraint functions thus become

$$f(\mathbf{x}^* + \Delta\mathbf{x}) = f(\mathbf{x}^*) + \nabla f^T(\mathbf{x}^*)\Delta\mathbf{x} \quad (4.4)$$

$$g_j(\mathbf{x}^* + \Delta\mathbf{x}) = g_j(\mathbf{x}^*) + \nabla g_j^T(\mathbf{x}^*)\Delta\mathbf{x} \quad (4.5)$$

The gradient information for equations 4.4 and 4.5 is obtained using design sensitivity analysis. A sensitivity coefficient is defined as the partial derivative of a response with respect to a design variable, i.e.

$$\frac{\partial r_j}{\partial x_i} \quad (4.6)$$

where  $r_j$  is a general response quantity. Using the chain rule for differentiation, equation 4.6 can be written as

$$\frac{\partial r_j}{\partial x_i} = \frac{\partial r_j}{\partial\{u\}} \frac{\partial\{u\}}{\partial x_i} \quad (4.7)$$

with  $\{u\}$  being the displacement solution.

The first term of equation 4.7 can be easily determined using the relationship between stress and displacement which is for the finite element analysis defined by

$$\{\sigma\} = [D][B]\{u\} \quad (4.8)$$

where  $[D][B]$  is the stress-displacement transformation matrix,  $\{\sigma\}$  the stress vector,  $\{u\}$  the displacement vector.

The second term of equation 4.7 is obtained based on Hooke's law. [65] [66]

$$[K]\{u\} = \{P\} \quad (4.9)$$

Differentiating equation 4.9 with respect to a given design variable  $x_i$  gives

$$\frac{\partial[K]}{\partial x_i}\{u\} + [K]\frac{\partial\{u\}}{\partial x_i} = \frac{\partial\{P\}}{\partial x_i} \quad (4.10)$$



Since the applied external loads  $P$  are not changed by any change in design variables, one can assume that

$$\frac{\partial\{P\}}{\partial x_i} = 0 \quad (4.11)$$

Thus equation 4.10 can now be reduced to

$$[K] \frac{\partial\{u\}}{\partial x_i} = -\frac{\partial[K]}{\partial x_i} \{u\} \quad (4.12)$$

Having the global stiffness matrix  $[K]$  as only unknown, equation 4.12 can be solved for  $\frac{\partial\{u\}}{\partial x_i}$ . The expression  $\frac{\partial[K]}{\partial x_i}$  is directly related to  $\frac{\partial[k_i]}{\partial x_i}$  with  $k_i$  being the element stiffness matrix in which the design variable  $x_i$  is present. (all other terms of the global stiffness matrix are zero, since they are not a function of  $x_i$ ) The term  $\frac{\partial[k_i]}{\partial x_i}$  is then solved using finite differences.

The method described above, and used by MSC/NASTRAN is a Semi-Analytical method, being a compromise between pure analytical methods and finite differencing. Computationally, they are a reasonably efficient and fast procedure as it generally does not take long to generate element stiffness matrices and as it does not require much changing or involvement within the finite element code. [66]

MSC/NASTRAN uses two basic types of function approximation, the direct approximation (see equation 4.5) and the reciprocal approximation which is defined as

$$g_{jR}(\mathbf{x}^* + \Delta\mathbf{x}) = g_j(\mathbf{x}^*) + \nabla g_j^T(\mathbf{x}^*) \frac{\mathbf{x}^*}{\mathbf{x}} \Delta\mathbf{x} \quad (4.13)$$

These two approximation methods can be used in three different ways within MSC/NASTRAN:

- Direct approximations.  
With this option both objective function and constraints use direct approximation. NASTRAN advises to only use this option if it is known that all the structural responses are well approximated by linear functions in the design variables. If not, a greater number of approximate optimisation cycles might be required before convergence is achieved.
- Mixed approximations.  
This option is the default option. It is a combination of direct and reciprocal approximations. The direct approximations are used for volume, weight, element force and buckling load responses. The reciprocal

approximation is used for all other responses, such as stress, flutter etc. This method works well in a wide variety of problems and is very reliable. [65] [68]

- **Convex Linearisation.**

In this option, the response functions are approximated using direct or reciprocal approximation. To decide which method to use, one looks at the difference between the direct and reciprocal approximation:  $g_j(\mathbf{x}^* + \Delta\mathbf{x}) - g_{jR}(\mathbf{x}^* + \Delta\mathbf{x})$ . Depending on the sign of this difference one of the approximation methods is chosen. Always the method which gives the largest estimation of the function is chosen. Hence, if the difference is positive, the direct approximation is selected and if the difference is negative, the reciprocal approximation. This selection is carried out on an individual design variable basis, and thus for a single constraint a combination of both approximations may be present. [65]

#### 4.3.1.2 Method of Feasible Directions

The method of feasible directions has been used by many algorithms for solving a constrained non-linear programming problem by using a sequence of one-dimensional minimisations along usable-feasible directions. This method has been classified as a direct method, as it considers the constraints directly as the limiting surfaces. [66] The non-linear optimisation problem can be written as

$$\begin{aligned} \text{Minimise } W &= f(x) \\ \text{Subject to } g_j(x) &\leq 0 \quad j = 1, \dots, m \end{aligned} \tag{4.14}$$

This method assumes that the equality constraints can be eliminated and that the derivatives  $\frac{\partial f}{\partial x_i}$  and  $\frac{\partial g_j}{\partial x_i}$  are available. The method is based on successive one-dimensional problems, each of which minimises the objective function along a search direction. As shown in equation 4.15 a line search is being performed in a direction  $\mathbf{S}_q$ , starting from a feasible point  $\mathbf{x}_q$  and leading to a new feasible point  $\mathbf{x}_{q+1}$  which minimises the value of the objective function.

$$\mathbf{x}_{q+1} = \mathbf{x}_q + \alpha \mathbf{S}_q \tag{4.15}$$

with  $\mathbf{x}$  being an n-dimensional vector,  $\alpha$  being the step size and  $\mathbf{S}$  being the direction vector.

The feasible direction algorithm essentially consists of two parts [65] [67] [66]:



1. Find a direction vector  $S_q$  for each step of the iteration so that in equation 4.15 the vector  $x_{q+1}$  will improve the design, while still being in the feasible region.
2. Determine the step size  $\alpha$  by which one will move along the selected direction vector and which still leads to a feasible point  $x_{q+1}$ .

For the selection of the direction vector  $S_q$  two conditions must be satisfied: the vector has to be *feasible* and *usable*. The following definitions apply:

- Feasible = the direction vector has to remain in the feasible region, i.e.

$$S_q \nabla g_j(x_q) \geq 0 \quad (4.16)$$

- Usable = the direction vector is in the direction of descent, i.e.

$$S_q \nabla f(x_q) < 0 \quad (4.17)$$

Figure 4.5 shows the cone of feasible directions and shows which ones are feasible and usable. As can be seen many useable and feasible directions are possible.

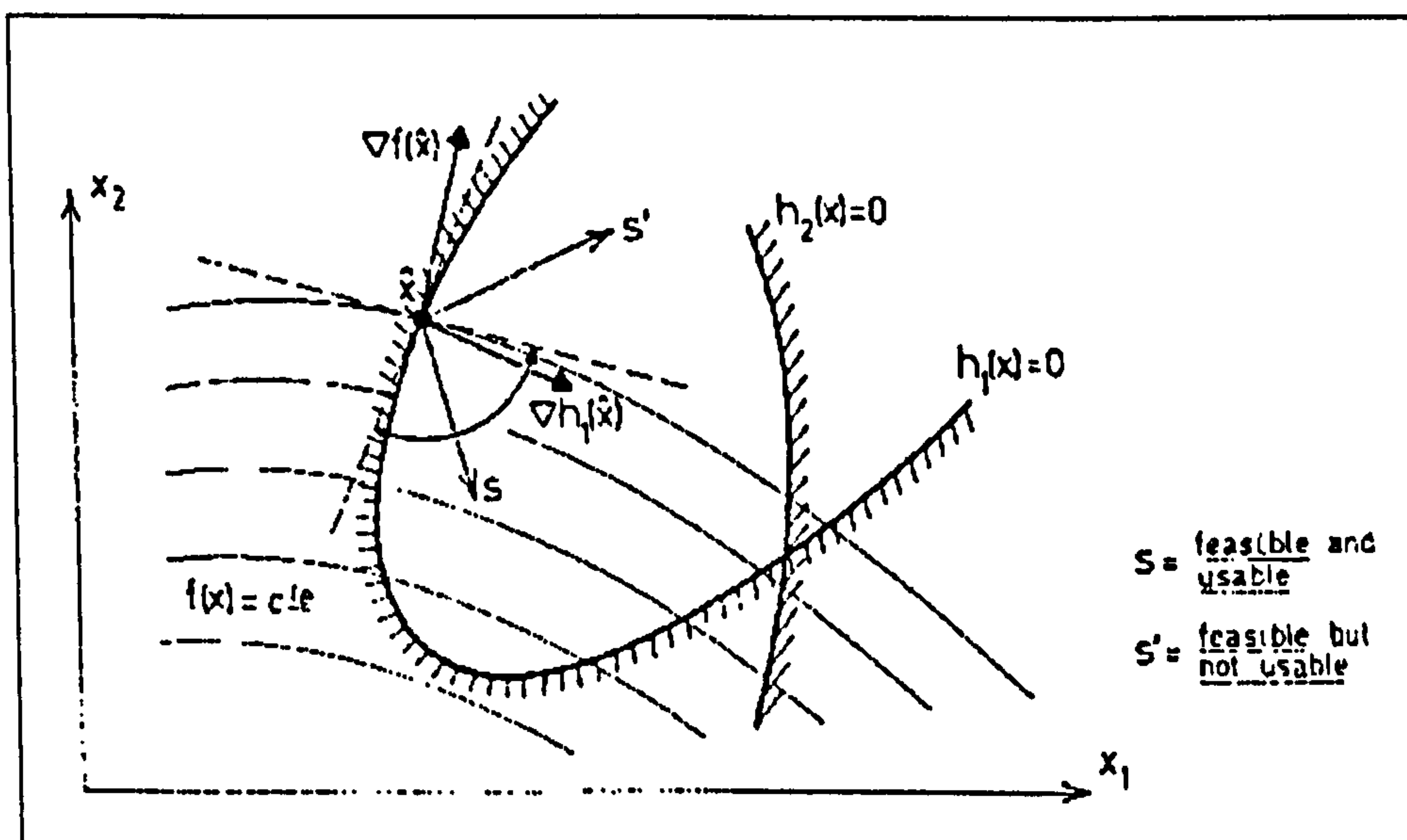


Figure 4.5: Cone of feasible directions.

Within MSC/NASTRAN, the direction vector can be determined in several ways. The steepest descent is used as direction vector for cases where

there no active or violated constraints, i.e.

$$\mathbf{S}_q = -\nabla f(\mathbf{x}_q) \quad (4.18)$$

In other cases the direction finding problem is defined as a linear programming sub-problem. This problem is solved to find a direction vector which leads to the maximum decrease of the objective function. The principle of the sub-problem is given in equation 4.19. [66] [67] Nastran uses similar sub-problems to find search directions which reduce the objective without violating any active constraints or which, in case of constraint violation finds a direction vector which brings the problem back in the feasible region.

$$\begin{aligned} & \text{Maximise } \beta \\ & \text{Subject to } \mathbf{S}_q^T \nabla f + \beta \leq 0 \\ & \mathbf{S}_q^T \nabla g_j - \theta_j \beta \geq 0 \\ & -1 < \mathbf{S}_q < 1 \end{aligned} \quad (4.19)$$

Where  $\nabla g_j$  are the number of active constraints,  $\theta_j$  are arbitrary positive constants and the last constraint is a bound on the magnitude of vector  $\mathbf{S}_q$ .

The physical meaning of equations 4.19 is that by increasing  $\beta$  the term  $\mathbf{S}_q^T \nabla f$  needs to be decreased. As the scalar product of two vectors is the magnitude of the two vectors times the cosine of the angle between them, it is clear that if we want to decrease the term  $\mathbf{S}_q^T \nabla f$  the cosine of the angle between the two vectors has to be as large negative as possible. On the other hand, the second constraint has been introduced to ensure feasibility of the problem. For  $\beta_{max} > 0$  the selected feasible direction is also a direction of descent. If  $\beta_{max} = 0$ , the initial point  $\mathbf{x}_q$  is a local minimum.

In order to prevent that the direction vector would be perpendicular to the gradient vectors of the active constraints  $\nabla g_j$ , the constants  $\theta_j$  have been introduced. Having the direction vector and  $\nabla g_j$  perpendicular ( $\theta_j = 0$ ) or near perpendicular ( $\theta_j$  close to 0) can cause problems because of the curvature of the constraints. It would lead to a rapid decrease of the objective function, but with a feasible direction which closely follows the boundary of the feasible domain, hence the surface of the active constraints might be hit rapidly. For large  $\theta_j$ , the direction vector is less steep and there is no risk of running out the feasible domain, however the decrease of the objective function is much slower. Most of the times it is assumed that  $\theta_j = 1$ . Because  $\theta_j$  prevents the search direction of following to closely the surface of the constraints, they have been called push-off factors.

Having defined the direction vector, a next step is to determine the step size  $\alpha$  i.e. define how far one can move in the useable-feasible direction so



that the next point  $\mathbf{x}_{q+1}$  lays in the feasible region and reduces the objective. The step size is determined in different ways, depending on the following cases:

- $\mathbf{x}_{q+1}$  lies on the boundary of the feasible region, i.e. one or more constraints are active. To obtain this point, one wants to make an as large move as possible along the direction vector without violating the constraints. [66] To determine this, a trial step is taken, if the constraint is violated,  $\alpha$  is reduced and the results are checked again. If the trial step  $\alpha$  did not violate the constraint, then this step can be chosen as the result will be in the feasible region or a new and bigger step can be tried.
- $\mathbf{x}_{q+1}$  lies inside the feasible region, i.e. the point  $\mathbf{x}_{q+1}$  is an unconstrained minimum with respect to  $\alpha$ . The determination of the step size reduces now to a one-dimensional minimisation.

Figure 4.6 shows how the feasible directions method works.

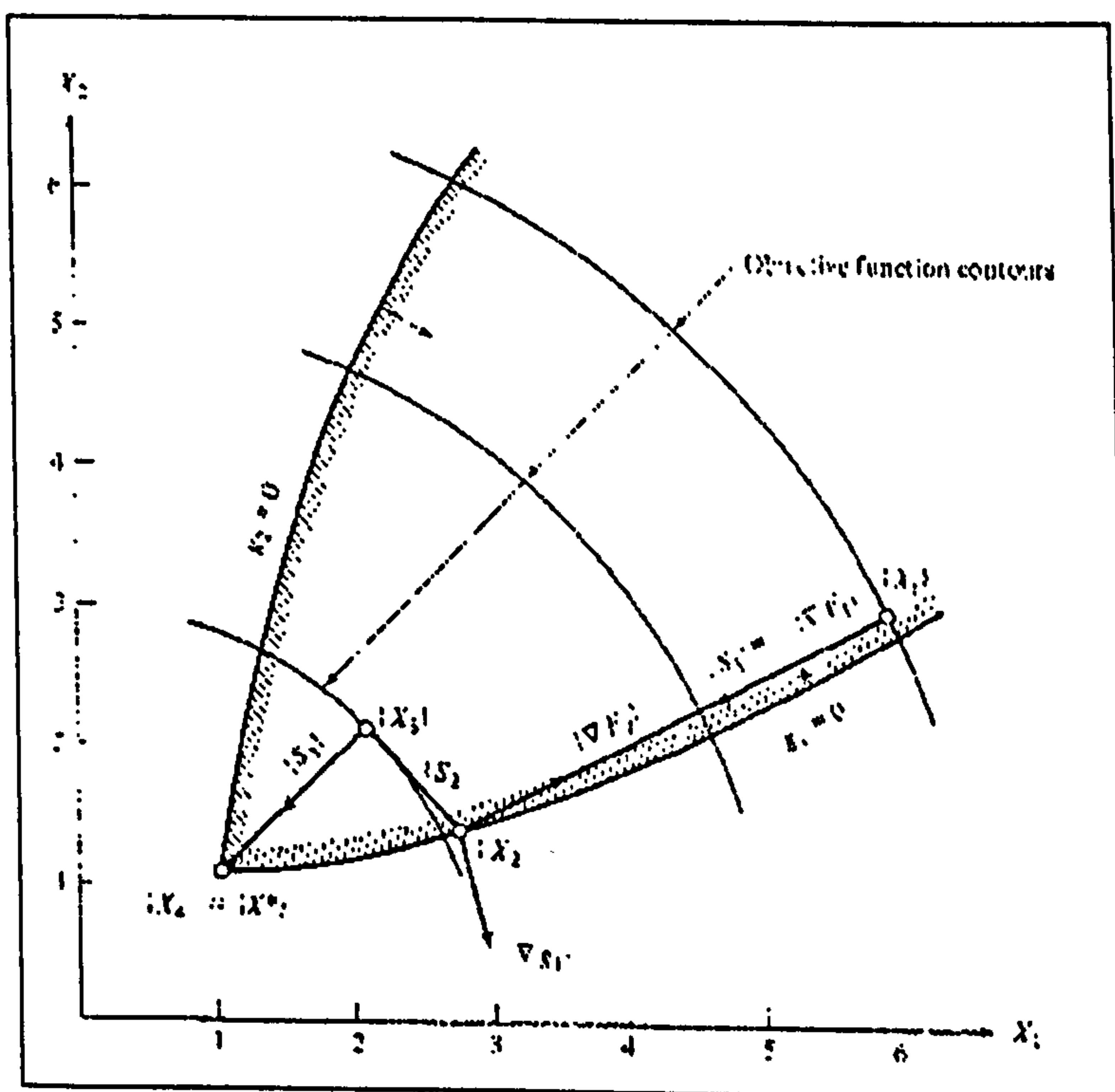


Figure 4.6: Method of feasible directions.

### 4.3.1.3 Method of Sequential Linear Programming

The method of Sequential Linear Programming reduces the non linear optimisation problem into a sequence of linear programming problems. For the linear programming both the objective and constraint functions are linearised. The linear approximation is done using Taylor series expansions (see equation 4.3 and reduces the non-linear optimisation problem to

$$\begin{aligned} \text{Minimise } W &= f(\mathbf{x}^*) + (\mathbf{x} - \mathbf{x}^*)^T \nabla f(\mathbf{x}^*) \\ \text{Subject to } g_j(\mathbf{x}^*) &+ (\mathbf{x} - \mathbf{x}^*)^T \nabla g_j(\mathbf{x}^*) \leq 0 \quad j = 1, \dots, n \end{aligned} \quad (4.20)$$

where  $\mathbf{x}^*$  is the point where the objective function and the constraints are linearised and  $j$  is the number of active constraints.

The method as shown in equation 4.20 has some strong limitations [66] [67]:

- The process will always converge to a vertex solution. If the minimum lies not in a vertex, then the problem converges to a non-optimal vertex or oscillates between two vertices.
- The design changes may become too large, thus invalidating the linear approximations.
- The problem can only provide a solution if the number of constraints is larger than the number of variables, if this is not the case, the problem may not have a bounded solution.

For these reasons, limits are imposed on the changes in the design. These constraints are called move limits (see figure 4.7).

The following additional constraint is added to equation 4.20:

$$\mathbf{x}_i^L \leq \mathbf{x}_i \leq \mathbf{x}_i^U \quad (4.21)$$

where  $\mathbf{x}_i^L = \mathbf{x}_i^* - \alpha_i$  and  $\mathbf{x}_i^U = \mathbf{x}_i^* + \beta_i$  with  $\alpha_i$  and  $\beta_i$  being vectors of properly chosen positive constants, i.e. move limits. Usually  $\mathbf{x}_i^L$  and  $\mathbf{x}_i^U$  are selected as some fraction of the current design variable values (this may vary from 1 to 100 %). Proper selection of move limits is of critical importance. The advantage of using linear approximation instead of the non-linear functions is that when the optimiser requires values of the objective and constraint functions that these are easily and inexpensively calculated from the linear approximation. Also, since the approximation problem is linear, the gradients of the objective and constraints are available directly from the Taylor series expansion. Within MSC/NASTRAN the method of feasible directions



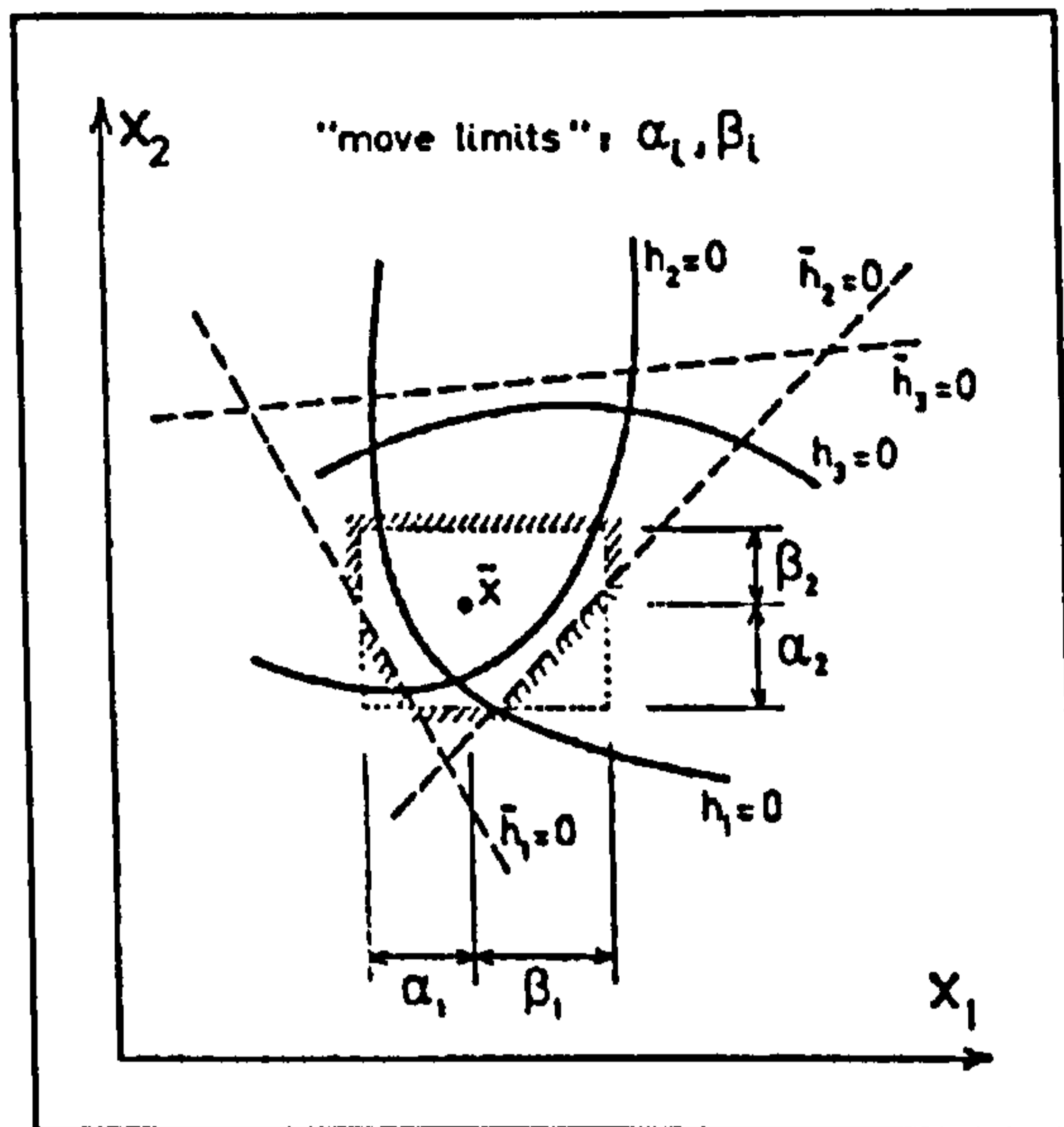


Figure 4.7: Move limits.

is used to solve this linearised optimisation problem. Typically, move limits allow the design variables to change by 20 to 40 %.

General comments on the sequential linear programming algorithm are [67]:

- The selection of the move limits is a trial and error process and can be best achieved in an interactive mode. Move limits can be too restrictive resulting in no solution or can slow down the rate of convergence.
- The rate of convergence depends largely on the move limit selection.
- The method can cycle between two points if the optimum solution does not lay in a vertex of the constraint set.

#### 4.3.1.4 Sequential Quadratic Programming

The basic concept is similar to the Sequential Linear Programming method (see section 4.3.1.3). The non-linear objective and constraint functions are approximated using Taylor Series (see equation 4.3 approximations. However, a quadratic approximation is used for the objective function and a linear approximation is used for the constraint functions. The problem is

now of the form

$$\begin{aligned}
 & \text{Minimise } W = f(\mathbf{x}^*) + \nabla f^T(\mathbf{x}^*)\Delta\mathbf{x} + \frac{1}{2}\Delta\mathbf{x}^T\mathbf{H}\Delta\mathbf{x} \\
 & \text{Subject to } g_j(\mathbf{x}^*) + (\mathbf{x} - \mathbf{x}^*)^T\nabla g_j(\mathbf{x}^*) \leq 0 \quad j = 1, \dots, n
 \end{aligned} \tag{4.22}$$

where  $g_j$  is the number of active constraints, and  $\mathbf{H}$  is the Hessian.

Within MSC/NASTRAN this Sequential Quadratic Programming problem is solved using the method of feasible directions. However, the Hessian matrix  $\mathbf{H}$  is not being calculated directly but is replaced by a positive definite matrix  $\mathbf{B}$ . Initially the identity matrix  $\mathbf{I}$  is taken for  $\mathbf{B}$  and in subsequent iterations  $\mathbf{B}$  is updated using an update formula. This formula is known as the BFGS (Broyden-Fletcher-Goldfarb-Shanno) formula and is shown below (equation 4.23). More information on this method can be found in [65] [67] [69] and a detailed description of the method can be found in [70].

$$\mathbf{B}^{(k+1)} = \mathbf{B}^{(k)} + \mathbf{D}^{(k)} + \mathbf{E}^{(k)} \tag{4.23}$$

Where  $\mathbf{B}^{(k+1)}$  is the new updated approximation for the Hessian. The correction matrices  $\mathbf{D}^{(k)}$  and  $\mathbf{E}^{(k)}$  are defined as

$$\begin{aligned}
 \mathbf{D}^{(k)} &= \frac{(\nabla f(\mathbf{x}^{k+1}) - \nabla f(\mathbf{x}^k))(\nabla f(\mathbf{x}^{k+1}) - \nabla f(\mathbf{x}^k))^T}{(\nabla f(\mathbf{x}^{k+1}) - \nabla f(\mathbf{x}^k))\alpha_k\mathbf{S}^k} \\
 \mathbf{E}^{(k)} &= \frac{\nabla f(\mathbf{x}^k)\nabla f(\mathbf{x}^k)^T}{\nabla f(\mathbf{x}^k)\mathbf{S}^k}
 \end{aligned}$$

where  $\alpha_k$  is the step size and  $\mathbf{S}^k$  is the search direction.

## 4.3.2 Results

### 4.3.2.1 Results reference model

Structural optimisation of the reference wing with MSC/NASTRAN gave an optimised weight of about 28000 kg. Table 4.2 shows the weight breakdown. In total 12 iterations were required .

An overview of the initial and optimised stresses is shown in figures 4.8 and 4.9. It can be seen that after the optimisation process most of the  $\sigma_x$  stresses are up to the allowable limits which were specified in figure 4.2 on page 73. In critical areas, where the stresses initially exceeded the limit stress, a stress reduction took place.

Figure 4.10 shows the changes in skin thickness before and after optimisation. The upper figures show the initial model with the fifteen spanwise



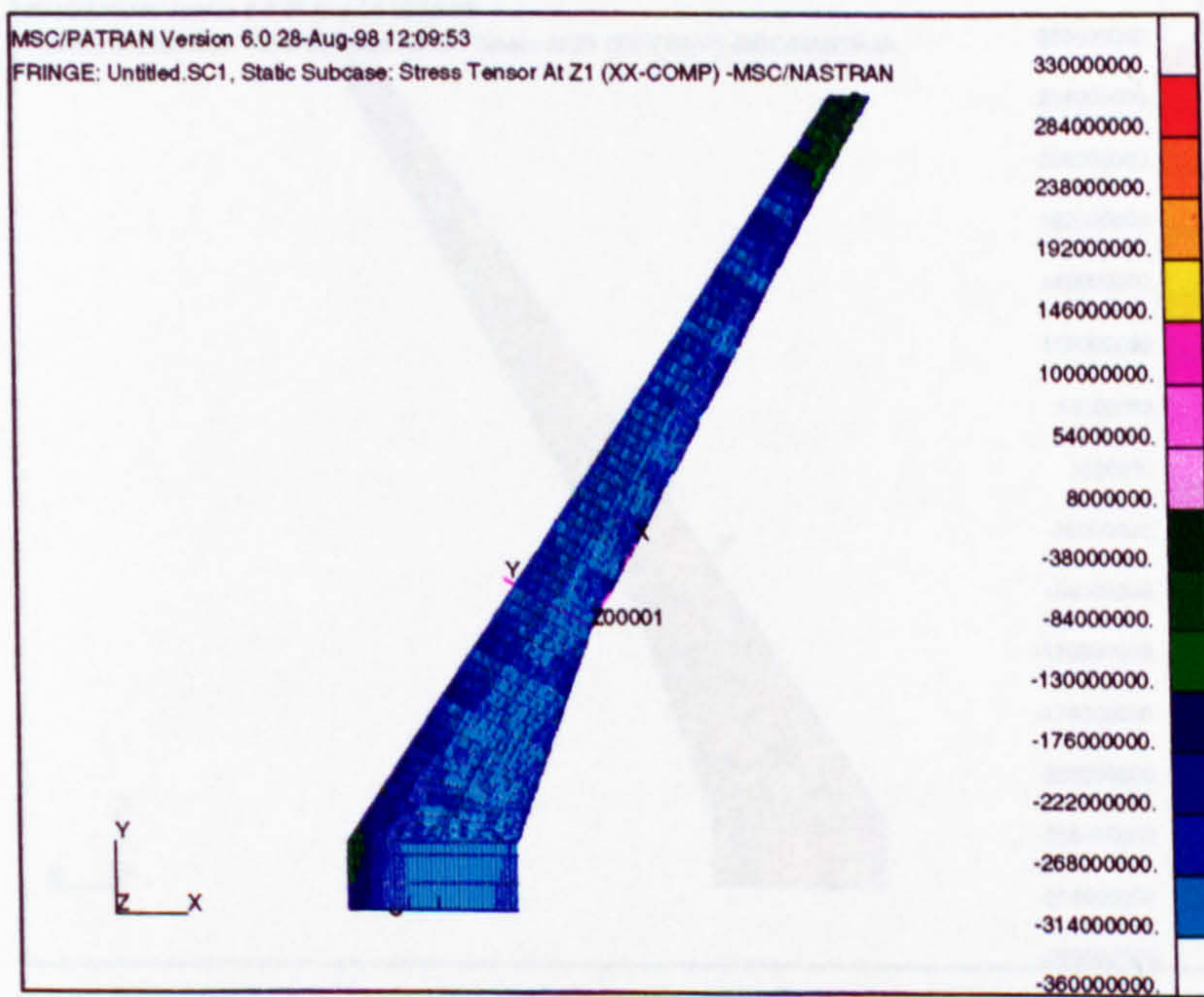
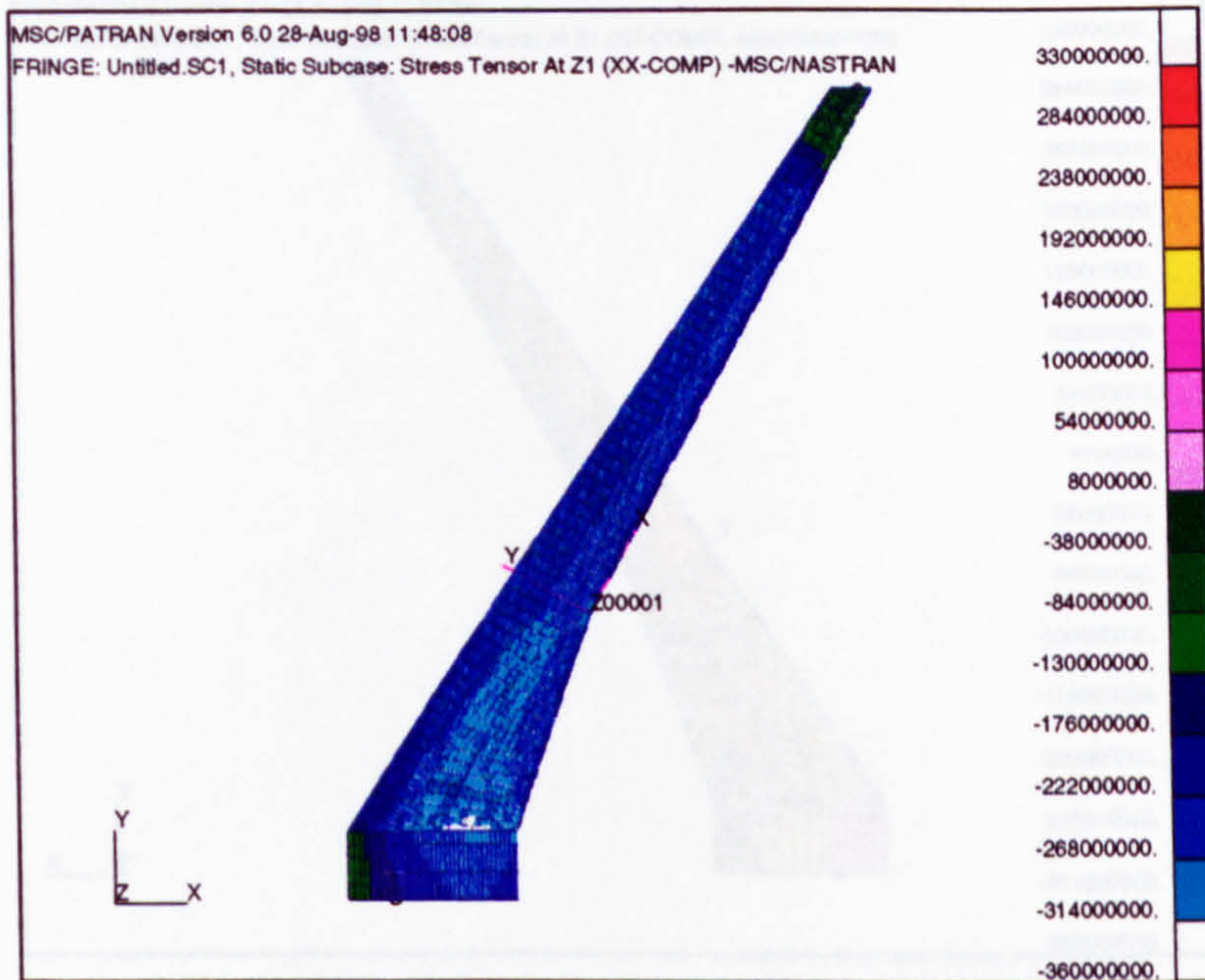


Figure 4.8: Initial and optimised stresses ( $\sigma_x$ ) in top skin, pull-up load case.



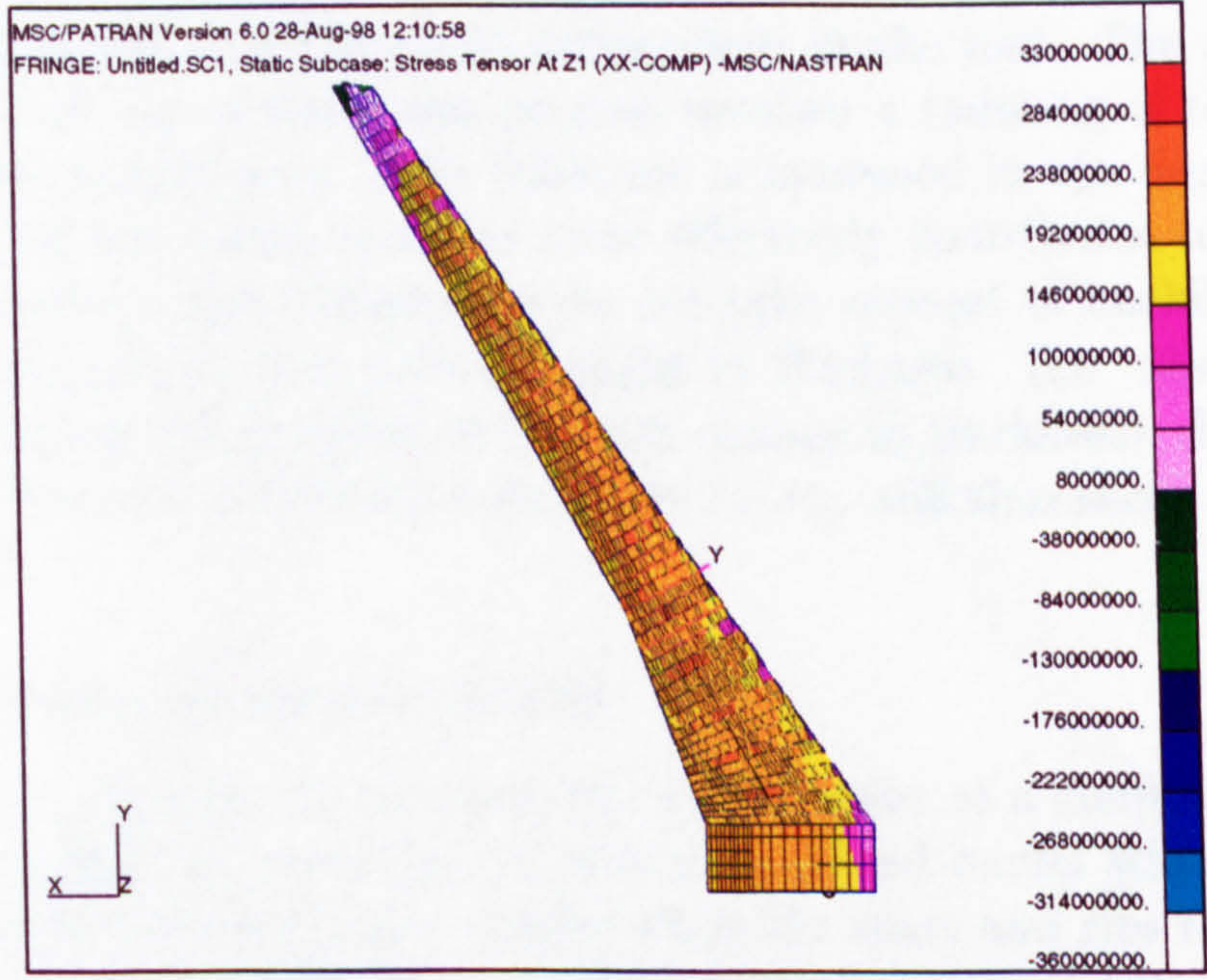
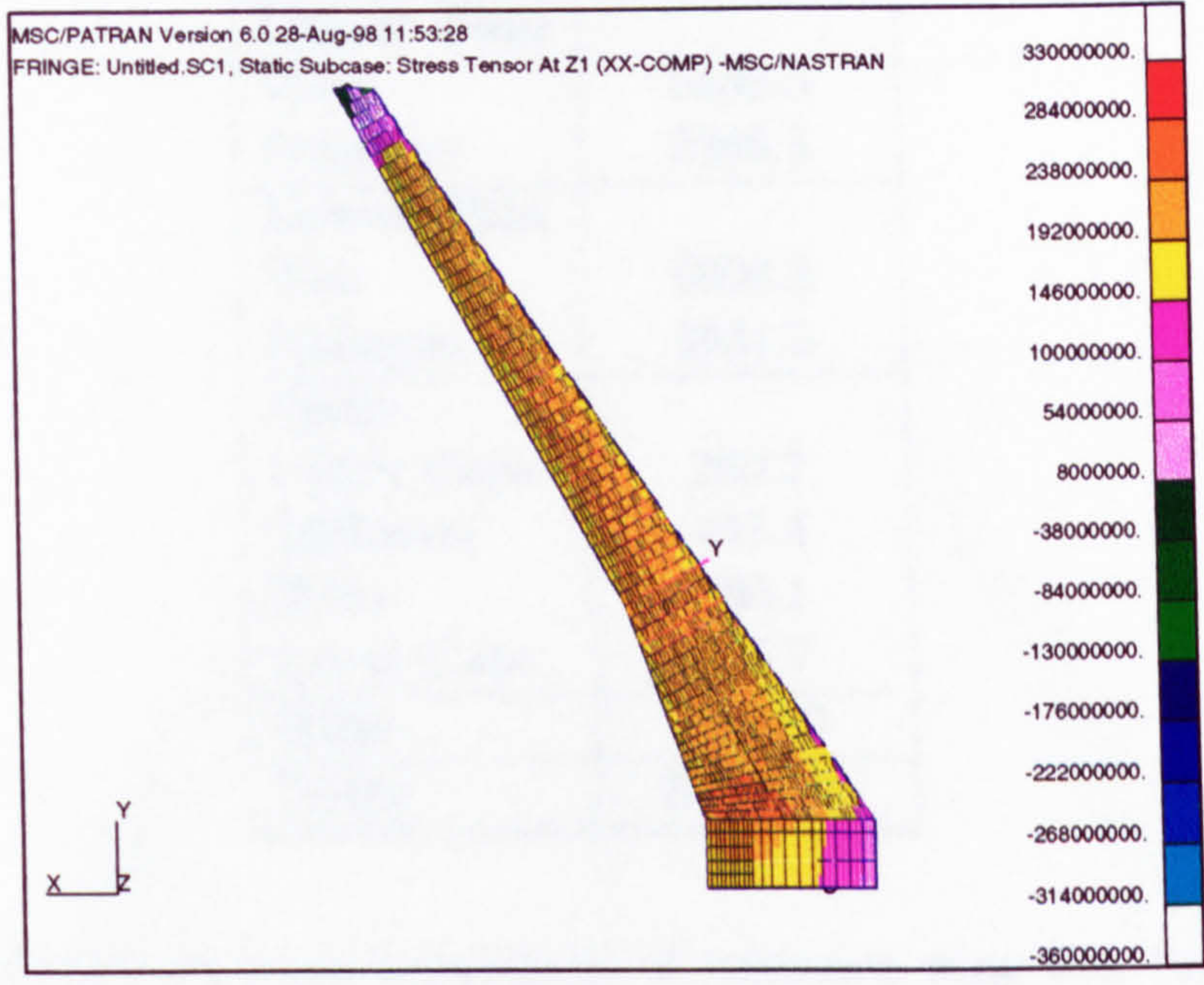


Figure 4.9: Initial and optimised stresses ( $\sigma_x$ ) in bottom skin, pull-up load case.



| Component         | Weight (kg)    |
|-------------------|----------------|
| <b>Upper Skin</b> |                |
| Web               | 5432.5         |
| Stringers         | 2366.5         |
| <b>Lower Skin</b> |                |
| Web               | 6908.2         |
| Stringers         | 3031.5         |
| <b>Spars</b>      |                |
| Upper Caps        | 280.2          |
| Stiffeners        | 445.4          |
| Webs              | 890.1          |
| Lower Caps        | 346.7          |
| <b>Ribs</b>       | 8300.9         |
| <b>Total</b>      | <b>28002.0</b> |

Table 4.2: Optimised mass breakdown of reference wing box (units in kg) using NASTRAN.

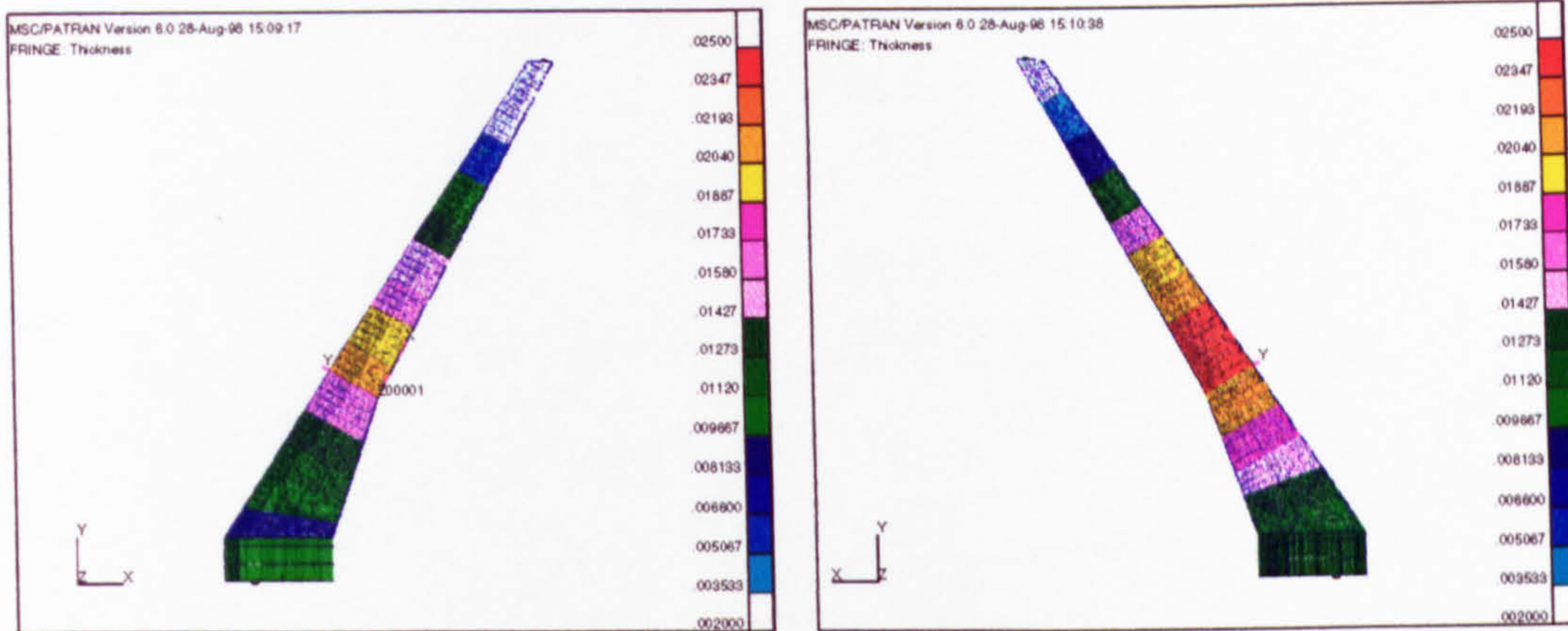
zones and constant distribution of thickness chordwise over each section. Because of the rapid tapering of the wing box depth over the inboard wing, the maximum thickness is at the crank rather than at the root. The optimised wing shows that the optimisation process involves a tailoring of the chordwise thickness distribution. Skin thickness is increased in the deeper parts of the wing section where material most effectively contributes to bending stiffness. However, these thicknesses do not take account of buckling as the buckling allowables do not reflect changes in thickness. (i.e. the buckling allowables are not being updated for each change in thickness). It also has to be noted that the constraints only apply to  $\sigma_{xx}$  and thus shear effects are not included.

#### 4.3.2.2 Results composite model

Section 3.4.1.1 on page 41 reported the development of a combined metal/composite model. As mentioned it was a simplified model which only assumed composite skin&stringer panels, while the spars and ribs remained in metal. Also, a 'black aluminium' approach was used to model the composite ply lay-up. Table 4.3 shows the optimised weight breakdown obtained by MSC/NASTRAN after 12 iterations. The wing model assumed an outboard composite wing with a joint located at rib 36. The design variable definition



Initial thickness distribution top and bottom skin.



Optimised thickness distribution top and bottom skin.

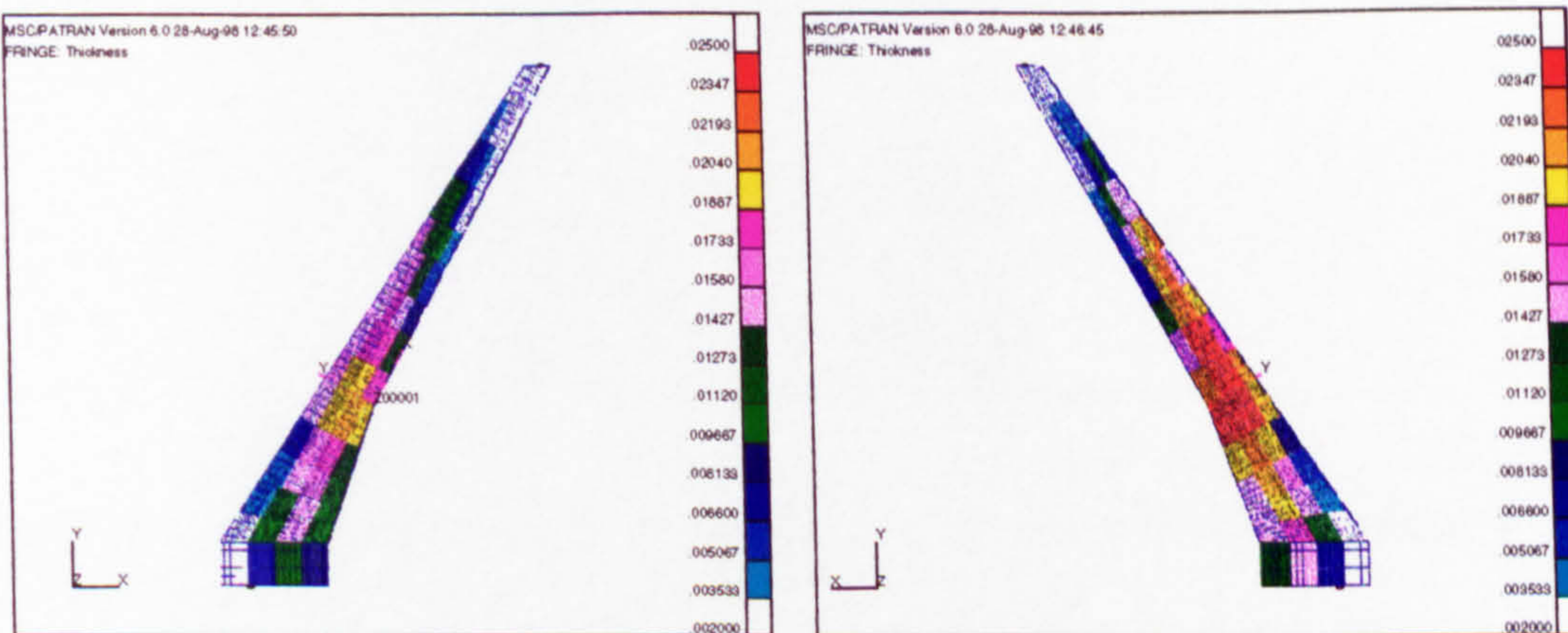


Figure 4.10: Thickness distribution (scaled values), pull-up load case.



was the same as for the metal wing, except for the spar caps which were treated as an integral part of the skin&stringer panel. An additional weight reduction of about 440 kg (i.e. 1.6 % reduction) was obtained in comparison to the full metal configuration.

| <b>Metal wing</b>     |                    |
|-----------------------|--------------------|
| <b>Component</b>      | <b>Weight (kg)</b> |
| <b>Upper Skin</b>     |                    |
| Web                   | 4136.7             |
| Stringers             | 2012.6             |
| <b>Lower Skin</b>     |                    |
| Web                   | 5229.7             |
| Stringers             | 2531.9             |
| <b>Spars</b>          |                    |
| Stiffeners            | 366.7              |
| Webs                  | 675                |
| <b>Ribs</b>           | 7016               |
| <b>Sub Total</b>      | <b>21968.6</b>     |
| <b>Composite wing</b> |                    |
| <b>Component</b>      | <b>Weight (kg)</b> |
| <b>Upper Skin</b>     |                    |
| Web                   | 1507.6             |
| Stringers             | 729.4              |
| <b>Lower Skin</b>     |                    |
| Web                   | 1178.2             |
| Stringers             | 597.3              |
| <b>Spars</b>          |                    |
| Stiffeners            | 78.7               |
| Webs                  | 219.3              |
| <b>Ribs</b>           | 1284.9             |
| <b>Sub Total</b>      | <b>5595.4</b>      |
| <b>Total</b>          | <b>27564.0</b>     |

Table 4.3: Optimised weight breakdown combined metal/composite wing, joint at rib 36.



## 4.4 Optimisation using STARS

### 4.4.1 Optimisation Method

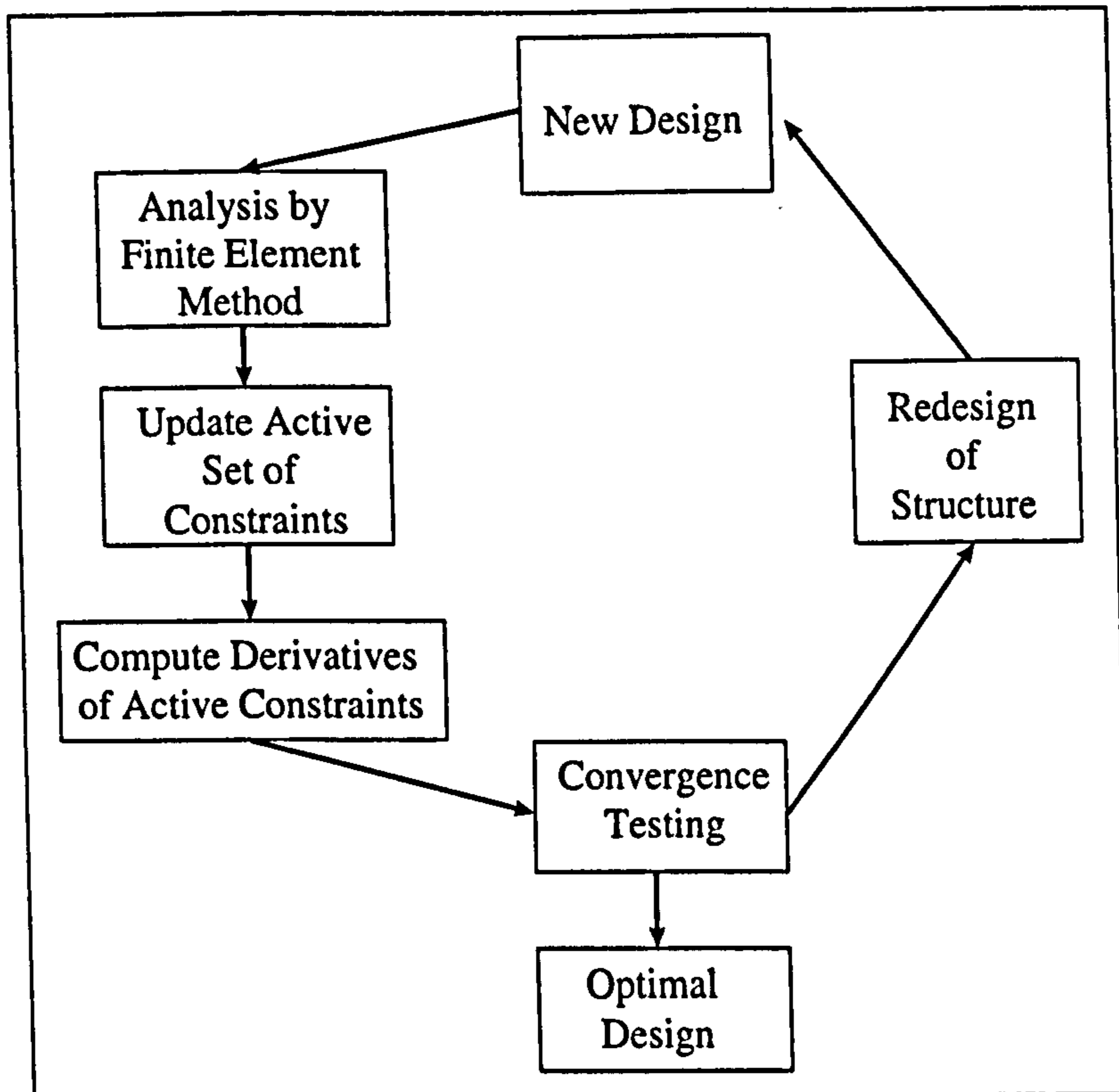


Figure 4.11: Stars process flow.

The Structural Analysis And Redesign System (STARS) has been developed by DERA. Figure 4.11 shows how the STARS design system works. For the finite element analysis, STARS can use the DERA ANALYSIS tool or external FE packages such as MSC/NASTRAN or UAI/NASTRAN. As can be seen from the figure, an active set module can be used to identify which constraints are active during a particular iteration such as procedure is called active set strategy (see section 4.6.3 on page 110. The STARS optimisation process is written in a modular way and the user can decide to add or remove modules such as update of active set constraints, estimation of lower bound to the optimum weight (dual) etc., depending on the optimisation algorithm applied and the optimisation problem. The STARS Fully Stressing algorithm and the QNewton algorithm have been used on the MDO wing.

#### 4.4.1.1 Fully Stressing

Fully Stressing is a stress ratio method which sizes each component of the structure to its stress limits. The fully stressing algorithm seeks an optimum on the basis of constraint satisfaction. Using the responses from finite element analysis, the stresses and displacements for the structure are evaluated and compared with allowable limits. With these limits, the structure is then scaled to a fully stressed state. This scaled structure is then analysed in the next iteration. In this method the number of active constraints is equal to the number of design variables and thus a specific constraint  $\sigma_j(x)$  is associated with each design variable  $x_j$ . The scale factor for a particular design variable is calculated as [71]:

$$x = x_0 \left( \frac{\sigma(x_0)}{\bar{\sigma}} \right)^{\frac{1}{b}} \quad (4.24)$$

with

$$b = \left( \frac{z}{\sigma} \right) \left( \frac{\delta \sigma}{\delta z} \right) \quad (4.25)$$

where  $x_0$  is the current value of the design variable,  $x$  is the new value of the design variable,  $\bar{\sigma}$  is the allowable stress and  $z$  is the inverse of the design variable.

From the above, it is clear that no attempt is made to specifically consider the weight of the structure during the fully stressing process. A single step of this method is economical on computer time as it requires no derivative calculation. Hence, this method is very valuable to make rapid, stable progress in the early stages of the optimisation run for a strength critical design. Fully stressing does not necessarily give a minimum weight design, particularly when the allowable stress in all the elements of the structure is not the same. It can also give a design with a bad distribution of material and an inefficient load path. Fully Stressing also gives a vertex solution. [66] For these reasons, the user may consider to switch to a more sophisticated optimisation algorithm at a later stage in the redesign process.

#### 4.4.1.2 QNewton

The Newton method uses a second-order Taylor series expansion (see equation 4.1) to approximate the objective function about the current design point. Hence, not only first-order derivative information, but also second-order derivative information is used in the approximation of the cost function.

For unconstrained optimisation the Newton method can be described as follows:



$$\text{Minimise} \quad f(\mathbf{x}_k + \Delta\mathbf{x}) = f(\mathbf{x}_k) + \nabla f^T(\mathbf{x}_k)\Delta\mathbf{x} + \frac{1}{2}\Delta\mathbf{x}^T\mathbf{H}\Delta\mathbf{x} \quad (4.26)$$

Where  $f(\mathbf{x}_k)$ ,  $\nabla f^T(\mathbf{x})\Delta\mathbf{x}$  and  $\mathbf{H}$  are the objective function, the gradient vector and the Hessian matrix at point  $\mathbf{x}_k$ . At the optimum, the condition  $\frac{\partial f}{\partial \Delta\mathbf{x}} = 0$  has to be satisfied. Applying this optimality condition to equation 4.26 gives

$$\nabla f(\mathbf{x}_k) + \mathbf{H}\Delta\mathbf{x} = 0 \quad (4.27)$$

This can be rewritten in the form

$$\Delta\mathbf{x} = -\mathbf{H}^{-1}\nabla f(\mathbf{x}_k) \quad (4.28)$$

With  $\mathbf{x} = \Delta\mathbf{x} + \mathbf{x}_k$  and  $k = 0, 1, 2, \dots$  the new estimate for the design is given as

$$\mathbf{x} = \mathbf{x}_k - \mathbf{H}^{-1}\nabla f(\mathbf{x}_k) \quad (4.29)$$

Since equation 4.26 is just an approximation for the function  $f$  at the point  $\mathbf{x}_k$ ,  $\mathbf{x}$  will probably not be the precise minimum point of  $f(\mathbf{x})$  and several iterations will be required to obtain improved estimates until the minimum is reached. The iterative numerical procedure which tries to improve the estimate of  $f(\mathbf{x})$  by calculating  $\Delta\mathbf{x}$  is the Newton-Raphson method.  $\Delta\mathbf{x}$  can be defined as the direction vector. In order to improve the convergence and stability of the method, a step length (or step size) parameter  $\alpha_k$  is added which is associated with the direction vector. As mentioned earlier in section 4.3.1.2 on page 79 a one-dimensional search can be used to find an optimum step length which minimises the cost function in the direction  $\Delta\mathbf{x}$ . At each iteration of equation 4.29 a computation of the inverse Hessian is required. Calculating second derivatives is already often difficult or impossible, hence the computation of the inverse Hessian for large problems becomes impractical. For this reason, the calculation of the Hessian is often replaced by an approximated Hessian which has been calculated using first order derivatives (see BVGS method in section 4.3.1.4 on page 84). The methods which employ an approximated Hessian for the Newton-Raphson method are called Quasi-Newton (QNewton) methods. Other names for this method are Constrained Variable Metric (CVM), Sequential Quadratic Programming (SQP) or Recursive Quadratic Programming (RQP).

In case of constrained optimisation we now have the following problem:

$$\begin{aligned} & \text{Minimise } f(\mathbf{x}) \\ & \text{Subject to } g_i(\mathbf{x}) = 0 \quad i = 1, \dots, p \end{aligned} \quad (4.30)$$

For simplicity we only assume equality constraints. The Lagrangian for equation 4.30 can be written as

$$L(\mathbf{x}, \lambda) = f(\mathbf{x}) - \sum_{i=1}^p \lambda_i g_i(\mathbf{x}) = f(\mathbf{x}) - \lambda \mathbf{g}(\mathbf{x}) \quad (4.31)$$

where  $\lambda_i$  is the Lagrange multiplier for the  $i$ th equality constraint  $g_i(\mathbf{x})$ . The Kuhn-Tucker necessary conditions for an optimum point give

$$\nabla L(\mathbf{x}, \lambda) = 0 \quad \text{i.e. } \nabla f(\mathbf{x}) + \lambda \nabla \mathbf{g}(\mathbf{x}) = 0 \quad (4.32)$$

$$\mathbf{g}(\mathbf{x}) = 0 \quad (4.33)$$

The conditions stated in 4.32 and 4.33 form a set of non-linear equations. As the dimension of the design variable vector  $\mathbf{x}$  is  $n$  and as there are  $p$  inequality constraints, there are  $(n + p)$  equations in  $(n + p)$  unknowns. Using the Newton-Raphson method, these equations can be solved by using a first order Taylor series expansion for them. Equations 4.32 and 4.33 can be written in a compact notation as

$$\mathbf{F}(\mathbf{y}) = 0 \quad (4.34)$$

where  $\mathbf{F}$  and  $\mathbf{y}$  are identified as

$$\mathbf{F} = \begin{bmatrix} \nabla L \\ \mathbf{g} \end{bmatrix} \quad \text{and} \quad \mathbf{y} = \begin{bmatrix} \mathbf{x} \\ \lambda \end{bmatrix} \quad (4.35)$$

Assuming that  $\mathbf{y}^{(k)}$  is known at the  $k$ th iteration, we apply the iterative Newton-Raphson method so that  $\mathbf{y}^{(k+1)} = \mathbf{y}^{(k)} + \Delta \mathbf{y}^{(k)}$ . The change in  $\Delta \mathbf{y}^{(k)}$  is obtained after applying Taylor series expansion for the set of non-linear equations, defined in equation 4.34

$$\mathbf{F}(\mathbf{y}^{(k)}) + \nabla \mathbf{F}^T(\mathbf{y}^{(k)}) \Delta \mathbf{y}^{(k)} = 0 \quad (4.36)$$

with  $\nabla \mathbf{F}$  being an  $(n + p) \times (n + p)$  Jacobian matrix containing the gradient of the function  $F_i(\mathbf{y})$  with respect to the vector  $\mathbf{y}$ .

Substituting the definitions of  $\mathbf{F}$  and  $\mathbf{y}$  from equation 4.35 in equation 4.36 and reorganising 4.36 gives



$$\begin{bmatrix} \nabla^2 L & \mathbf{N} \\ \mathbf{N}^T & 0 \end{bmatrix}^{(k)} \begin{bmatrix} \Delta \mathbf{x} \\ \Delta \lambda \end{bmatrix}^{(k)} = - \begin{bmatrix} \nabla L \\ \mathbf{g} \end{bmatrix}^{(k)} \quad (4.37)$$

With  $\nabla^2 L$  being the  $n \times n$  Hessian matrix of the Lagrangian function,  $\mathbf{N}$  being an  $n \times p$  matrix whose  $i$ th column is the gradient of the equality constraint  $g_i$ ,  $\Delta \mathbf{x}^k = \mathbf{x}^{k+1} - \mathbf{x}^k$  and  $\Delta \lambda^k = \lambda^{k+1} - \lambda^k$ .

Equation 4.37 can be written in a different way by manipulating the equation of the first row. Substitution of  $\Delta \lambda^k$  by  $\lambda^{k+1} - \lambda^k$  in the first row and simplifying the first row equation leads to the following form

$$\begin{bmatrix} \nabla^2 L & \mathbf{N} \\ \mathbf{N}^T & 0 \end{bmatrix}^{(k)} \begin{bmatrix} \Delta \mathbf{x} \\ \lambda^{(k+1)} \end{bmatrix}^{(k)} = - \begin{bmatrix} \nabla L \\ \mathbf{g} \end{bmatrix}^{(k)} \quad (4.38)$$

Solving equation 4.38 gives a change in the design  $\Delta \mathbf{x}^{(k)}$  and a new value for the Lagrange multiplier vector  $\lambda^{(k+1)}$  and this iterative procedure is continued until the optimum is reached.

The value for  $\Delta \mathbf{x}^{(k)}$  can also be found by solving the following quadratic problem at the  $k$ th iteration

$$\text{Minimise } \nabla L^T \Delta \mathbf{x} + \frac{1}{2} \Delta \mathbf{x}^T \nabla^2 L \Delta \mathbf{x} \quad (4.39)$$

$$\text{Subject to } \mathbf{g}(\mathbf{x}) + \mathbf{N}^T \Delta \mathbf{x} = 0 \quad (4.40)$$

With equation 4.40 representing the linear approximations of the equality constraints. Writing the Lagrangian and Kuhn-Tucker conditions for equation 4.39 and treating  $\Delta \mathbf{x}$  as unknown variable gives

$$\bar{L} = \nabla L^T \Delta \mathbf{x} + \frac{1}{2} \Delta \mathbf{x}^T \nabla^2 L \Delta \mathbf{x} + \lambda (\mathbf{g}(\mathbf{x}) + \mathbf{N}^T \Delta \mathbf{x}) \quad (4.41)$$

Kuhn-Tucker conditions:

$$\nabla \bar{L} = \nabla L^T + \nabla^2 L \Delta \mathbf{x} + \mathbf{N} \lambda = 0 \quad (4.42)$$

$$\mathbf{g}(\mathbf{x}) + \mathbf{N}^T \Delta \mathbf{x} = 0 \quad (4.43)$$

If one writes equations 4.42 and 4.43 in matrix form, the formulation of equation 4.38 is obtained. Thus the problem of minimising  $f(\mathbf{x})$  subject to  $\mathbf{g}(\mathbf{x}) = 0$  can be solved iteratively solving the quadratic problem of equation 4.39. Basically the objective function has now become a second-order

approximation of the Lagrangian (based on Taylor series expansions) calculated with the active constraints and where the constraints are linearly approximated using Taylor series expansion. [66] [67] The second derivative of the Lagrangian is often again obtained from the BFGS formula (see section 4.3.1.4 on page 84) .

## 4.4.2 Results

### 4.4.3 Datum results

The STARS results were obtained using four iterations of fully stressing, followed by nine QNewton iterations. Figures 4.12 and 4.13 show the optimisation history of the actual, feasible and dual weight. The feasible weight, which can be seen in the design cycle history, is a factored value of the actual weight. At any given point in the optimisation, the current (= actual) design may or may not be feasible. If it is not feasible, a scale factor is computed which, when applied to all the design variables, will give a structure that is just feasible. Hence, the feasible weight is the weight at which no constraint violation occurs. However, when a big part of the structure is not being optimised (= fixed structure), the calculation of the scale factor might cause problems. [71] As the ribs are not being optimised and account for a large part of the wing structure, a lot of fixed structure is present in the design model. Figures 4.12 and 4.13 show that there are sometimes big differences between the feasible weight calculations from iteration six to thirteen due to the large amount of fixed structure. At the first iteration, some constraints are heavily violated and this explains the high initial feasible weight. Up to iteration four, fully stressing reduces the feasible weight. From the figures it is clear that the fully stressing algorithm is very useful to make rapid progress in the early design stages and it is able to give a design which is close to the minimum solution without much computational expense. The drawback is that fully stressing can give a design with a bad distribution of material and an inefficient load path as it does not take into account derivative information. Hence the sudden increase in feasible weight after iteration four, when STARS switches to QNewton and calculates the derivatives. At this moment the dual (see section 4.6.6 on page 114) also appears. It can be seen that the dual acts as a lower bound to the design.

Optimising the reference wing using STARS gives a slight difference in optimised weight, in comparison to the MSC/NASTRAN results. Table 4.4 gives an overview of the weight results. Comparing these results with the weight breakdown obtained using MSC/NASTRAN (see table 4.2 on page 88) the STARS results indicate a minimum weight which is about 130 kg less



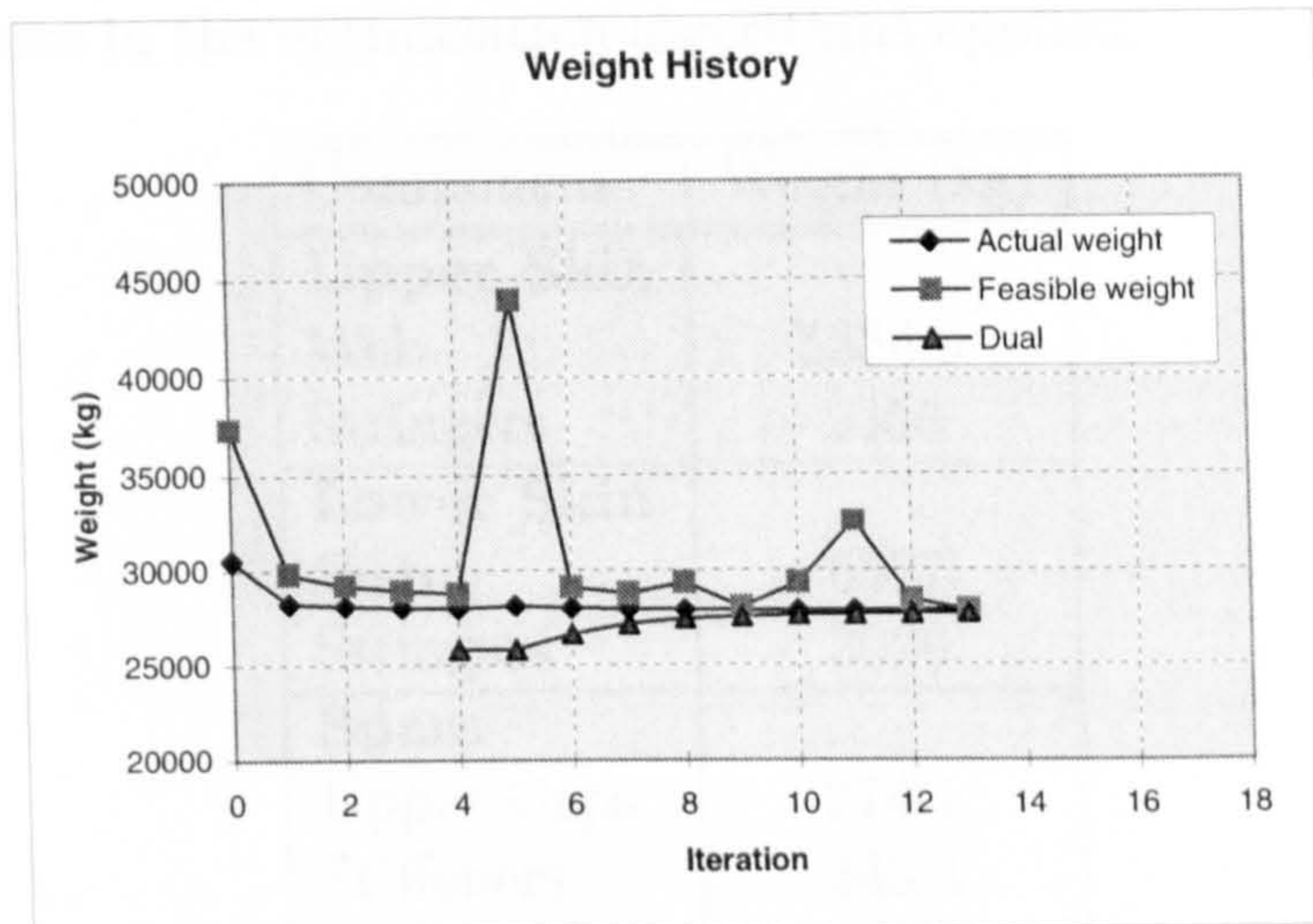


Figure 4.12: STARS optimisation history

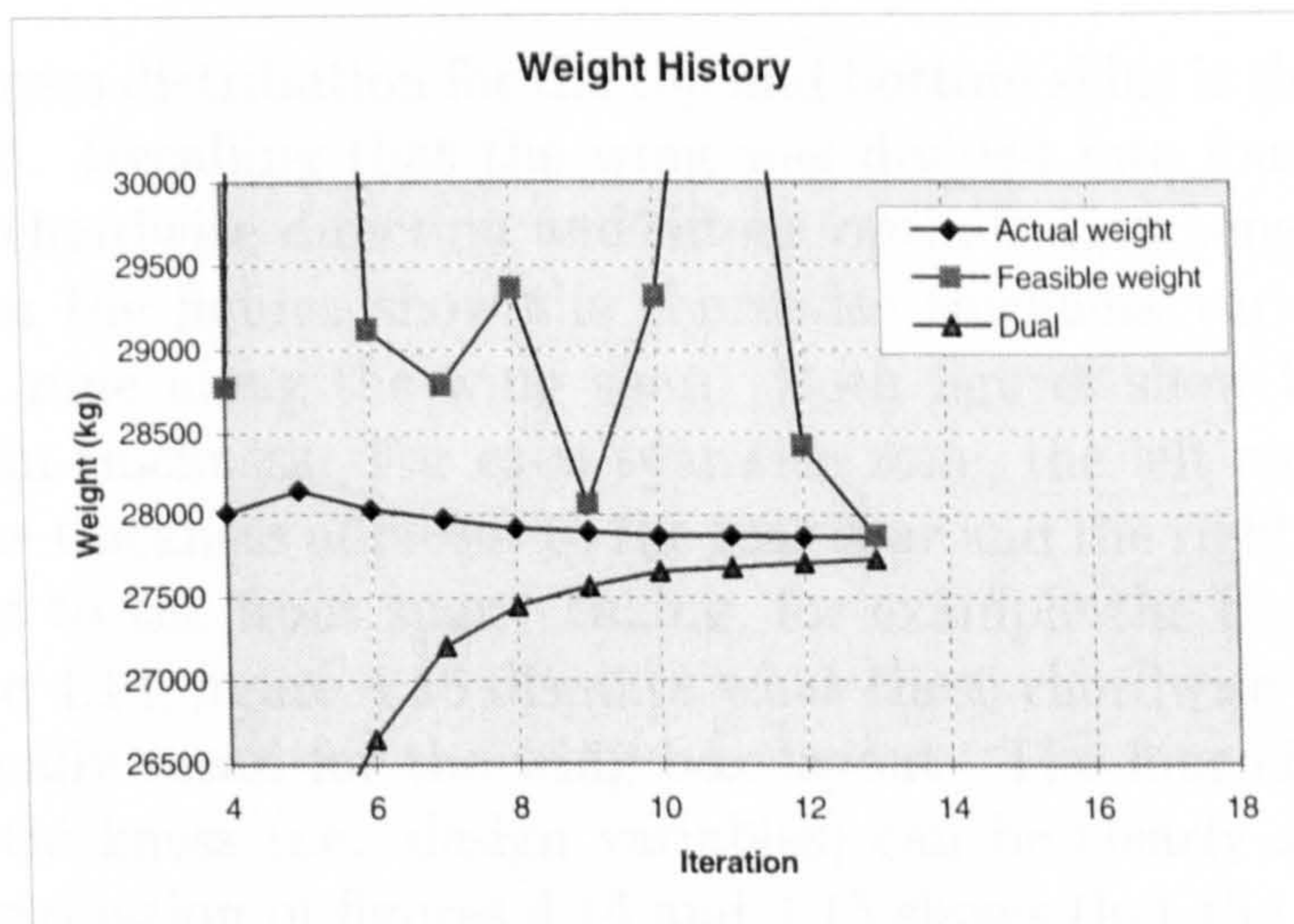


Figure 4.13: Magnified view STARS optimisation history, QNewton iterations

than that calculated by NASTRAN. STARS indicates a lower weight for each of the wing components. The difference in results can be explained because of the difference in the optimisation algorithms applied.

| Component         | Weight (kg)    |
|-------------------|----------------|
| <b>Upper Skin</b> |                |
| Web               | 5384.9         |
| Stringers         | 2358           |
| <b>Lower Skin</b> |                |
| Web               | 6887           |
| Stringers         | 3026           |
| <b>Spars</b>      |                |
| Upper Caps        | 274.1          |
| Stiffeners        | 445.4          |
| Webs              | 861.6          |
| Lower Caps        | 334.8          |
| <b>Ribs</b>       | 8300.9         |
| <b>Total</b>      | <b>27872.8</b> |

Table 4.4: Optimised mass breakdown of reference wing box (units in kg) using STARS.

The thickness distribution for the top and bottom skins is shown in figures 4.14 and 4.15. Recalling that the wing was divided into four optimisation zones in the chordwise direction and fifteen optimisation zones in the spanwise direction the figures show the chordwise thickness variation for each optimisation zone along the wing span. Both figures show the initial and optimised skin thickness. For each spanwise zone, the left end of the curve represents the thickness adjacent to the rear spar and the right end the thickness adjacent to the front spar. Taking, for example the first optimisation zone of figure 4.14, figure 4.16 displays what these chordwise thickness variations physically mean for the wing box layout. The four chordwise zones of constant thickness (i.e. design variables) can be clearly seen. The skin thickness distribution of figures 4.14 and 4.15 shows that the highest loaded area is around the crank of the wing (zones 6 and 7) rather than at the wing root. As mentioned before, this is caused because of the rapid tapering in wing box depth and the presence of the inboard engine. The tailoring of the chordwise thickness distribution to increase the efficiency in bending which was shown in figure 4.10 (see page 89) can also be observed.

Figure 4.17 shows the thickness distribution for the rear, centre and front



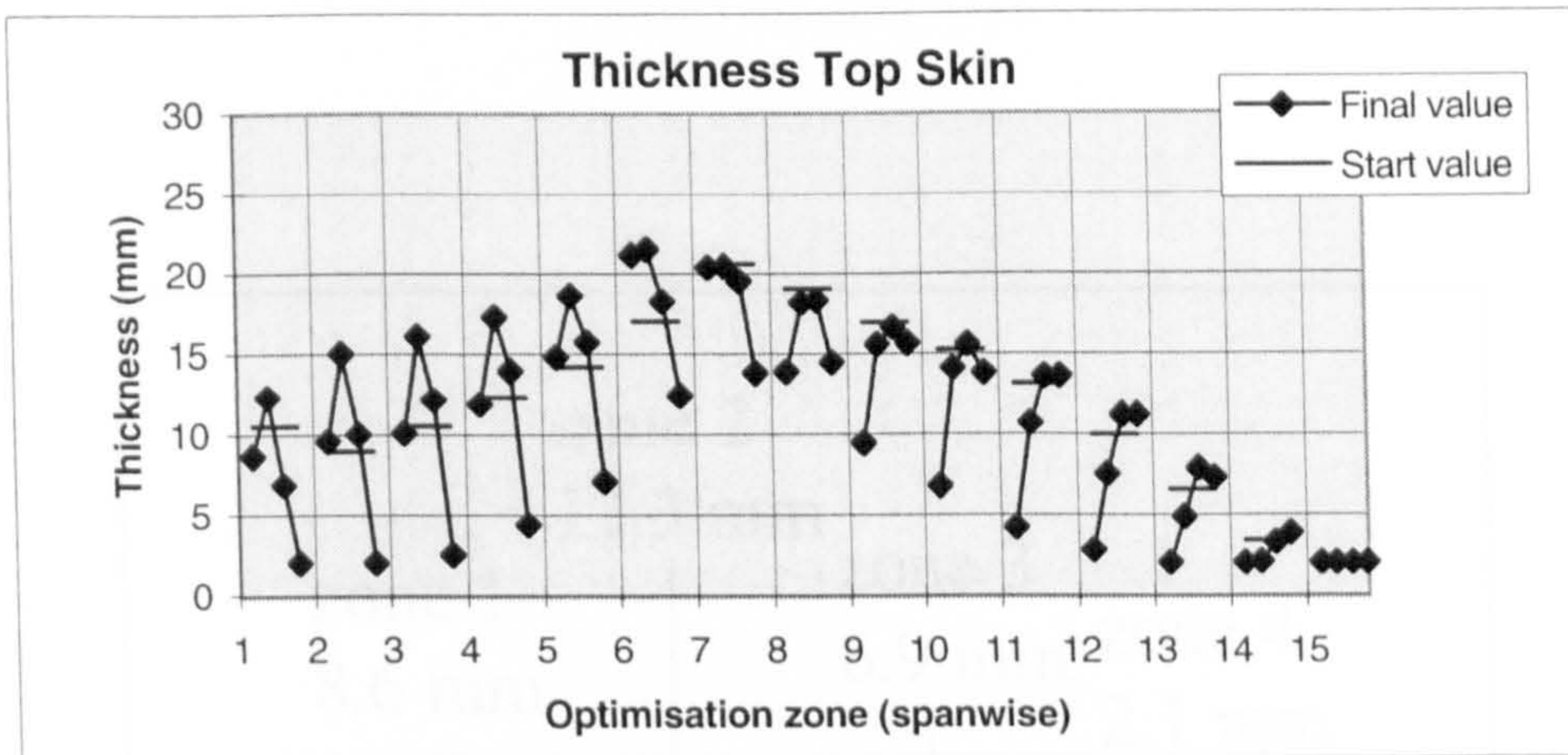


Figure 4.14: Thickness distribution top skin.

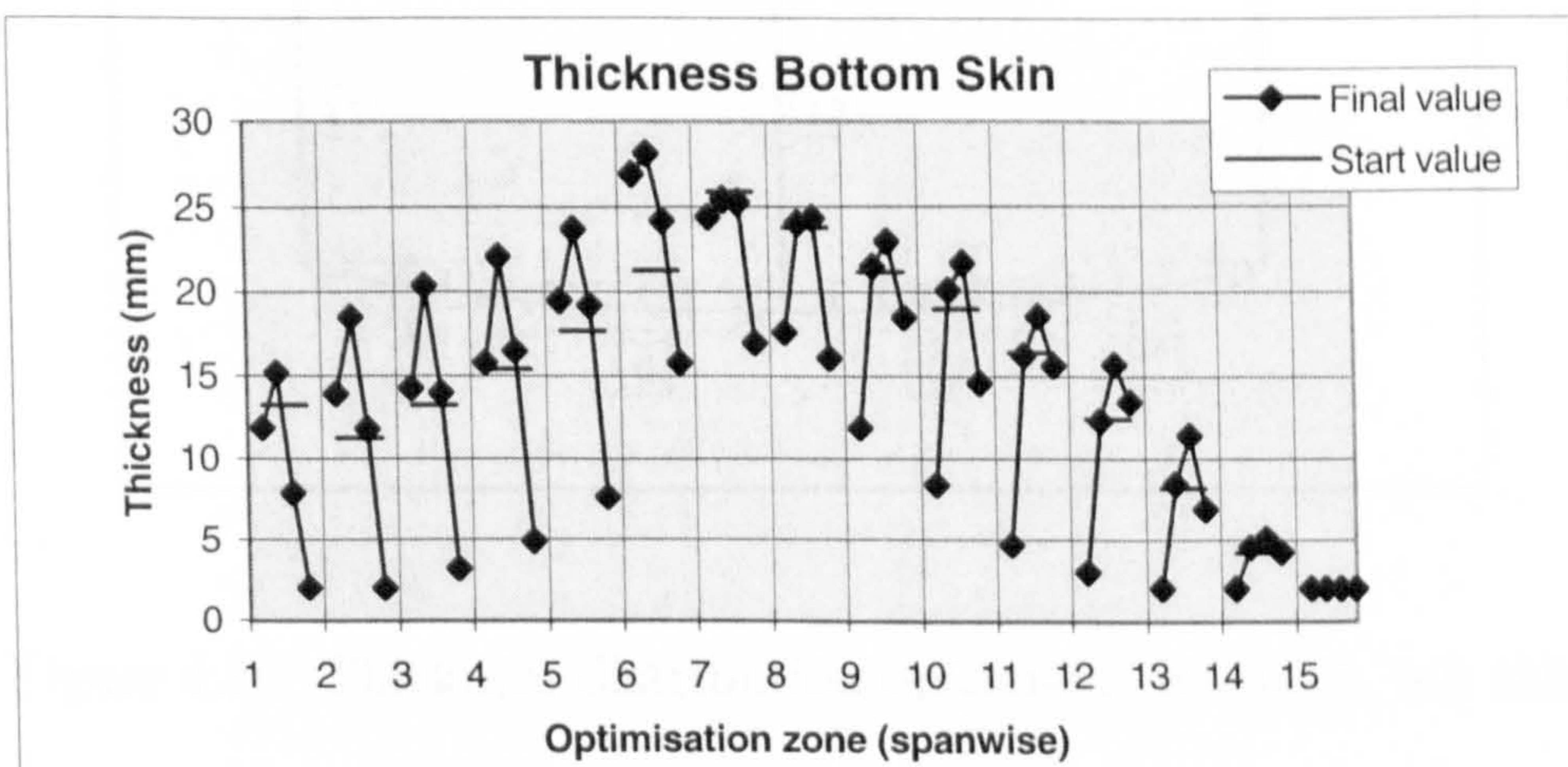


Figure 4.15: Thickness distribution bottom skin.

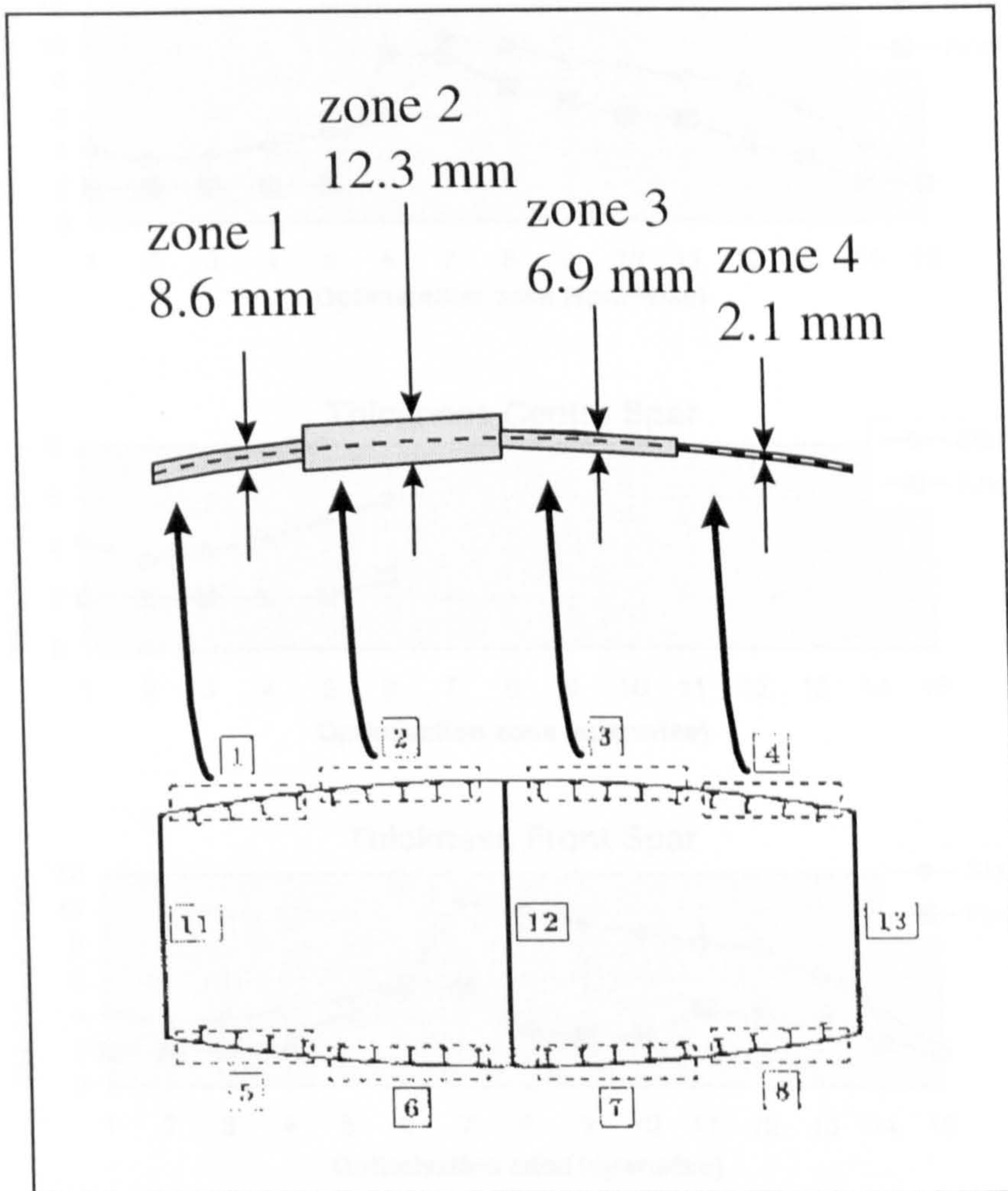


Figure 4.16: Thickness distribution optimisation zone 1, top skin.



spar webs. For many zones, the thickness is close to the minimum allowable thickness of 2 mm. Higher thickness is observed at the zones six and seven (i.e. at crank and inboard engine) and at zone eleven (i.e. at the outboard engine).

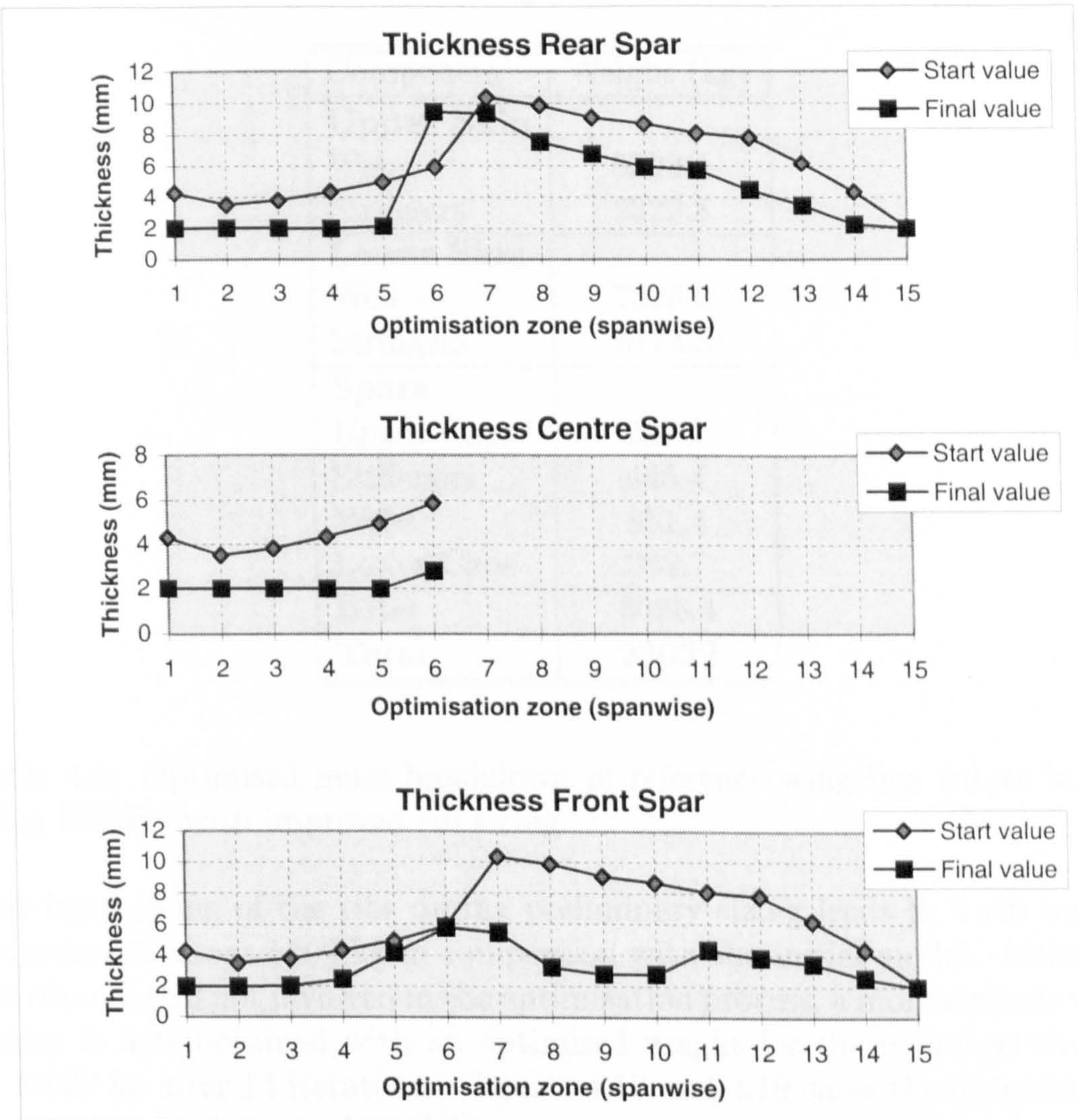


Figure 4.17: Thickness distribution spar webs.

#### 4.4.4 Results using improved preliminary sizing

Using the improved preliminary sizing which includes the preliminary sizing of the ribs (see section 3.5.2.1 on page 55) a more realistic design problem was generated. As part of this thesis, this improved model was optimised with STARS. Table 4.5 shows the weight breakdown. The results proved that

| Component         | Weight (kg)  |
|-------------------|--------------|
| <b>Upper Skin</b> |              |
| Web               | 5300.4       |
| Stringers         | 2322.8       |
| <b>Lower Skin</b> |              |
| Web               | 7228.8       |
| Stringers         | 3174.8       |
| <b>Spars</b>      |              |
| Upper Caps        | 267.2        |
| Stiffeners        | 445.4        |
| Webs              | 851.4        |
| Lower Caps        | 342.7        |
| <b>Ribs</b>       | 3098.4       |
| <b>Total</b>      | <b>23032</b> |

Table 4.5: Optimised mass breakdown of reference wing box (units in kg) using STARS with improved rib sizing.

individual sizing of the ribs during preliminary sizing leads to a rib weight reduction of about 5202 kg in comparison with the initial model. Although the ribs are still not involved in the optimisation process, a more realistic wing design is now obtained with an optimised weight for the reference aircraft of 23032 kg after 14 iterations. Figures 4.18 and 4.19 show the optimisation history for this improved model.

The thickness distribution for skins and spars is shown in figures 4.20, 4.21 and 4.22. When comparing these thickness distributions with the previous results, it can be noted that the thickness distribution of the bottom skin has increased for zones 1 up to 7 with a maximum thickness of 31.5 mm. Looking at the bottom skin distribution it is clear that even more tailoring of the chordwise thickness distribution has occurred to increase the bending efficiency.



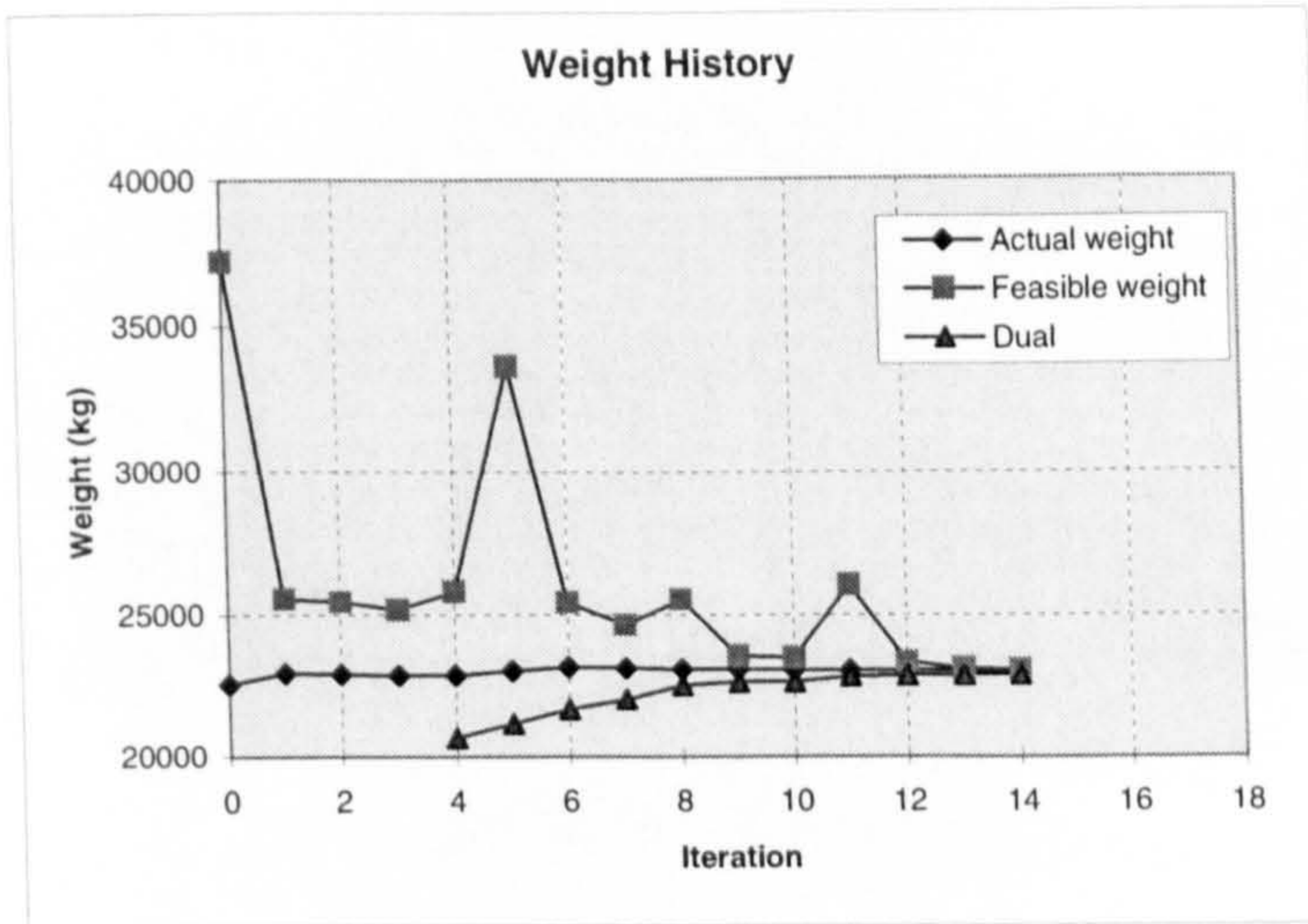


Figure 4.18: Optimisation history, modified model.

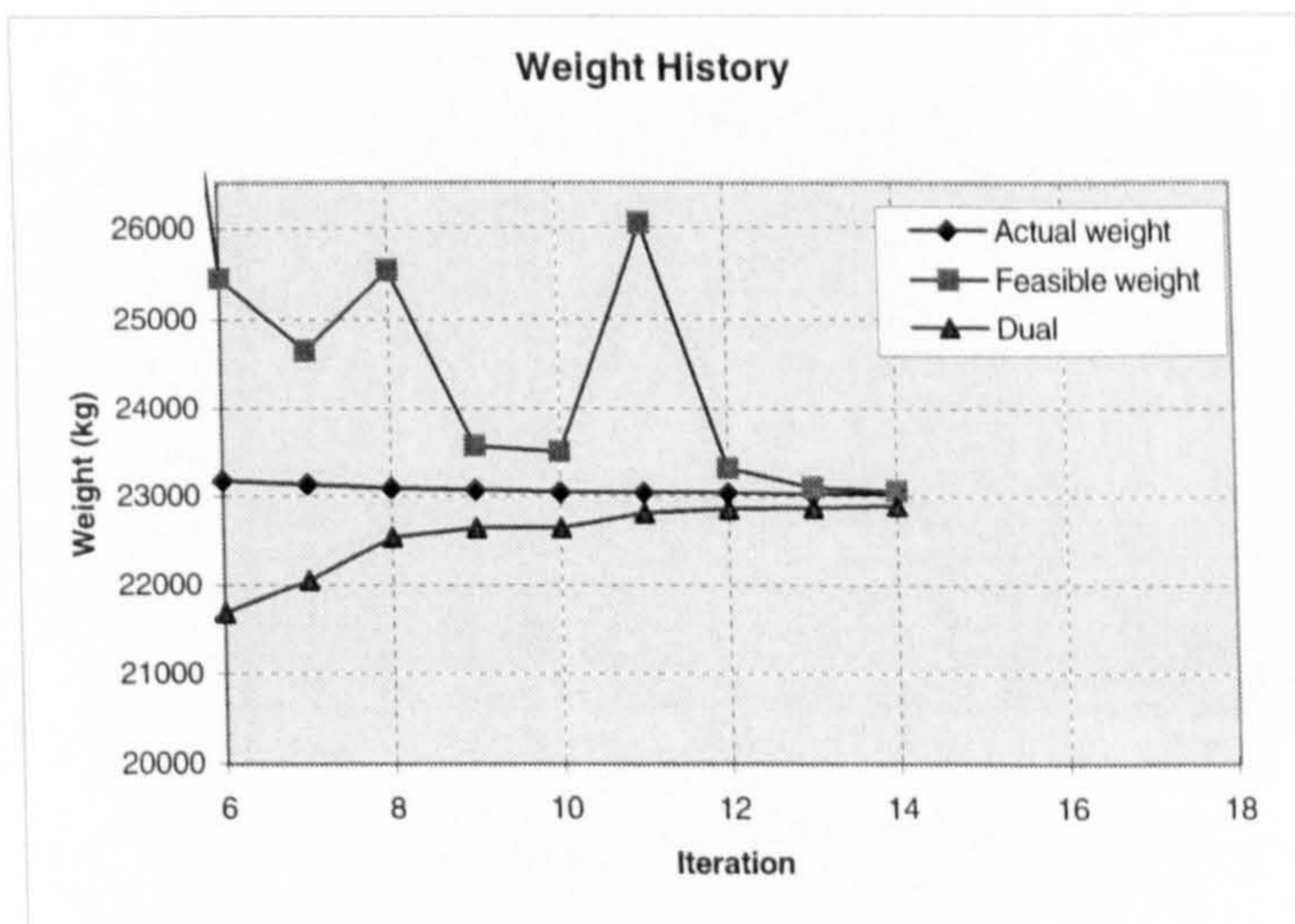


Figure 4.19: Magnified view QNewton optimisation history, modified model.



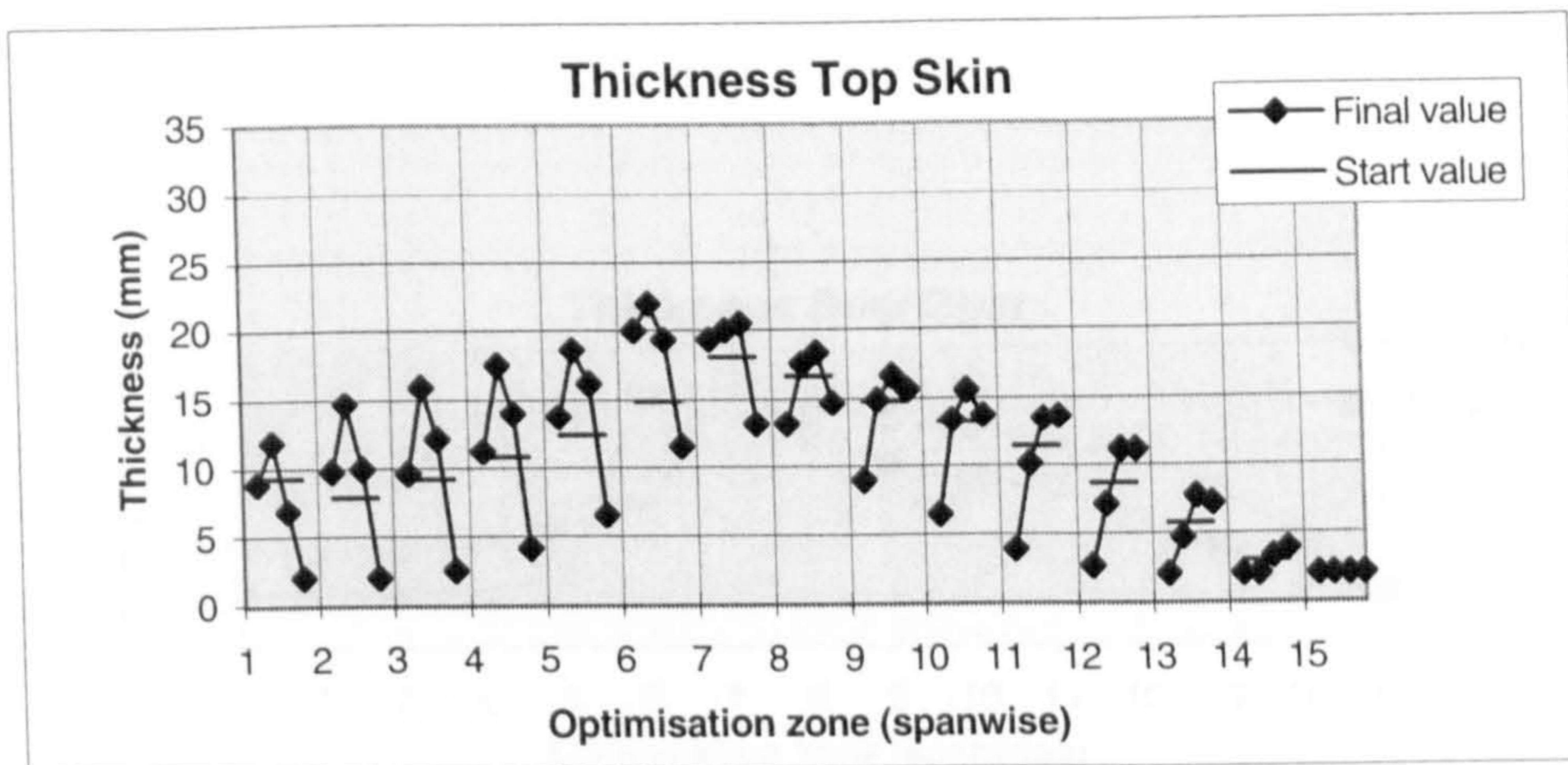


Figure 4.20: Thickness distribution top skins, modified model.

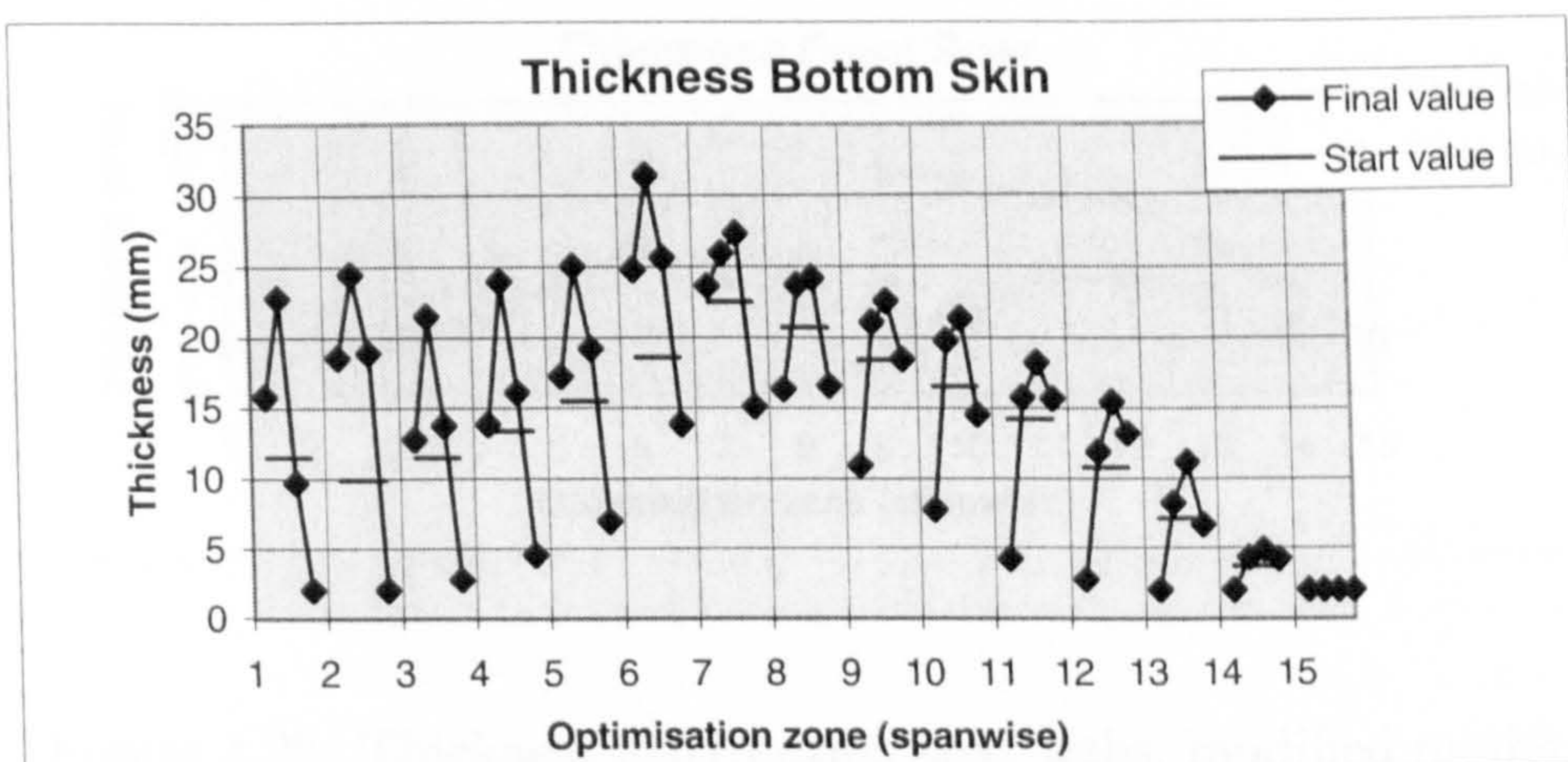


Figure 4.21: Thickness distribution bottom skins, modified model.



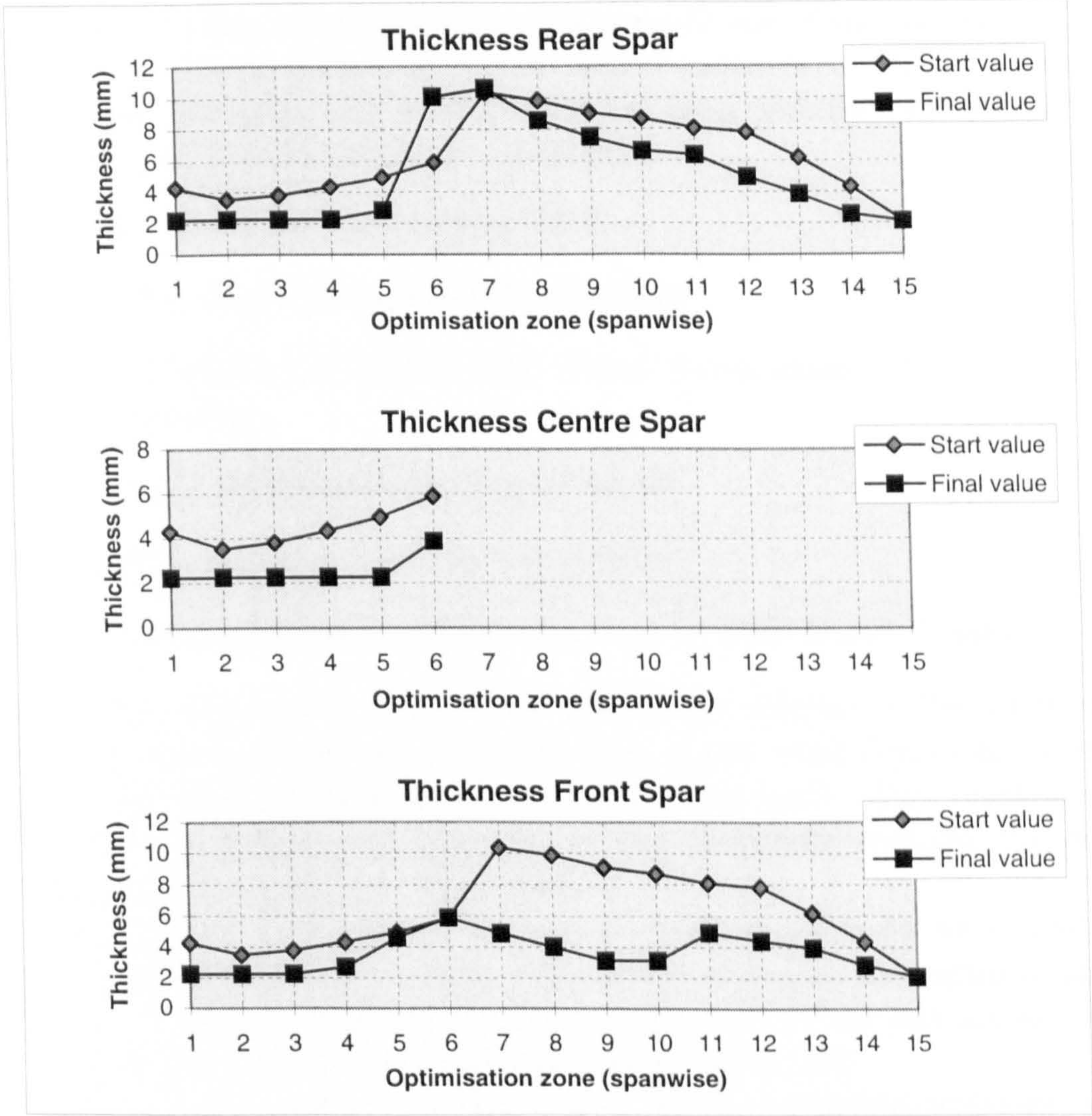


Figure 4.22: Thickness distribution spar webs, modified model.



## 4.5 Comparison with other optimisation packages

This section compares different optimisation results obtained during the MDO project. It also discusses the generation of sensitivities for a 3% change in primary design variables. More information can be found in reference [55]. As indicated earlier in section 3.4.1.4 on page 49 one of the purposes of the MDO project was to allow the various project partners to employ their own design tools. Thus, the MDO wing model has been optimised using a variety of in-house and commercial tools: packages:

- B2000, developed and used by NLR
- ELFINI, developed and used by Dassault
- MSC/NASTRAN, used by BAe, Onera, Aerospatiale, TU-Delft, Alenia and Cranfield.
- OPTSYS, developed and used by SAAB
- ALACA, developed and used by CASA
- STARS, developed by DERA and used by DERA and Cranfield.

The program employed by CASA, ALACA, is different to the other optimisers employed as it is based on optimising at the detailed stressing level so that components are optimised on an individual basis. This method also includes more design variables such as skin thickness, stringer thickness, stringer height and stringer foot width.

The availability of different structural optimisation codes permitted extensive experimentation involving comparison of results and solution methods. As is described in section 4.6 the optimisation process was not so secure and it required a lot of effort to obtain satisfying results.

Table 4.6 shows the mass breakdown for the optimised reference wing box when different optimisers were employed. The results for NASTRAN and STARS are different than those stated in section 4.3.2 and section 4.4.2 as the results of table 4.6 were generated half-way the MDO-project, for a given datum model at that time. As the optimisation process was dynamic in nature it involved many cycles of improving the design model and optimisation approach. The results shown earlier were obtained with a more improved version of the optimisation problem.

As can be seen from table 4.6 the mass of the ribs is significant because these results were generated before the updated rib sizing was introduced. As



|                   | B2000          | ELFINI         | NASTRAN        | OPTSYS         | ALACA          |
|-------------------|----------------|----------------|----------------|----------------|----------------|
| <b>Upper Skin</b> |                |                |                |                |                |
| Web               | 5647.4         | 5593.0         | 5628.0         | 5607.1         | 3798.0         |
| Stringers         | 2445.5         | 2434.7         | 2447.8         | 2438.9         | 3644.0         |
| <b>Lower Skin</b> |                |                |                |                |                |
| Web               | 7220.0         | 7183.9         | 7196.7         | 7192.8         | 6283.0         |
| Stringers         | 3152.9         | 3152.1         | 3168.6         | 3150.6         | 3965.0         |
| <b>Spars</b>      |                |                |                |                |                |
| Upper Caps        | 306.5          | 293.7          | 292.2          | 292.6          | 468.0          |
| Stiffeners        | 445.4          | 445.4          | 445.5          | 445.4          | 445.0          |
| Webs              | 1036.5         | 931.7          | 918.6          | 921.4          | 1033.0         |
| Lower Caps        | 380.9          | 356.3          | 351.3          | 358.5          | 433.0          |
| <b>Ribs</b>       | 8300.9         | 8300.9         | 8301.0         | 8300.9         | 8301.0         |
| <b>Total</b>      | <b>28936.0</b> | <b>28691.7</b> | <b>28749.2</b> | <b>28708.2</b> | <b>28370.0</b> |

Table 4.6: Optimised mass breakdown of reference wing box (units in kg).

shown in the STARS result section (section 4.5 on page 103) the more realistic ribsizing leads to a rib weight estimation of 3098.4 kg which is a reduction of about 5 tons. The mass breakdowns obtained with most optimisers agree well with each other, except those of B2000 which show bigger variations. ALACA predicts different material distribution between skin and stringers, because of they have been optimised independently. It was also found that the close agreement in mass breakdown between ELFINI, NASTRAN and OPTSYS results was also matched by close agreement in the chordwise and spanwise distribution of the skin and spar web thicknesses. [55]

Optimisation results for a 3% change in the primary wing variants are shown in table 4.7 for the different optimisers. For the variants, the results of ELFINI, NASTRAN, OPTSYS and STARS agree consistently to within about 20% with even much closer agreement for most of the variants. [55] [31]

The optimisation of the primary variants showed the following overall tendencies:

- Area change increases mass ( $\approx +1.8\%$ )
- Aspect ratio change increases mass ( $\approx +2.5\%$ )
- Rear spar position change increases mass ( $\approx +1.6\%$ )
- Sweep change increases mass ( $\approx +1.8\%$ )

|              | B2000   | ELFINI  | NASTRAN | OPTSYS  | STARS   | ALACA   |
|--------------|---------|---------|---------|---------|---------|---------|
| Reference    | 28936.0 | 28691.7 | 28749.2 | 28708.2 | 28632   | 28370.0 |
| Area         | 29331.2 | 29131.6 | 29272.0 | 29162.4 | 29150.0 | 28816.0 |
| Aspect Ratio | 29218.9 | 29489.0 | 29474.4 | 29425.5 | 29312.0 | 29016.0 |
| Rear Spar    | 29218.9 | 29132.5 | 29213.6 | 29196.7 | 29180.0 | 30117.0 |
| Sweep        | 29303.0 | 29278.4 | 29281.7 | 29234.8 | 29055.0 | 29182.0 |
| Thickness    | 28638.9 | 28335.8 | 28482.7 | 28392.3 | 28306.0 | 28216.0 |
| Twist        | 29058.9 | 28716.6 | 28818.5 | 28761.1 | 28700.0 | 28364.0 |

Table 4.7: Optimised mass breakdown for variant wings (units in kg).

- Thickness change reduces weight ( $\approx -1\%$ )
- Twist change has not much effect on mass ( $\approx 0.15\%$ )

## 4.6 Problems and aspects of optimisation

At the start of the MDO project it was believed that structural optimisation was a mature technology which would give far less problems than the CFD analysis. However, optimisation of this realistic aircraft design problem has proved that classical structural optimisation is not totally secure. The use of a wide variety of in-house and commercial structural optimisation tools to solve the same problem revealed that different codes gave different solutions. This was not because multiple optima existed, but because certain codes indicated a converged solution when still distant from the optimising point. In order to deliver the results reported in section 4.5, extensive tuning of optimisation control parameters, comparison of different solution methods and reorganisation of the optimisation model was necessary. Fortunately, the availability of different structural optimisation codes permitted extensive comparison of methods and results. This section discusses some of the issues which influenced the optimisation process.

### 4.6.1 User experience

One of the reasons why optimisation has difficulties to expand its field of application in industry, is because of the lack of user friendliness and problems robustness. Despite recent efforts to make commercial optimisation tools more accessible, they still require a lot of experience from the user. [30] [54] [72]



Structural Optimisation, especially when applied to large realistic problems still requires an experienced user to set-up a proper optimisation model and select the proper optimisation method. Especially, when obtaining the first results, the critical assessment of results by an expert is required in order to find out whether the optimisation process working properly or needs tuning.

Another aspect is that a commercial optimisation tool such as MSC/NASTRAN offers a wide variety of methods and control parameters which need careful selection for each optimisation problem. Unfortunately, there is no clear strategy available which could assist the user with determining the right parameters in an efficient way. Instead, it often requires performing numerous optimisation runs in order to find the best set of control parameters. When dealing with a large real world problem, this might not be such a straight forward task and can turn out to be very computational expensive.

In hindsight, user un-experience with MSC/NASTRAN optimisation, has been a factor which influenced the MDO project. Project partners using their own in-house developed optimisation codes were able to obtain much faster results which were acceptable.

#### 4.6.2 Number of design variables

At the start of the MDO project it was decided to assign a separate design variable to the spar web and the upper and lower spar caps. This led to an optimisation model containing 228 design variables. However, optimisation of this model revealed convergence problems. Unaccountable variation in material distribution between wing variants and different optimisation methods was observed. For example, it was found that material in the upper spar caps varied by a factor of 2 between different tools and also varied between aircraft variants with no consistent pattern. Often, from one iteration to the other, material would be swapped from top caps to bottom spar caps and from bottom to top spar caps. In order to obtain more consistent results, the spar caps sizes were linked to the adjacent stringer sizes instead of having them as independent design variables. This led to the optimisation model described in section 4.1 which has 156 design variables.

#### 4.6.3 Constraint screening

Most optimisation problems contain more constraints than are necessary to adequately guide the design. The MDO wing problem, for example contains 23056 stress constraints while it only has 156 design variables. At each design point, only a fraction of the total number of constraints might actually be



influencing the design. A strategy is employed to filter the constraints and to activate or deactivate constraints during the design iterations while the optimiser progresses to the optimum point. This strategy is called an active set strategy. It is a set of rules for changing the working set at a point  $x_k$  to another working set through constraint addition or deletion. Geometrically this means a set of decisions for moving from one constraint surface to another while progressing towards the optimum. [69] A working set can be defined as the set of equality and inequality constraints which are taken as active at a given point  $x_k$ .

Within MSC/NASTRAN, the active set strategy is very limited. Constraints are considered active when their normalised current value is greater than a 'truncation threshold' value. Constraints that exceed this threshold are retained, constraints whose values are less than this threshold are temporarily deleted for the current cycle. This type of active set strategy yields a large number of active constraints. Each optimisation zone contains a large number of elements, hence if a high stress is obtained for one element, it is more than likely that the stress response of all the neighbouring elements will have similar values. This means that the stress constraints of more than one element are generally added to the active constraint set, while they belong to elements which all depend on the same design variable. When the value of the design variable is changed, the stresses of all the elements within that zone will probably vary more or less in unison and hence, even with the current active set there is much more information present than really needed. The number of active constraints becomes much larger than 156 design variables and due to the excess number of active constraints, the optimiser is not able to find a satisfactory search direction. This leads to premature convergence away from the optimum. Ideally, one would only like to have the few largest stresses in one region as active constraint with the other constraints which also exceeded the threshold being ignored temporarily. For this purpose, NASTRAN has introduced the option of Constraint Screening. For each region, only a certain number of active constraints are allowed to enter the active set. For the MDO problem, after a lot of 'trial and error' the constraint screening was selected so that only one constraint from each of the 156 optimisation zones was allowed to belong to the active set. This measure significantly improved the convergence of the optimisation.

Contrary to MSC/NASTRAN, STARS has a much more robust active set strategy which is not based on a threshold value and constraint screening value (to be tuned by the user) but which is based on the Lagrange multipliers. The following active set strategy is used at the end of each iteration [69]:



- Violated constraints are added to the active set. To stabilise the active set, only the most violated constraint of each design variable is included.
- Zero or negative Lagrange multipliers are deleted. To avoid zigzagging only the constraint associated with the largest negative multiplier is deleted.

#### 4.6.4 Definition of FE model

In most finite element models of the MDO wing, the number of property cards was equal to the number of finite elements. For the optimisation problem, the wing is subdivided into structural parts (e.g. upper skin, lower skin, front spar etc.) which are divided into optimisation zones. All the finite elements within each zone have the same property value. Hence instead of defining a unique property card for each finite element, one property value can be assigned for the whole zone and all the elements of that zone refer to that property. This means that the number of properties is equal to the number of zones. In later FE models this has been implemented. For MSC/NASTRAN optimisation the benefits were that the definition of the optimisation model was much smaller. It also resulted in a large reduction (>50 % for CRAY J916) in CPU time for an optimisation analysis. This large reduction in CPU can be explained for the fact that during the analysis a lot of time was spent looking up property and element cards and linking design variable to property and element cards.

#### 4.6.5 Fidelity of FE model

The finite element model which has been used for the optimisation studies has proven to be appropriate for its preliminary design task: a model which can be easily generated and used for static analysis, structural optimisation and aeroelastic studies. However, for detailed stress analysis a more detailed model would be required which uses different element formulations (e.g. bending elements instead of membrane, Quad 8 instead of Quad 4 etc.) and which models areas which show to be stress critical in greater detail. The investigation of other structural wing box features such as pylon attachments, landing gear attachments, manholes, lightning holes, etc. also require a more detailed finite element model. As the current model is already computationally demanding it is clear that for multidisciplinary studies, a higher fidelity model would be impractical because MDO requires several design iterations. Hence the need for several structural models which could be used at several levels at the design:



- Low level models used for interdisciplinary trade-off studies.
- High level models used for a disciplinary study.

Using several levels of modelling, however, requires careful consideration of the correlation between the optimised result of a high level model and that of a low level model. It is not clear whether one can just extrapolate the results, as both models might have been in essence two different optimisation problems. For example, a low level model can be made of membrane and rod elements, while the high level model has replaced these elements by shell elements. Another issue is the frequency that the validity of the current design based on a low level model is being checked with a high level model. Here a trade-off needs to be made between loss of accuracy and computational expense.

A brief study was carried out which increased the fidelity of the current finite element model and which also looked at the effect of a change in the type of elements used.

- **Increase number of elements.**

The current finite element model showed high stress levels in the region of the crank where the inboard engine is located and where also rapid tapering of the wing occurs (i.e. rib 27). A modified finite element model was generated which had an increased number of skin membrane elements in this critical area. Also, because of the high aspect ratio of the spar shear panels the number of shear panels was increased from 1 to 4 in the critical areas. In total this modified finite element model had 15749 elements. Optimisation with MSC/NASTRAN and STARS showed a slight weight increase of 0.6 % in comparison with the optimised reference model. (optimum reference weight = 28677 kg, optimum weight modified model = 28843 kg)

- **Change type of elements.**

Membrane and shear panels are replaced by plate elements. This also included the ribs. Optimisation of this model gave an optimum of 23846 kg. However, the initial weight before optimisation was 26424 kg instead of a starting weight of 31294 kg for the reference and previous model. The lower starting weight in this model can be explained because the rib weight contribution was much less than in the previous models. All models did not take into account the corrected rib sizing (see section 3.5.2.1 on page 3.5.2.1). Hence the ribs counted for about 8 tons. In this model, however, the ribs are modelled as plate elements. In contrast to the rib shear panels of the reference model,



these plate elements do not require additional rod elements to provide axial stiffness. This clarifies the reduced initial weight. As mentioned earlier, the difficulty is now the comparison of these results with those of previous models, as the model is physically different. Because of these differences it is difficult to quantify advantages/disadvantages of this model in comparison to the previous ones. This problem has not been investigated further but it is found to be an important issue for further research.

#### 4.6.6 Dual bound

STARS provided a feature which was found very useful in checking whether the objective was close to its minimum value: the Dual. Duality originates from the definition of a saddle point, which suggests that associated with each minimisation problem which yields the solution  $x = x_*$ , there exists a maximisation problem with the solution  $\lambda = \lambda_*$ . [66]

The minimisation of the objective function, called the *standard* or *primal* problem has always been defined as:

$$\begin{aligned} & \text{Minimise } f(x) \\ & \text{Subject to } g(x) \geq 0 \end{aligned} \tag{4.44}$$

Associated to this minimisation problem one can define an alternative function which is called the *dual* problem and which can be formulated as:

$$\begin{aligned} & \text{Maximise } L(x, \lambda) \\ & \text{Subject to } \nabla_x L(x, \lambda) = 0 \\ & \lambda \geq 0 \end{aligned} \tag{4.45}$$

where  $L(x, \lambda) = f(x) - \lambda^T g(x)$ .

If  $x^*$  is a solution of the primal problem, then there is a  $\lambda^*$  such that  $(x^*, \lambda^*)$  solves the dual problem and  $f(x^*) = L(x^*, \lambda^*)$ . This is called the *Dual Theorem*[66]. Thus, from the saddle point condition and the dual theorem, the following relation follows:

$$L(x, \lambda(x)) \geq L(x^*, \lambda^*) \geq L(x(\lambda), \lambda) \tag{4.46}$$

Equation 4.46 shows that for a given  $x$  and  $\lambda \geq 0$  that we can use  $L(x, \lambda(x))$  and  $L(x(\lambda), \lambda)$  as upper and lower bounds for  $f(x^*)$ . For the MDO problem, the value of the lower bound  $L(x(\lambda), \lambda)$  has been very useful to track whether the weight minimisation problem was close to the optimum.

## 4.7 Conclusions

Solving the MDO optimisation problem has shown that the classical structural optimisation, which is often regarded as 'mature' technology, is not as secure as is generally assumed. Proper definition of the optimisation model, knowledge of the optimisation algorithms, and user experience, and often 'trial and error' is required in order to obtain good results. In contradiction to how it is often sold by commercial software developers it is not 'plug and play'.

For the MDO model, the optimised weight (taking into account rib sizing) is of the order of 24000 kg. However, the actual mass will be higher because the simplified finite element model did not model all structural features in detail. A weight increase of about 15 % is not unusual. [55]

Optimisation of a combined metal/composite wing, using a simplified composite model, lead to an additional weight reduction of about 1.6%.

The MDO model contains a total of 69 ribs. As is shown in the next chapter there are too many ribs present and in reality about 40 ribs would be required. This will of course reduce again the total weight of the wing box.



# Chapter 5

## Manufacturing Cost

This chapter discusses the issue of manufacturing cost and the development of a manufacturing cost model for a multidisciplinary design and optimisation process. Manufacturing issues of both metal and composite wings are covered. A general discussion about manufacturing costing for MDO, cost drivers, application of composite versus metal and examples of current cost models for MDO is given in the introduction section. Section 2 gives an overview of different methods to estimate cost and the specific issues and requirements which are related to the cost model outlined in this thesis. A description of the developed cost models for both metal and composite wings is given in section 3. The final section of this chapter discusses results from applying the cost models to the design of a civil airliner wing.

### 5.1 Introduction

In applying multidisciplinary design optimisation (MDO) to preliminary design, the emphasis has traditionally been on obtaining the best trade-off between weight and drag. The target being to improve some specified measure of performance such as direct operating cost, range, payload or speed. However, while attempts to lower aircraft weight, in general, must be encouraged, the process through which this goal is achieved must be mindful of the entire aircraft development including the manufacturing processes involved and the cost associated with each process. [73] For a commercial aircraft, for example, the number of engines, the shape and size of wing and wing components, the manufacturing methods, etc. must be determined very early in the design cycle. This is because all the successive design studies and manufacturing costs, such as for tooling or building of a new assembly factory depend on these choices. Unfortunately, these important decisions have to



be made when least is known about the design. [1]

It has been recognised that although most of the product development cost occur late in the development cycle (e.g. manufacturing cost), a large percentage of these cost are committed very early during the conceptual and preliminary design phases. Boothroyd et al pointed out that while design activities make up about 10 % of the product cost, they indirectly contribute to about 70 % of the product cost [74]. The decisions made in the early stages of the design process have a greater impact on the final product than those made in the later stages. In the early stages of a traditional design process, products are often designed without giving adequate consideration to manufacturing limitations. This can result in designs that cannot be manufactured, requiring further modifications until the product makes it to the product shop. [5] Usually this occurs because the designer is not familiar with the limitations of the manufacturing process and because of the fact that the design and manufacturing groups are separate and the information flow between them is minimal. Hence, it can be argued that with proper attention to product manufacture in the design phase, the effort in the prototyping, test and evaluation, and production phases could be substantially reduced. Recognising that aircraft design is a vital part of aircraft development, it becomes clear that factors affecting manufacturability and cost of the aircraft need to be considered early in the design stage, this in parallel to those of weight and performance [73] As Cooke stated, it is currently important to reject the idea that a product can be designed without taking into consideration the manufacturing and support costs. *'Rather than asking how a product can be manufactured more economically, in order to reduce costs, industrial organisations should ask how the product can be redesigned so that it can be manufactured more economically'* [75].

In the context of MDO methodologies, as applied to preliminary aircraft design, the combination of manufacturing and cost influence factors with structural allowables and performance constraints would form a more rigorous set of requirements and result in more realistic optimum designs. Integrating manufacturing issues into the MDO process, would allow the designer to study relevant design issues such as:

- How can the aircraft manufacturing cost be reduced?
- How do the variations in manufacturing processes influence the aircraft manufacturing cost?
- Can the aircraft cost be lowered by using light weight advanced composite materials instead of metal alloys?



While in recent years substantial progress has been made in the field of MDO and its application to the design of advanced aircraft systems (e.g. High Speed Civil Transport, HSCT) there still remain many issues which need further research in areas such as reliability, manufacturability and cost factors. Identification of the aircraft design drivers affecting these factors and their integration into an MDO based design process would enhance the quality and efficiency of aircraft design. Hence it is essential to consider the limitations associated with the use of any material, fabrication technique, assembly procedure and manufacturing cost when judging which design has a better optimum.

Improved aircraft performance, within cost limitations, depends upon the engineering design excellence. Affordable aircraft performance depends upon identifying cost drivers, recognised by both designers and manufacturing engineers, controlling these cost drivers in new designs and improving manufacturing methods for existing products. As mentioned earlier, the preliminary design phase provides the best opportunity to achieve a low cost design: innovative materials, design concepts and manufacturing technologies can have a significant impact on cost. However, this can only be obtained when design teams are provided with adequate tools which:

- Consider cost as a primary design objective
- Identify cost drivers in early decisions
- Provide designers with meaningful cost data at the start of the development
- Increase the number of performance cost trade-offs of alternative designs
- Determine the cost of changes in design objectives and engineering solutions
- Improve interaction between design and other disciplines

### 5.1.1 Cost drivers

Manufacturing and cost issues in aircraft design are measured by qualitative and quantitative information. Qualitative information identifies the cost drivers and shows their relative effects. It indicates to the designer which cost aspects can be influenced by the design process. Quantitative information provides the designer with man hour data. [76] Cost drivers can be related to various categories of aerospace system development and manufacturing.

This fact is important to keep in mind, when analysing the cost drivers for a discrete parts and assembly. Noton has identified manufacturing cost drivers in conventional and new manufacturing technologies. The cost drivers are listed in the following four general categories: concept and performance requirements, design, material selection, and manufacturing. Cost drivers are often common for both primary and secondary aircraft structures. Examples of typical cost drivers for a mechanical system are:

1. Concept and performance related

- Reduced weight
- Higher operating speeds
- Increased reliability
- Improved maintainability

2. Design related

- Part count
- Non standardisation
- Special tolerances

3. Material related

- Cost
- Availability
- Utilisation
- Energy requirements
- Inventory

4. Manufacturing related

- Cyclic production
- Small lot size
- Job shop environment
- Highly skilled labour
- Material removal
- Hand fit-up
- Hand finishing/Deburring



- Energy requirements (e.g. use of autoclave)
- Facilities
- Qualification
- Test and Inspection

Cost drivers sometimes result from progress in technology. For example, aircraft structural concepts using advanced composites or titanium require new developments in manufacturing technology. Typical cost drivers for advanced composites are [77]:

- Complexity of assembly operation
- Part count
- Part type and function
- Fibre types
- Quality and repairability issues
- Number of plies
- Automatic versus manual lamination
- Overlaps
- Autoclave costs (dedicated autoclaves and cycles)
- Lot size
- Quality requirements
- Gaps (shimming)
- Non-automated production
- Facility requirements
- Design, manufacture and refurbishment of tooling
- Lamina form
- Material storage and shelf life requirements
- Resin Systems

- Finishing requirements
- Paint removal
- Waste disposal and environmental concerns
- Special fasteners
- Production support costs
- Skills turnover
- Training requirements
- Part size
- Corrosion of lightning protection materials
- Curing method

To be able to identify the relevant cost drivers, good understanding of the manufacturing processes involved is necessary. Main resources to help determining cost drivers for the designer are the cost estimator and manufacturing specialist. Close interaction between design and manufacturing is therefore essential. As mentioned above, automation of the lay-up process for composites can produce cost savings, hence the cost driver. In general, however, automation is not cost effective if production rates are low. Automation is also more cost effective on larger parts and may not be cost effective at all on small parts.

### 5.1.2 SAVE, an example of recent manufacturing cost modelling developments

[16] The Simulation Assessment Validation Environment (SAVE) program, carried out by Lockheed Martin, was initiated in 1995 by the Joint Strike Fighter Program. It is funded by the Joint Strike Fighter Program Office and runs up to the end of 1999. The objective of SAVE is to demonstrate, validate and implement integrated modelling and simulation tools and methods which are used to assess the impacts on manufacturing of product/process decisions early in the development process. The key anticipated results of the SAVE program are the demonstration of an initial virtual manufacturing capability and the validation of this capability to reduce the maturation costs and risks associated with the transition of advanced product and process technologies into production. Several commercial simulation tools are



used to assess the cost, schedule and risk of product and process design decisions: CAD(CATIA), Cost Modelling (CostAdvantage), Schedule Simulation (FACTOR/AIM), Assembly Simulation (IGRIP/ERGO), Factory Simulation (QUEST), Risk Assessment (ASURE), System Optimisation (Production Simulation). The SAVE cost modelling system uses the commercial CostAdvantage software and follows a feature based approach. It contains a series of knowledge bases which are used to define cost and producibility rules for manufacturing processes based on information about product features. Four cost models are developed covering 5-axis machined parts, hand lay-up composite parts, sheet metal and assembly cost. Each of the cost models relies on the extraction of features from the CAD model. The cost models require inputs such as feature parameters, material selection, process selection, number of units, units per aircraft and rules. Cost outputs from the models are recurring manufacturing labour and material cost, non-recurring tool manufacturing and tool material cost, non-recurring engineering cost, first unit cost, sustaining tool engineering and manufacturing cost, quality assurance cost and process plan simulation. Three major demonstrations are included in the SAVE program which:

- Validate that a set of disparate commercial off-the-shelf simulation tools can be integrated and produce results which closely correlate to real manufacturing data. The F-16 horizontal stabiliser was used for this validation, which was carried out successfully in December 1996. Estimation of cost was within 15%, schedule was within 18% and risk was within 3% of the actual F-16 program data.
- Perform a design/manufacturing trade study scenario: resizing of F-22 gun port. (June 1998)
- Assembly optimisation scenario using F-22 forward fuselage. (begin 1999)

The two JSF prime contractors, Lockheed Martin and Boeing were selected as test sites to Beta test the SAVE system mid 1998. This to more rapidly mature the SAVE software and to address the issues of real production implementations.

### 5.1.3 Metal versus Composites

This section discusses some of the main characteristics of composites and gives advantages and disadvantages in comparison to metal. A detailed discussion on the use of composites for aerospace purposes can be found in



*Composite Airframe Structures* [78] and *Airframe structural design* by Niu [79], and in *Composite Materials in Aircraft Structures* by Middleton [80].

The development of advanced fibre composites in the 1960's gave aircraft designers a new material option, comparable to the introduction of aluminium some 40 years earlier. Carbon fibres, with moduli and strengths comparable to steel and a density of half that of aluminium, created visions of 50 % weight saving for airframe structure. Although such weight savings have been achieved on a few specific components, weight savings of 15-40 % appear to be a more realistic and achievable goal. This due to: the added weight associated with load introduction, the need to satisfy multiple design conditions, design and producibility requirements that usually need a balance of in-plane properties, accessibility for maintenance, inspection and damage repair, and production cost constraints. [81]

In comparison with metal (aluminium alloys) the main differences of composites are that:

- Composites are anisotropic with properties not being uniform in all directions
- Strength and stiffness can be tailored to meet load requirements
- Composites offer a greater variety of mechanical properties (determined by the type of composite)
- Composites have a poor through thickness strength
- Composites are more sensitive to environmental heat and moisture.
- Composites have a greater resistance to fatigue damage.
- Damage propagation through delamination rather than through-thickness cracks

The main advantages of composites are the lower weight, the high resistance to corrosion and fatigue damage, the tailoring of fibre orientation in directions where high strength/stiffness is needed and the low thermal expansion. Other advantages are the ability to absorb radar microwaves in stealth applications and the reduction of the number of assemblies and fastener count. The reduction of part count is made possible through cocuring of large assemblies and innovative designs which minimise the number of parts that have to be joined in separate assembly operations. However, with the increase of part size it is more likely that the part will become more complex and consequently the tooling may become more complex and costly. Hence,



the benefits of reduced assembly cost must be weighed against the possible increase in tooling cost and the risks involved in curing larger parts. Designers must also understand that not everything on a composite aeroplane needs to be made of composites. Small parts, for example, can be very expensive when made of composites and metal may be the most cost effective choice. Disadvantages of composites are:

- High material cost
- Problems of galvanic corrosion due to improper coupling between composites and metal
- Poor energy absorption and impact damage
- Degradation of structural properties due to temperature extremes and wet conditions
- The need for lightning strike protection
- Expensive and complicated inspection methods
- Difficulty for detection and precise location of defects
- Lack of established design allowables.

Due to the high composite material cost in comparison to the cost of aluminium alloy sheet and plate, care must be taken to minimise composite material waste and scrap. Designing for producibility is essential. Assembly costs and composite manufacturing cost must be considered when selecting a design and manufacturing process. Another aspect of composites are the dimensional tolerances which are more critical than in metals. Composites have problems dealing with out of plane loads, induced by joints, structural discontinuities and other areas of stress concentration. The costs associated with assembly of mechanically attached composite joints are very high and require close manufacturing tolerances at faying surfaces and rigid control of the thicknesses of the parts to be joined. To prevent the introduction of out of plane loads in the composite parts at the joint during assembly, liquid or structural shimming must be used. The high costs of special fasteners, their installation, the control, inspection and measurement of the thicknesses of the composite parts (to assure proper fastener selection), and shimming are the primary reason for minimising mechanical joints in the design. Installation cost of mechanical fasteners for composites are so much higher than for metal because the time required for hole preparation, measurement and inspection of each hole, and the cost of special fasteners (titanium or stainless steel).



Tooling is another critical element in the composite manufacturing and assembly process. Composites require more high quality tools. They are essential to the production of quality parts with high tolerances and are a cost effective element of low cost production (hence a cost driver, as mentioned in the previous section). The selection of tool material for composites is dependent on the part size and configuration, the production rate and quantity, and company experience. Tools often require modifications before or during the early phases of production of composite parts. The tool designers should anticipate the need to modify tools to adjust for part springback, ease of part removal, or to maintain dimensional control of critical interfaces. Hence, tool design and tool material selection must be an integral part of the overall design process, especially with cocured structures.

Quality control is another area which is more costly than for metal structures. Most of the quality control costs can be attributed to non-destructive inspection of completed composite parts. Post assembly inspections are also essential to verify the assembly process and to assure that the part has not been damaged in the assembly process.

Another aspect is the in service maintenance, damage and repair. Although composites have a much longer life due to their high durability and corrosion resistance, maintenance and repair issues can't be neglected. For example, civil and military aircraft must be often repainted. In the case of composites, paint stripping poses a special problem since commonly used solvents can damage the epoxy matrices. The repair of damaged composite parts takes much more time and is much more expensive compared to metal components. A reason for the higher repair cost of composites is that specially approved fasteners and spare parts have to be used. However, service experience with composite primary and secondary structures has proved to be very positive, in comparison to similar metal components and no aircraft has been lost due to failure of composite structures. Airline and military service experience has shown that most damage to composite structures (especially secondary components) occurs during aircraft servicing and routine maintenance which is often not related to the composite part. Supportability should be adequately addressed during the design and composite structures should also be designed to be inspectable, maintainable and repairable. Directly related to maintenance and repair are the maintenance staff. They have to be adequately trained so they understand the characteristics and problems of composites. Maintenance staff needs to be aware that, for example, dropping a spanner on a composite component might not show any visual damage while in fact damage might have been introduced further down the component. Even when no damage is visual, it needs to be properly recorded and ignorance might be fatal. It is clear that, to obtain this, mutual effort



is needed from both management and service staff.

A final aspect is the disposal of composites at the end of a composite component's life time. Waste treatment is an issue which is often neglected, but should be taken into account as soon as composite implementation is considered. Unlike metal components, composites are not suitable for reuse and will pose waste problems in the long term. For Eurofighter (Typhoon), for example, one is aware that this problem will arise in 30 years. Currently no real effective ways of disposal have been found, but one expects to have found one by the time the Eurofighter reaches the end of its operational life. One certainty is that the disposal problem will be an expensive issue.

To conclude, an interdisciplinary team of experienced knowledgeable people working together can make the technical risk of applying composites comparable to that of any other advanced structure. Weight savings alone are no longer considered sufficient enough to justify the use of composites. Composite structures must be cost effective. The weight saved and other in service benefits, such as durability or corrosion resistance, must have enough value to offset any added costs that may arise from the use of composites. Optimisation for producibility and supportability can significantly reduce the cost of composite structures with little or no weight penalty or loss of structural performance. [81]

#### **5.1.4 Implementation of composite structures for commercial transport aircraft**

This section gives a brief historic overview of composite implementation for large commercial transport aircraft. More information can be found in references [79], [81], [80].

##### **5.1.4.1 Airbus Industrie**

From the early seventies airbus invested in research and development of composite components. In 1985 it was the first airframe manufacturer to use composite materials for a series production of primary structures when it started the assembly of the A310 which has fins built of carbon/epoxy. The use of composite fins has resulted in a weight saving of 22 % compared to its aluminium counter part. In addition it only consisted of 95 parts with 2076 parts in the previous aluminium box structure and insuring a reduction of assembly cost. Main Airbus CFRP manufacturers are DASA (A310, A320 and A330/A340 CFRP vertical stabilisers) and CASA (A320 and A330/A340 CFRP horizontal stabilisers). The progress of Airbus composite development has been as follows:

- 1972: A300 early design of fibreglass (GFRP) fin leading edge and fairings.
- 1978-1979: A300-600 components include CFRP elevators and rudders, spoilers, air brakes, nose landing gear doors and main landing gear leg fairings.
- 1980-1985: A300/A310 vertical fin.
- 1985-1987: A320 horizontal tail and vertical fin, elevators, rudder, ailerons, spoilers, flaps, wing leading and trailing edge access and fixed panels, landing gear doors, engine cowls, engine doors and fairings (of GFRP). Composites account for about 15 % of the structure of the A320.
- 1987: Development of new CRRP tooling and manufacturing methods in the design of the A330 and A340 vertical fin and horizontal tail. The A330/A340 uses composites for the same components as the A320. Although the total weight of composite structure used is much higher than for the A320, CFRP components take up 12 % of the total A330/A340 structural weight.

#### 5.1.4.2 Boeing, McDonnell-Douglas, Lockheed

The first airliner advanced composite component was a Boeing 707 boron/epoxy fore flap, flown in 1970. Between 1972 and 1986 most of the composite components for US aircraft were developed under several NASA Langley Research Center programmes, such as the Aircraft Energy Efficiency (ACEE) program. The NASA programs included limited production and airline service evaluation of various composite components and also looked at long term effects of exposure to moisture, ultraviolet radiation, fuels and hydraulic fluids on the mechanical properties of composites. ACEE developments included three secondary and three primary structures:

##### 1. Secondary structures

- Lockheed L1011 inboard aileron. Eight components were designed and built and entered service in 1982.
- Boeing 727 elevator
- McDonnell-Douglas DC-10 upper rudder. Thirteen rudders were designed and built and entered service in 1975. Some are still in service. This design approach was later implemented on the MD-11 CFRP ailerons.



## 2. Primary structures

- Lockheed L-1011 vertical fin box. This fin has never been in service, as it failed during static test. (partly because of the method of load introduction) Extensive environmental and cyclic load tests were carried out on spar and skin panels. No flight articles were built.
- Boeing 737 horizontal stabiliser box. In total, 4 were produced and flown on 2 aircraft in 1984. These stabilisers were the first primary commercial transport CFRP structures certified for airline service.
- McDonnell-Douglas DC-10 vertical fin box. This multi-spar vertical stabiliser developed in 1977 and remained in service until 1993 with Finair.

The experience gained from the ACEE programs resulted in an increase use of composites on the next generation of US commercial transports. The Boeing 757 (first flight 1982) and 767 (first flight in 1981) have CFRP composites for the rudders, elevators, spoilers, landing gear doors and engine cowlings. The flaps of the 757 are also CFRP. The Boeing 737-300, first introduced in 1985 used CFRP composites for ailerons, elevators, the rudder, fairings and engine cowl doors. Boeing extended the use of composites in the 777. The 777 CFRP components are the tail, control surfaces, floor beams, main landing gear doors and engine nacelles. Other composite components include the wing-fuselage fairings and wing fixed trailing edge panels.

The structure of the McDonnell-Douglas MD-11 includes CFRP elevators, winglets, ailerons, outboard flaps, spoilers, wing fixed trailing edge panels, tail cone, engine cowls, centre engine inlet duct, cabin floor beams and AFRP/GFRP wing body and aft body fairings. The MD-11 first flew in 1990. Some of MD's suppliers include Fuji and Mitsubishi (Japan), Embraer (Brazil) and Westland (UK).

In 1988 NASA initiated the Advance Composite Technology program (ACT) with the emphasis on development of advanced materials, mechanics, innovative concepts and low cost manufacturing methods. This in cooperation with Boeing and McDonnell-Douglas. Examples of new developments initiated by the ACT program are the use of the Resin Transfer Moulding (RTM) technique, using woven fabrics and stitching methods for the fabrication of aircraft components.



### 5.1.4.3 Illushin Il-86 and Tupolev Tu-204

The Illushin Il-86 wide-body airliner entered service with Aeroflot in 1980 and has CFRP cabin floors. Its derivative, the Il-96, has CFRP flaps and cabin floors. The horizontal and vertical stabiliser leading edges are made of GFRP/CFRP. It first entered service in 1988.

About 18 % of the structural weight of the Toupolev Tu-204 medium range airliner is made of composites. It first flew in 1989 and is the Russian counter part of the Boeing 757. CFRP components include spoilers, air-brakes, flaps, elevators and the rudder. Other composite components include part of the wing skins, stabiliser leading edges and wing fuselage fairings.

## 5.2 Estimation of manufacturing cost

### 5.2.1 Classification of cost models

Over the past decades several manufacturing cost estimation programs and methods have been developed or proposed. A detailed overview of different costing approaches has been carried out by Rais-Rohani and Dean [73]. It shows that most cost models could be classified under two headings: parametric and process cost models.

#### 5.2.1.1 Parametric cost models

The origins of parametric cost estimating date back to World War II. The war caused a demand for military aircraft in numbers and models that far exceeded anything the aircraft industry had manufactured before. Parametric costing was used to predict the unit cost of aeroplanes.

Parametric cost models are widely used in industry. Their formulation is relatively easy, but their accuracy depends strongly on the accuracy of the manufacturing data and manufacturing history on which they are based. Hence, when estimating the costs for processes involving new manufacturing technologies or materials, the accuracy is often poor as little historical data is available. The use of parametric cost models requires a good understanding of what the cost drivers are for the manufacture of specific product. The cost driver information and its relationship to designer controlled parameters needs to be translated into a quantitative form which is meaningful for MDO purposes.

A frequently used and easily defined cost driver for aircraft based parametric models is weight. Weight based cost estimation relationships, however, do not always accurately represent the actual manufacturing cost and it



may not provide accurate sensitivity data for the MDO process. [29] Often a weight reduction will result in a cost increase due, for example, to a requirement for more machining time, closer tolerances etc. An example of this type of cost augmentation, with weight reduction, is shown later in the results of section 5.2.4.1 on page 147. Hence fabrication costs are better correlated to structural layout and complexity than to weight. Accurately representing all the details of manufacturing complexity is difficult for a parametric model, as it must include all the product and process specific parameters (drivers) which can influence the parametric cost model.

#### **5.2.1.2 Manufacturing process cost models**

Manufacturing process cost models require a thorough understanding of the manufacturing processes involved. They are based on a detailed estimation of the main manufacturing cost categories such as material use, fabrication, assembly. These type of models also focus on labour and process time and cost. The models are formulated in such a way that they cover the costs associated with individual processes and assembly operations. For the estimation of the full production cost, the recurring and non-recurring manufacturing costs are identified and calculated separately. Manufacturing process cost models are more accurate than parametric models, but need much more detailed information at the start of an estimation process. Thorough identification and analysis of the separate manufacturing processes is necessary.

#### **5.2.2 MDO cost model approach**

Depending on the level where multidisciplinary design and optimisation is being applied to, different types of cost models are suitable. For conceptual design studies, where most analysis is based on parametric studies and where mainly historic data is used, a parametric manufacturing cost model would satisfy. However, the application of MDO approaches to the preliminary design requires manufacturing cost models which have a higher level of detail and complexity. The examples of section 5.1.2 on page 121 have highlighted the highest level of cost modelling which not only allows to study manufacturing costs of different configurations, but also the investigation of different manufacturing methods, assembly processes, production methods etc. A great disadvantage of these all-comprehensive models is the amount of information and analysis time required. As for the MDO implementation of other disciplines such as CFD and structural analysis, one needs to make a trade-off between accuracy and computational effectiveness for the cost model.

From the point of view of the European MDO project, which was distributed in nature and involved a high number of different companies, the cost model also had to provide:

- A generic, company independent model which can be used by all fourteen project partners.
- A model which can be easily customised or expanded by individual partners to the needs of their company.
- A manufacturing cost estimation for both metal, composite or hybrid wings.
- A clear visualisation of the cost changes with respect to design changes.
- An integration of the cost model into the MDO-project software.

Because of the specific nature of the MDO project, a number of practical problems influenced the choice and appearance of the cost model. This would not occur if a cost model was developed for a single company. Some of these problems related to the confidential nature of company cost information, others to the fact that manufacturing and assembly processes are influenced by company policies, market trends or company infrastructure. Hence it was difficult to develop a cost model which suited all the project partner's specific needs. For this reason it was decided that adaptability and expandability of the cost model were important requirements.

The cost model approaches, mentioned in section 5.2.1.1 appeared to be unsuited in providing a cost model which met the specific needs of the MDO project. It was found that most cost approaches did not have the required flexibility and generic character or the level of detail required to handle the complex wing design problem at the preliminary design level. In consequence, it was decided to develop a model which combines aspects of both parametric and process cost models as part of this PhD research. The model follows a feature based approach employing weight (component volume), component layout and manufacturing rules. Through this approach, the traditional weight based cost model has been expanded to one which directly takes into account assembly and detailed manufacturing cost. This is further discussed in the next section.

The feature based approach and all the cost models which are described in the next sections have been developed by the author only as part of his PhD research. As a member of the European MDO project team, the study of manufacturing cost and the integration of it into the MDO software has been the exclusive responsibility of the author.



| Feature type | Examples   |
|--------------|--|
| 1. Geometric | Length, Width, Depth, Volume, Area, Perimeter, etc.                        |
| 2. Attribute | Tolerance, Mass, Density, Finish, Material composition, etc.               |
| 3. Physical  | Hole, Pocket, PC board, Skin, Spar, Rib, Wing, etc.                        |
| 4. Process   | Drill, Lay, Weld, Machine, Form, Mill, etc.                                |
| 5. Assembly  | Interconnect, Insert, Align, Attach, Engage, etc.                          |
| 6. Activity  | Design engineering, Structural analysis, Quality assurance, Planning, etc. |

Table 5.1: Example of feature categories.

### 5.2.3 Cost Model Description

This section details the feature based manufacturing cost models developed for both metal and composite wings. An earlier feature based costing model was discussed in a paper by Taylor [82] from British Aerospace Military Aircraft & Aerostructures at Warton. It shows that features can be a vital element in the cost prediction process as it allows engineering information to be encapsulated within the feature. The engineering intent is additional to the base geometrical definition and can be in several forms such as product function, performance, manufacturing process, behaviour etc. Currently feature based costing is being implemented at Warton in their latest generation of cost estimation programs. In the absence of a recognised standard, the Warton Cost Engineering group have defined a series of feature categories to meet their requirements. Table 5.1 gives an overview of some of the feature classifications defined in the developed cost models.

The cost models calculate the recurring manufacturing costs, i.e. the cost which are directly used for the manufacture and assembly of an aircraft wing box. Non-recurring cost such as engineering, testing, tooling, equipment, utilities, depreciation of infrastructure, etc. are not included.

Each of the cost models, described below, are based on a definition of the product structure (bill of material) for the wing box. This is currently based on the A340 but the MDO software, discussed in section 3.5 allows for a definition of different product models. Figure 5.1 shows the current product structure for the manufacturing of the A300 series wing box. An exploded

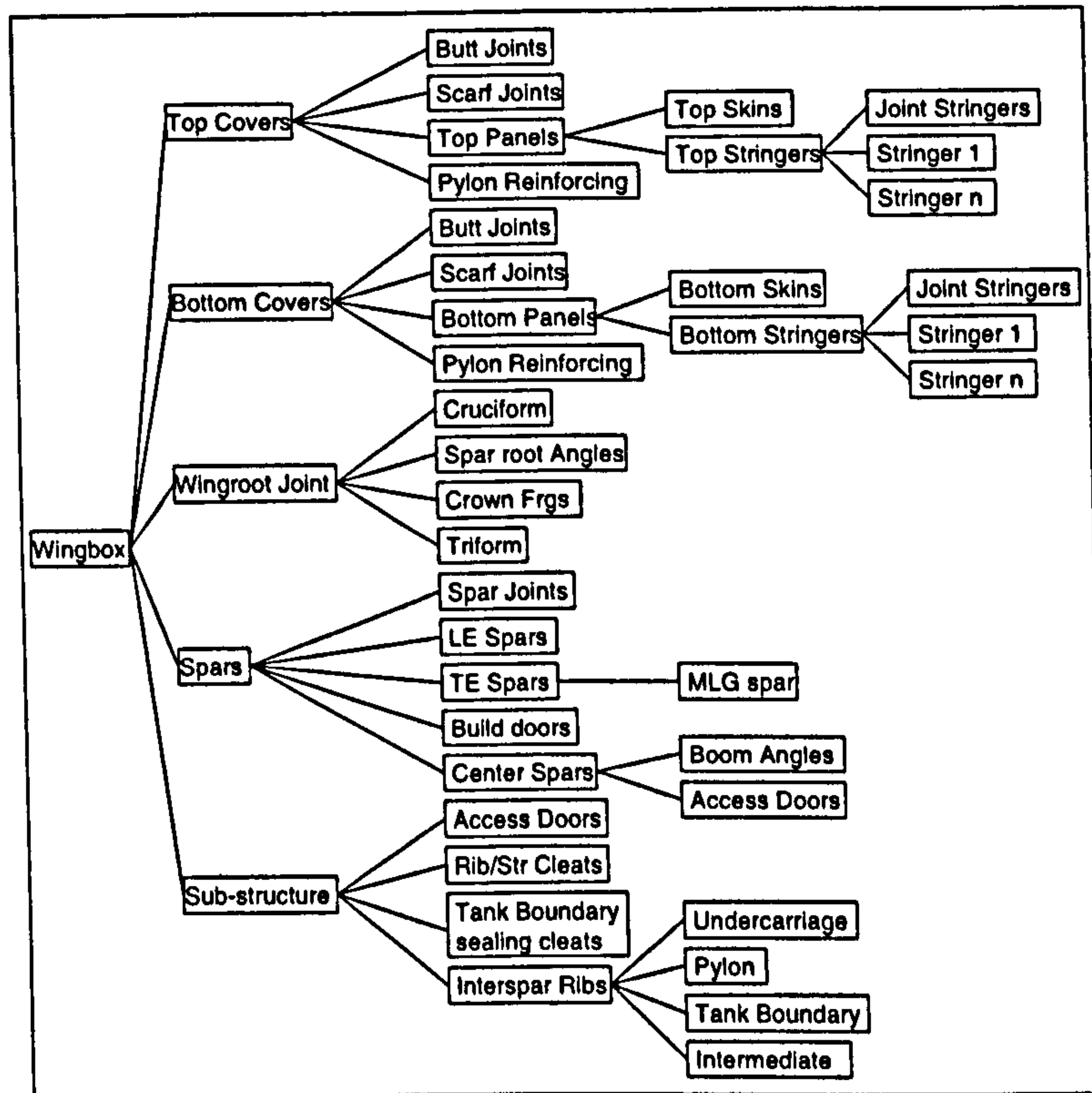


Figure 5.1: Product structure wing box A300 series.

view of the wing box structure of A330 and A340 is given in figure 5.2.

Once the whole product structure is defined, each of its components is analysed with respect to material, manufacturing and assembly processes. For each component, several features which drive the component cost are assigned. These features, or cost drivers, can be: geometrical such as length, area, volume; processes such as milling, drilling, assembly including joints, inserts; etc. The present cost models, described below, concentrate on geometric and assembly features. For simplicity, a certain manufacturing process was chosen and kept fixed. However, the software allows for easy change of processes if required. The only requirement is that proper cost information on the new process needs to be available. The feature information is obtained from the MDO software and is automatically updated each time the wing design is changed by the Multi-Model Generator (MMG), described in section 3.5.2. After identification of the relevant features, cost factors (e.g.  $\$/m$ ,  $\$/m^2$ ,...) need to be defined. This is done using an estimation program (described later in this chapter) or by directly assigning the values. The recurrent cost can now be calculated as:



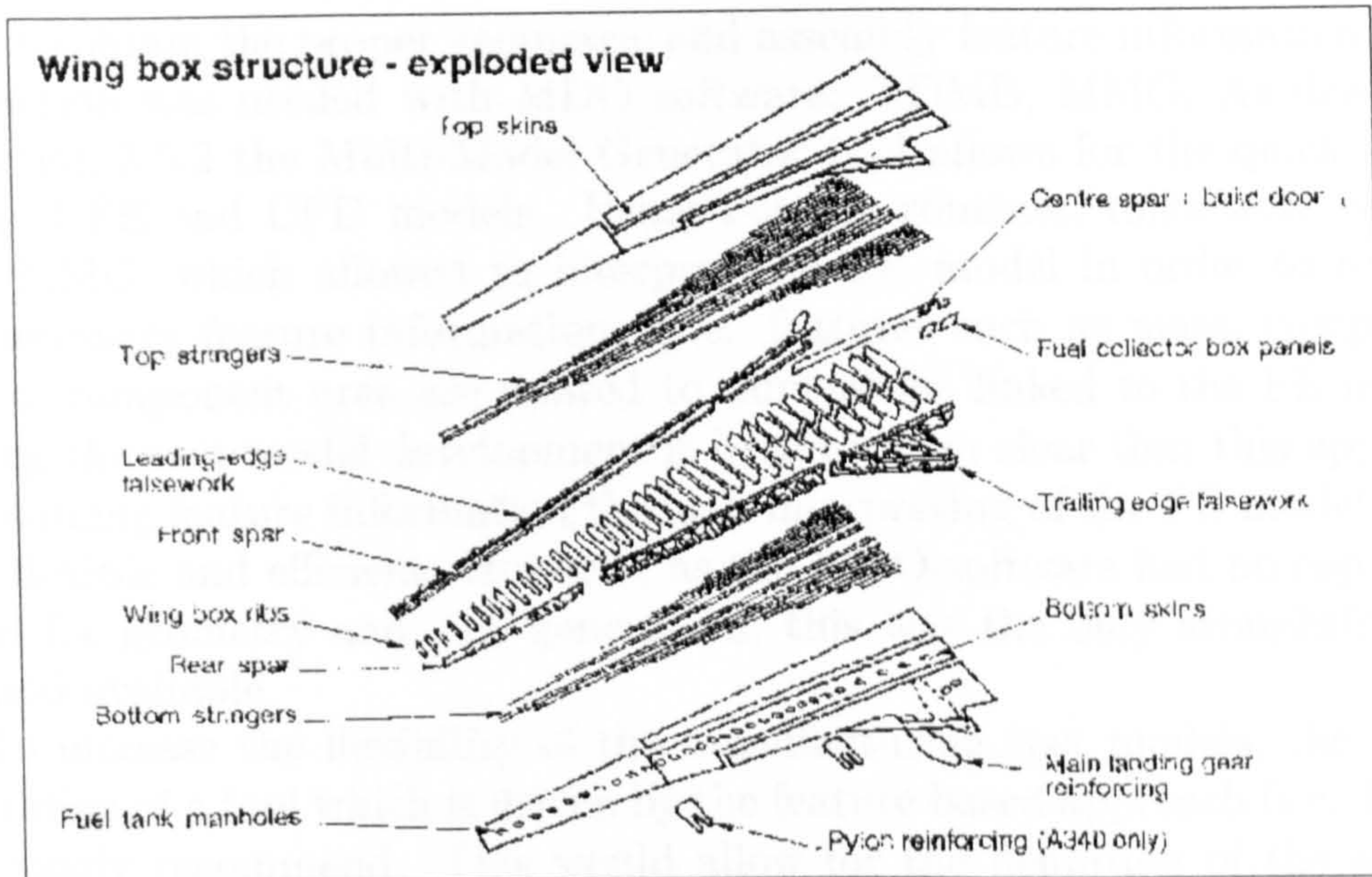


Figure 5.2: Wing box structure A330A340, exploded view.

$$RC = F_1C_1 + F_2C_2 + F_3C_3 + \dots \quad (5.1)$$

with RC=recurrent cost, F=feature and C=cost factor.

The primary aim of the models is to have a fast tool which can accurately predict and visualise the cost changes and cost trends, when going from one design to an other. For all the cost models, the cost analysis is done in two stages:

1. Calculate or assign cost factors for a reference wing. (top-down cost approach)
2. Calculate the cost of an aircraft variant using the reference cost factors. (bottom-up cost approach)

By using the Multi-Model Generator, the manufacturing cost can be calculated for a change of both external wing geometry variants (e.g. Area, Sweep, etc.) and internal wing layout (e.g. stringer pitch, rib pitch, etc.).

### 5.2.3.1 Interaction of cost models and MDO software

As mentioned earlier, a feature based approach method was developed by the author which forms the foundation of the manufacturing cost models. In



order to obtain the proper geometric and assembly feature information, close interaction was needed with MDO software: TDMB, MMG. As described in section 3.5.2 the Multi-Model Generator tool allows for the quick generation of FE and CFD models. Using Fortran routines, links were created with MMG, which allowed to interpret the FE model in order to retrieve the necessary feature information. (i.e. features such as mass, component length, component area are related to dimensions linked to the FE nodes). During the cost model development it became soon clear that this approach of obtaining feature information through interpreting of the FE model is not very flexible and efficient. However, as the MDO-software had no capability at all for geometry and rule generation, this was the only straightforward method available.

To increase the flexibility of the manufacturing cost models, the implementation of a tool which is driven by the feature based approach (i.e. ICAD) is strongly recommend. This would allow for the definition of the aircraft geometry, including manufacturing rules. From this geometry model, the finite element model and cost driver data could then be generated.

### 5.2.3.2 Prototype cost model for metal wing box

This section describes the development of a prototype cost model for a metal wing box. It was developed to demonstrate and test the interaction between the other MDO software. A reason for its coarseness was also caused by the lack of initial cost data available because of the confidential nature of cost information. This prevented the immediate development of a more accurate, detailed cost model. As mentioned before, the first stage of the cost modelling process was the definition of a product structure. For the metal cost models, a simplified product structure was created, based on the A340. Figure 5.3 describes this simplified product structure. As can be seen from the figure, the product structure exists of:

- **Skin panels.**

Machined from large, flat aluminium billets. After machining they are formed using shot peening and heat treatment techniques to give their aerodynamic shape. For the A340, top and bottom skin are made of 4 skin panels.

- **Stringers.**

Stringers are obtained from extruded aluminium sections which are machined to the particular stringer shape. After machining they are formed mechanically of by using heat treatment techniques.



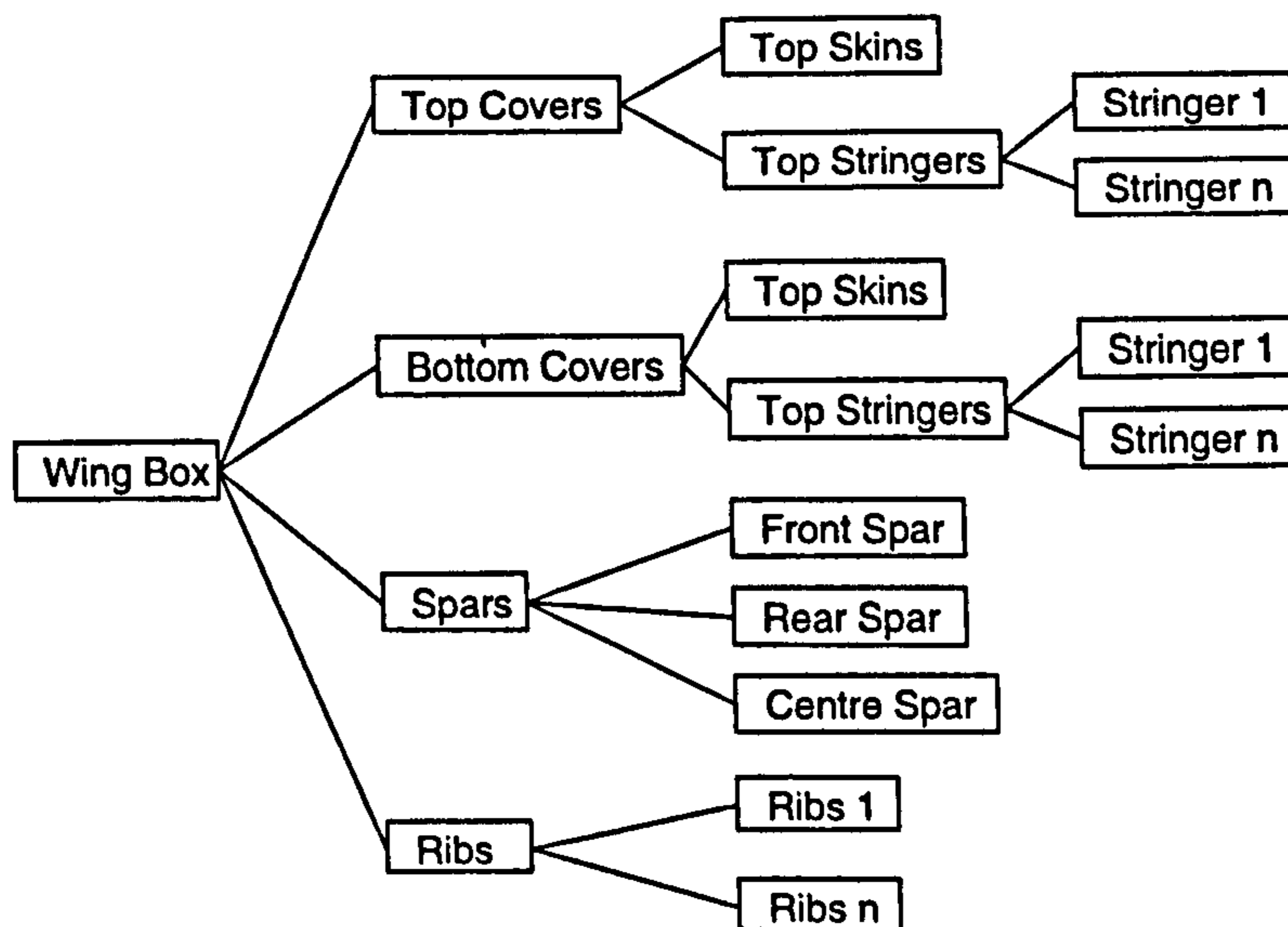


Figure 5.3: Product structure, prototype cost model.

- **Ribs.**  
Ribs (including rib feet) are machined from aluminium billets or press formed.
- **Spars.**  
Spars are machined from aluminium billets.

The wing box components are brought together in an assembly jig where all elements are being correctly positioned and connected in order to achieve the desired aerodynamic shape and structural performance. Top and bottom skin are pre-assembled in a separate jig, where skin panels have been properly aligned. Using templates, holes are drilled in both skins and stringers. After applying sealant, the stringers are bolted and riveted onto the skins. In addition to the main wing box components, other components such as joints, reinforcements and leading and trailing edge structures (e.g.: hinge ribs, track ribs, etc.) are attached. Assembly processes which occur during the A340 wing box assembly are: lifting components in&out of assembly jigs, drilling, deburring, sealing and fastening.

The prototype model did not take into account the cost of different assembly and manufacturing processes which were identified by the product structure, because of the lack of cost data. Instead, it was decided to divide the recurrent cost calculation into three main cost categories:

- Material cost.
- Detailed manufacturing cost.
- Assembly cost.

For each component of the product structure, these three recurrent costs were calculated and then added to give the total recurrent cost per component. This was done to avoid a cost estimation model purely based on weight. As mentioned earlier, weight based models do not always accurately represent the actual manufacturing cost and it may not provide accurate sensitivity data for the MDO process. In the present case the optimised weight is only one of the components in the cost calculation, as shown in the figure 5.4 below. Using the Multi-Model generator and the Technical Data Modeller and Browser rules and geometry feature information is also provided as input.

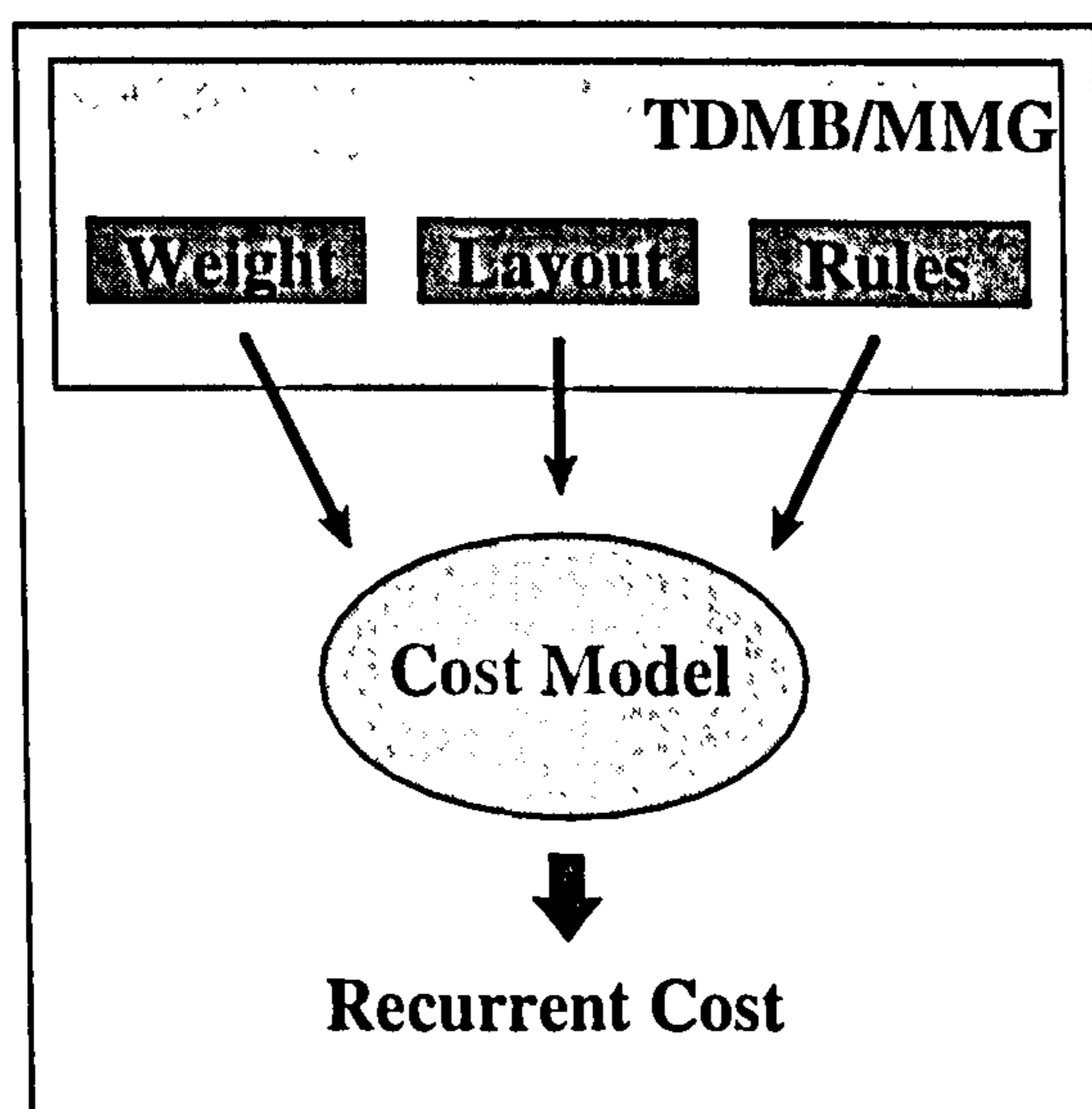


Figure 5.4: Inputs to cost model.

The following general geometry features were assumed for each cost category:

- Material: component weight (usually optimised).
- Detailed manufacturing: Component area and length.
- Assembly: Component length.



In case of the Rib assembly, for example, the component length is the sum of all the rib sides and in case of the spar assembly, the component length is the sum of all the spar lengths. The next figure (5.5) gives an overview of how the assembly lengths were calculated. By multiplying the feature information

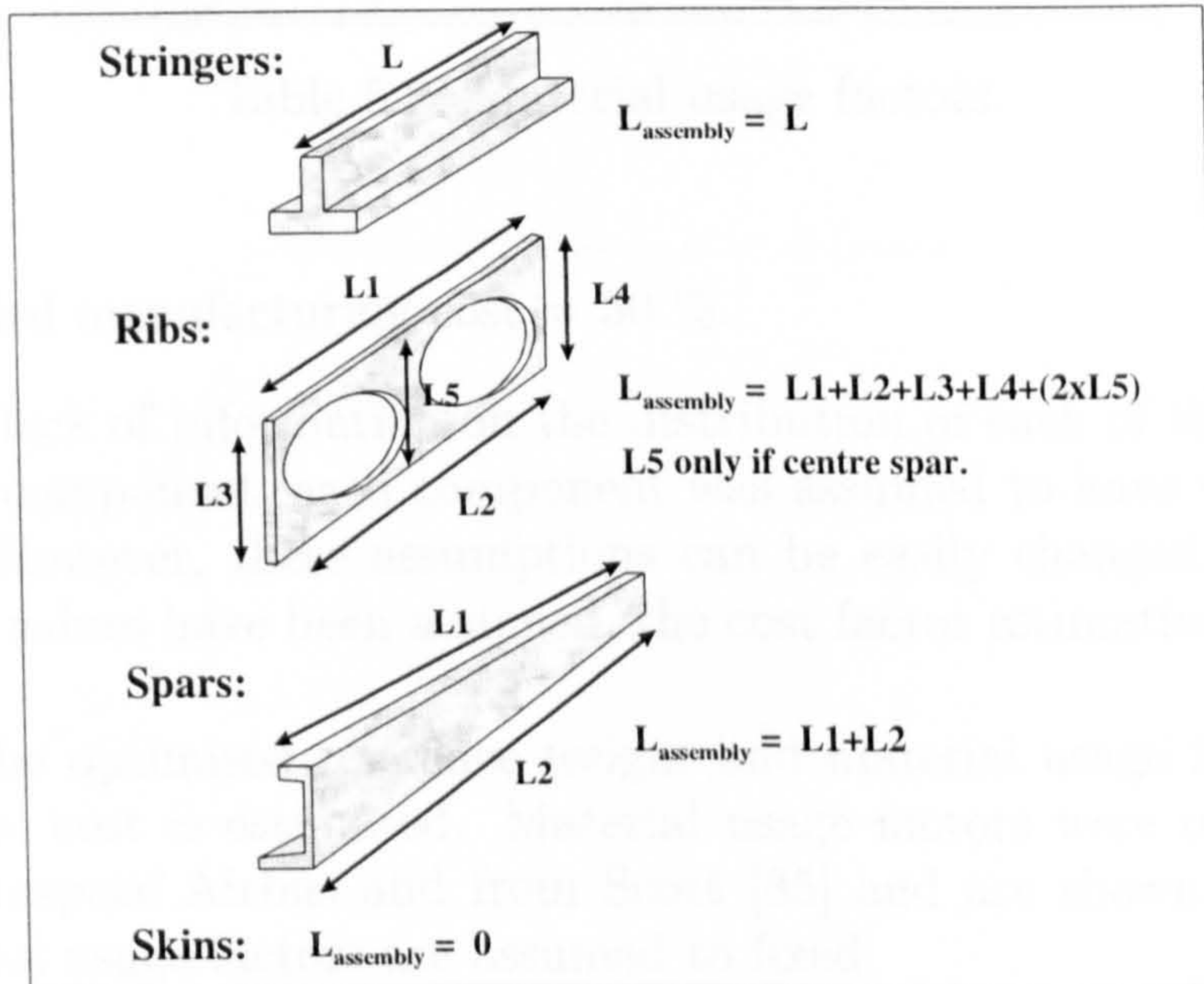


Figure 5.5: Assembly cost drivers: component length.

with cost factors for material, assembly and detailed manufacturing, the recurrent cost could be calculated. The main issue is the determination of the cost factor values. For this task, an estimation programme was developed.

For each component, the estimation of cost factors is based on a defined cost distribution between the three main cost categories. Based on literature information from the RAND corporation [83], NASA [84] and information from British Aerospace Airbus, it was identified that for aluminium wing manufacturing the material cost vary from 15 % to 40 %, assembly cost vary from 25 % to 40 % and the fabrication cost from 30 % to 40 %. A reason why the material cost can vary so much is, the fact that the material price is often a 'politically determined' price, agreed between material supplier and aircraft manufacturer. This agreed material price is again, of extremely competitive nature. The following distribution was assumed for the prototype:

- Material cost = 40 %.
- Assembly cost = 30 %.



| Component | Material Usage | Material form |
|-----------|----------------|---------------|
| Skin      | 35 %           | Plate         |
| Rib       | 7 %            | Plate         |
| Spar      | 5 %            | Plate         |
| Stringer  | 15 %           | Extrusion     |

Table 5.2: Material usage factors.

- Detailed manufacturing cost = 30 %.

Because of lack of information on the distribution of each of these cost categories per component, each component was assumed to have the same distribution. However, these assumptions can be easily changed by the user. Once these values have been assigned, the cost factor estimation process can start.

Using the optimised reference weight and material usage factors (MU), the material cost is estimated. Material usage factors were obtained from British Aerospace Airbus and from Scott [85] and are shown in table 5.2. The material usage factors are assumed to fixed.

Having calculated the material cost, the assembly and detailed manufacturing cost can now be obtained, using the cost distribution. The estimation of the detailed manufacturing cost is further refined by taking the total detailed manufacturing cost of all components (skin, rib, etc.) and by then redistributing this cost in the same way as the component weights are distributed. For example, if the spar weight contributes 15 % to the total wing box weight, the detailed manufacturing cost will be 15 % of the total detailed manufacturing cost.

The reference cost factors are obtained after dividing the reference cost by the cost driver (geometry feature) value. With a reference cost of 500 \$ and a cost driver of  $50 m^2$ , for example, the reference cost factor is  $10 \$/m^2$ . Figure 5.6 gives a schematic summary of how the reference cost can be calculated. No learning curve was taken into account for the cost calculation. However, an adjustment to simulate the cost of the wing box after a certain number of wings have been manufactured can be easily added.



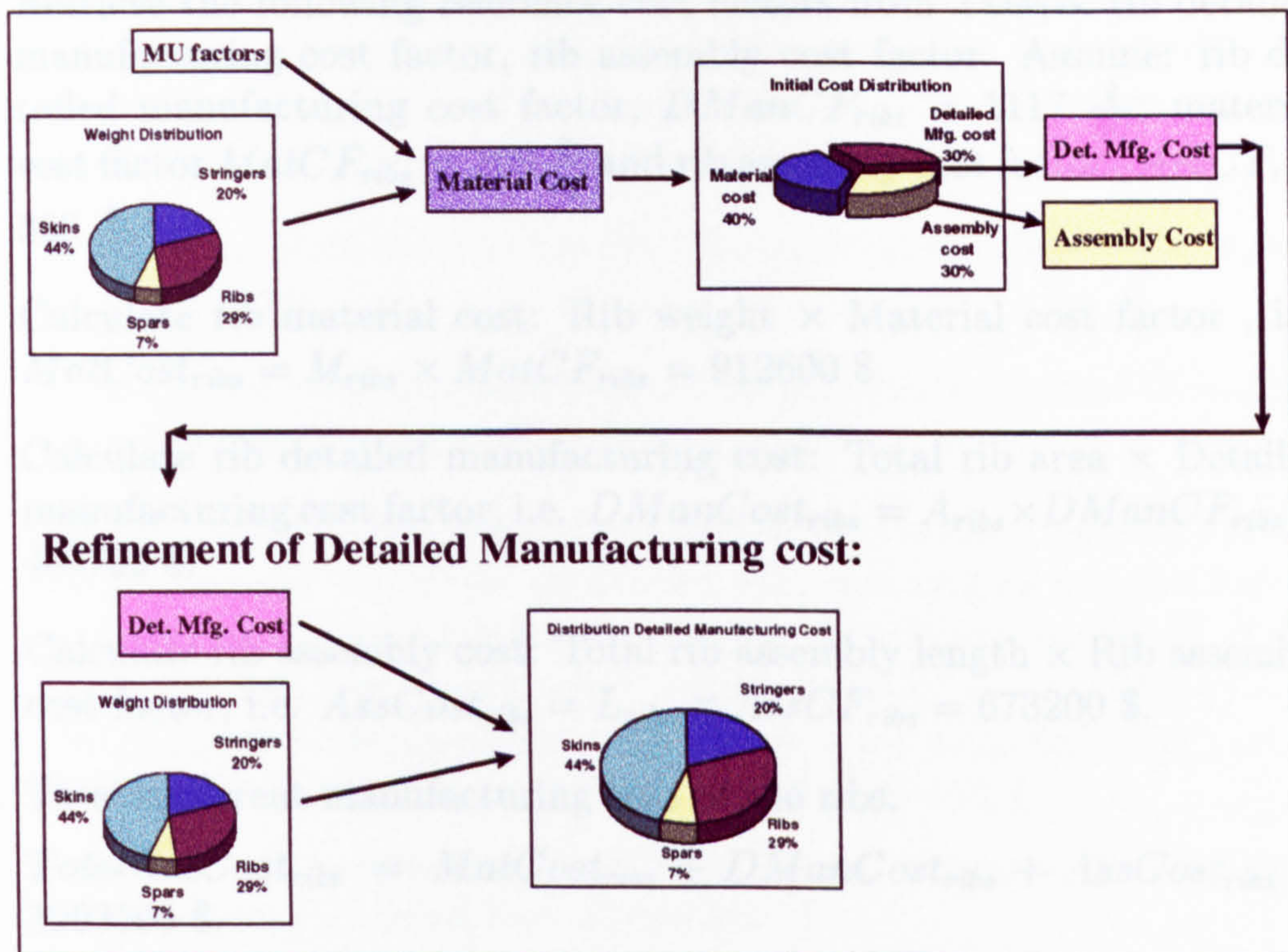


Figure 5.6: Calculation of reference cost.

### 5.2.3.3 Example

This section demonstrates how the recurrent manufacturing of a rib is being calculated for the simplified metal cost model. The following sequence is undertaken:

1. Run the Multi-Model Generator for a wing variant. During the generation of this wing variant, all necessary feature information is also being retrieved from the Finite Element model and stored in the Technical Data Modeller and Browser (TDMB). For a rib the following information is calculated and stored: rib height, rib width, rib area, total assembly length of all the ribs (i.e. see figure 5.5) and total rib area. Assume total rib assembly length,  $L_{ribs} = 792 \text{ m}$  and total rib area,  $A_{ribs} = 365 \text{ m}^2$ .
2. Perform structural optimisation. After the optimisation, the optimum rib weight is stored in the TDMB. Assume optimised rib weight,  $M_{ribs} = 8450 \text{ kg}$ .



3. Retrieve the following reference cost factors from TDMB: rib detailed manufacturing cost factor, rib assembly cost factor. Assume: rib detailed manufacturing cost factor,  $DManCF_{ribs} = 1117 \frac{\$}{m^2}$ ; material cost factor  $MatCF_{ribs} = 108 \frac{\$}{kg}$  and rib assembly cost factor,  $AssCF_{ribs} = 850 \frac{\$}{m}$ .
4. Calculate rib material cost: Rib weight  $\times$  Material cost factor, i.e.  $MatCost_{ribs} = M_{ribs} \times MatCF_{ribs} = 912600 \$$ .
5. Calculate rib detailed manufacturing cost: Total rib area  $\times$  Detailed manufacturing cost factor, i.e.  $DManCost_{ribs} = A_{ribs} \times DManCF_{ribs} = 407705 \$$ .
6. Calculate rib assembly cost: Total rib assembly length  $\times$  Rib assembly cost factor, i.e.  $AssCost_{ribs} = L_{ribs} \times AssCF_{ribs} = 673200 \$$ .
7. Total recurrent manufacturing cost of the ribs:  
 $TotalRecCost_{ribs} = MatCost_{ribs} + DManCost_{ribs} + AssCost_{ribs} = 1993505 \$$ .
8. Store results in TDMB.

The reference cost factors which were used for the calculation, were defined by the user or have been calculated using the estimation process which was described by figure 5.6. To estimate the reference cost factors of a rib, the following process is undertaken:

1. Assume an initial manufacturing cost distribution for the rib, i.e. Material cost = 40 %, Assembly cost = 30 %, Detailed manufacturing cost = 30 %.
2. Perform structural optimisation on a reference aircraft wing and obtain the optimised reference weight for the ribs.
3. Assume a value for material usage factor of a rib, i.e.  $MU_{ribs} = 7 \%$ . (see table 5.2)
4. Assume a value for the cost of the material, i.e.  $MC_{ribs} = 7.56 \frac{\$}{kg}$ .
5. Using the optimised reference rib weight, the material usage factor and the material cost, the reference rib material cost can be calculated. Assuming the reference rib weight = 8301 kg, the reference rib material cost is:  $RefMatCost_{ribs} = 896508 \$$ .



6. With the initial manufacturing cost distribution which was assumed for the rib, the reference detailed manufacturing cost and the reference detailed assembly cost are calculated for the rib. Hence we obtain:  $RefDManCost_{ribs} = 672381$  \$ and  $RefAssCost_{ribs} = 672381$  \$.
7. The previous steps are repeated for the other wing components: spars, skins, stringers.
8. Calculate the total reference detailed manufacturing cost of all components. Assume  $TotalRefDManCost = 1364040$  \$.
9. Use the total reference detailed manufacturing cost and redistribute this cost in the same way as the optimised reference weights are distributed. Assuming that the reference rib weight contributes for 29 % of the total optimised wing weight (see weight distribution pie of figure 5.6), the improved estimate of the reference detailed manufacturing cost is:  $NewRefDManCost_{ribs} = 1364040 \times 0.29 = 395571$  \$.

Reference cost values are now defined for the assembly, detailed manufacturing and material cost of the rib.

10. Using the rib feature information of the reference wing (i.e. the optimised rib weight, the total rib area and the total rib assembly length) the reference rib cost factors are now calculated by the dividing the reference cost by the respective reference feature. Assuming a reference weight of 8301 kg, a reference assembly length of 791 m and a reference rib area of  $354 \text{ m}^2$  the reference cost factors are:  $MatCF_{ribs} = \frac{RefMatCost_{ribs}}{8301} = 108 \frac{\$}{kg}$ ;  $DManCF_{ribs} = \frac{NewRefDManCost_{ribs}}{354} = 1117 \frac{\$}{m^2}$ ;  $AssCF_{ribs} = \frac{RefAssCost_{ribs}}{791} = 850 \frac{\$}{m}$ .

#### 5.2.3.4 Detailed cost model for metal wing box

This section describes how the prototype model can be expanded to a more detailed low-level model. Such a model has not been implemented as part of this thesis, mainly because this thesis focused on the integration of manufacturing cost into an MDO process and not on the development of high level cost models. Another reason was the fact that CASA and British Aerospace showed more interest in the development of a composite cost model. While British Aerospace has already a lot of experience on the field of metal cost modelling, composite cost modelling is a relative new area for them.

A detailed metal cost model would follow more closely the product structure shown in figure 5.1 and would take into account the different detailed manufacturing and assembly processes which occur. The three main cost

| Detailed Manufacturing |                |
|------------------------|----------------|
| Skins                  | Milling        |
|                        | Shot peening   |
|                        | Heat treatment |
| Stringers              | Milling        |
|                        | Forming        |
| Spars                  | Milling        |
| Ribs                   | Milling        |
|                        | Press forming  |

Table 5.3: Manufacturing processes for aluminium wing box.

categories could still be used, but for each wing box component it would now include a detailed breakdown of the processes. Rib assembly, for example would contain processes such as lift, drill, debur and fasten. Table 5.3 gives an overview of the manufacturing processes per component and table 5.4 shows the assembly processes based on the A340.

Cost drivers and cost factors could be assigned for these different processes. This allows the designer to make trade-off studies between different manufacturing methods. When assessing the manufacturing of a large wing, such as the A3XX wing, additional issues need to be considered such as:

- **Wing joint:** It would be impossible to transport the wing in its whole by the Beluga transport aircraft; hence a joint would be necessary. Where will be the wing joint location? Between inboard and outboard engine or at the outboard engine?
- **Composite outer wing:** A wing joint is needed. To save weight, would it be feasible to make the outboard wing of composites?
- **Number of skin panels and location:** Current A340 has 4 top and 4 bottom skin panels (see figure 5.2 on page 134). Do we need more?
- **Assembly skin-stringers:** Current assembly of stringers to skin is an elaborate task, involving a lot of lifting and putting skins, stringers and templates into place. Can this process be speeded up?

The current software (with prototype cost model) is capable of investigating some of these issues. Some of them will be discussed in the next section.



| Assembly processes for aluminium wing box. |                            |
|--|----------------------------|
| Skins                                      | Lift                       |
|  | Drill (skin-skin)          |
|  | Debur                      |
|  | Bolt (skin-skin)           |
|  | Add Reinforcings           |
| Stringers                                  | Lift                       |
|  | Drill (stringer-skin)      |
|  | Debur                      |
|  | Bolt (stringer-skin)       |
| Spars                                      | Lift                       |
|  | Drill (spar-skin)          |
|  | Debur                      |
|  | Bolt (spar-skin)           |
| Ribs                                       | Lift                       |
|  | Drill (rib-spar; rib-skin) |
|  | Debur                      |
|  | Bolt (rib-spar; rib-skin)  |
| Sealing                                    |                            |
| Lift assembled wing                        |                            |

Table 5.4: Assembly processes for aluminium wing box.

### 5.2.3.5 Composite cost model

The composite cost model is based on the current A340 horizontal CFRP tail plane manufacturing and assembly methods which are used by its manufacturer CASA. A similar product model as for metal was followed and can be seen in figure 5.7.

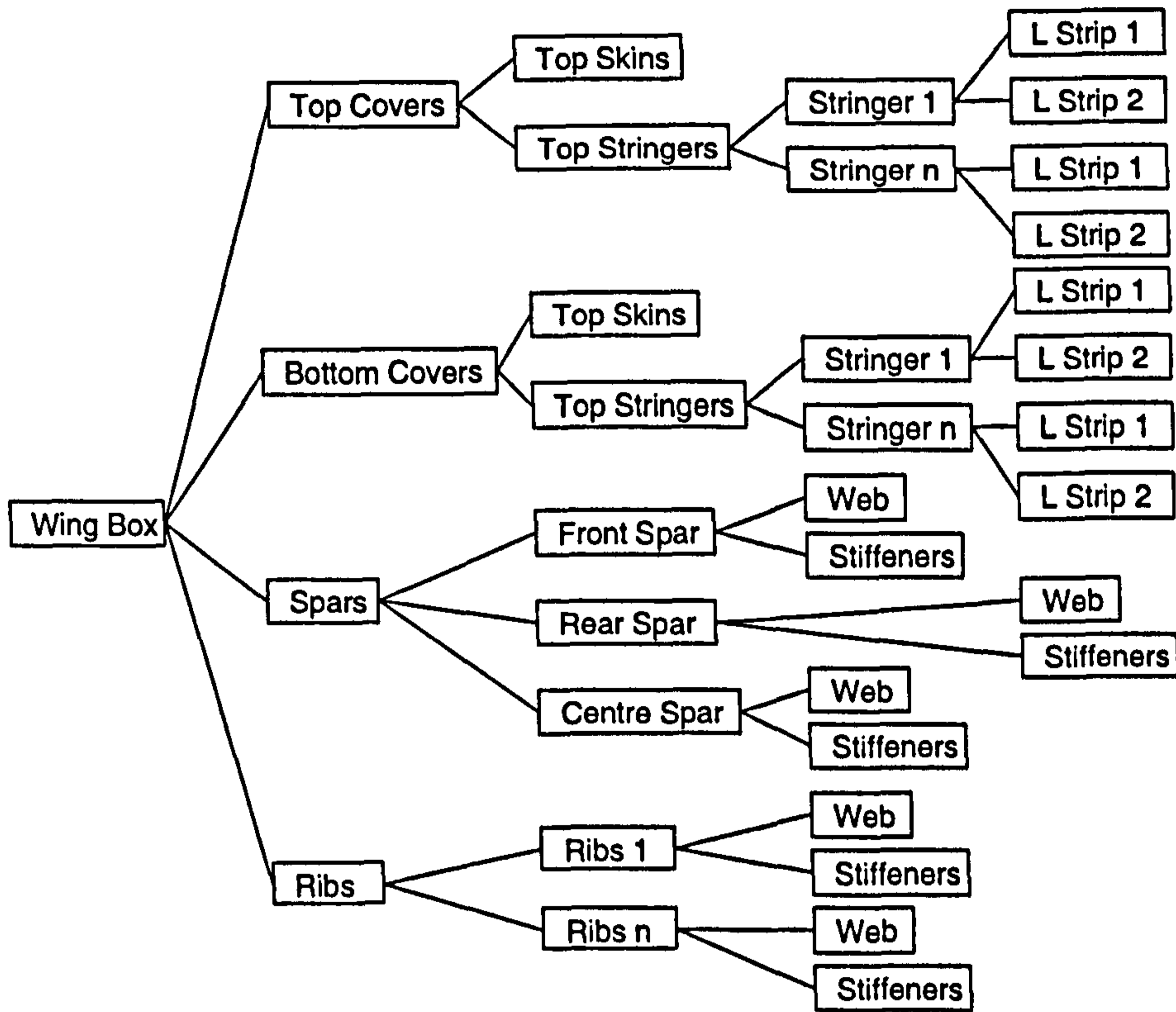


Figure 5.7: Product structure, composite cost model.

Based on current manufacturing and assembly methods, a composite wing for the A3XX could be made as follows:

- **Skin - Stringer panels**

Skin and stringer composite sheets are made using automatic tape lay-up. In this way the skin thickness can be tailored to obtain more optimum material usage. The stringer sheets are cut into strips and lengths appropriate to specific stringers. In a next process, these strips are hot formed (at  $\pm 70^{\circ}\text{C}$  with high pressure) into L-shapes. Two L-shaped strips are combined to form one T-stringer. In a last process, stringers are set up on the skin panels and the whole component is cured into the autoclave. The co-curing is a very critical process, as



pressure needs to be applied evenly over all stringers and skin. It is also a very expensive process because it requires a lot of tools (bagging, aluminium bars, etc.) and set-up time. In future one could consider co-bonding of skins and stringers, with the skin web being cured in advance. After removal from the autoclave, components are subjected to elaborate inspection and, possibly, repair.

- **Spars**

Spars are cut from composite sheets which have been produced using tape lay-up. Several cut sheets are put on top of each other, to obtain the required spar web thickness, and are then cured. Spar stiffeners are cut into strips from sheets made by tape lay-up. The strips are then hot formed into L-shape. The stiffeners are cut into their required length and co-bonded with the spar web. Approximately two stiffeners are needed between each rib. After co-bonding, the whole spar is inspected.

- **Ribs**

Ribs are made of fabric composite material and are made by hand lay-up. After the lightening holes have been cut out, the fabric is hot formed to shape the lightening holes. This is followed by curing and cutting of the stringer cut-outs. The rib stiffeners are made in the same way as the spar stiffeners. After being cut to the right size, they are co-bonded to the rib webs. Approximately one rib stiffener is used between each two stringers. In the same co-bonding process, the L-shaped rib-spar fittings are also put in place. For future rib manufacturing, one would like to produce the whole rib in 1 curing process.

Based on the current assembly of the composite A340 horizontal tail plane the assembly of the composite components is done using fasteners, as for a metal wing. In future applications, however, it would be better to exploit the advantages of composites by using more revolutionised assembly techniques, such as composite stitching.

The composite model takes into account aspects relating to the manufacturing of large wings such as the A3XX. These include component size (e.g.: autoclave size), curing method, manufacturing method etc. The size of a hot forming machine is build/customised for each project, in order to maximise its use (e.g.: 6...7m long, 2...3 m wide). The capacity of the hot forming machine is a main cost driver. Drilling operations are avoided if possible, as they are much more expensive than for metal. Drilling time strongly depends on the size of the hole. The time to drill a big hole is much higher than for a small one, as it requires several drilling operations, each with increasing drill diameter. An increase in drilling time of about 20 % to 25 % is estimated



in comparison with metal drilling operations. The cost of the curing process is mainly driven by the element complexity and machine capacity, as this influences the component set up and the bagging time. Current autoclave capacity at CASA is a useful length of 90 m with a diameter of 5 m, hence it is currently possible to manufacture large wing components.

Again three main cost categories are included; material, detailed manufacturing and assembly costs have been defined. For each component the cost category contains a breakdown of the required production processes. The following detailed manufacturing processes are considered:

- Automatic tape lay-up.
- Hand lay-up.
- Hot forming: set-up, cure, extract.
- Curing: bagging, set-up, cure, debuggng.
- Co-bonding.
- Cut and debur.

The selected Geometric features which drive the component manufacturing cost are: weight, ply stacking and % fibre material per stack, length, area, number of components, number of cut-outs and lightening holes. Table 5.5 gives an example of the processes and cost drivers which have been considered for the calculation of the detailed manufacturing of a composite wing.

Assembly cost includes processes such as part fit up, shimming, hole drilling, fastening, sealing, inspection and repair etc. The selected assembly cost drivers are geometric and assembly features, such as area and number of fasteners. Close co-operation with CASA allowed for cost feature and cost driver identification and for an intensive iterative process of model updating and refinement in order to make it more realistic. Although the model might be based on some conventional composite manufacturing techniques, the model allows for expansion to incorporate more advanced composite manufacturing methods such as resin transfer moulding and composite stitching.

## 5.2.4 Results cost models

### 5.2.4.1 Prototype metal cost model

Although the metal cost model is very coarse and deals with high level costs, it proved to be useful in quickly visualising cost trends to the designer. Using



| Detailed Manufacturing |                      |                          |
|------------------------|----------------------|--------------------------|
| Component              | Process              | Feature                  |
| Skins                  | Lay-up               | Weight                   |
|                        |                      | % fibre material per ply |
| Stringers              | Lay-up               | Weight                   |
|                        |                      | % fibre material per ply |
|                        | Cut & Debur          | Length                   |
|                        | Hot forming          | Number of components     |
|                        |                      | Number of cycles         |
| Cure                   | Area                 |                          |
| Spar Web               | Lay-up               | Weight                   |
|                        |                      | % fibre material per ply |
| Spar Stiffeners        | Cut & Debur          | Length                   |
|                        | Cure                 | Area                     |
|                        | Lay-up               | Weight                   |
|                        |                      | % fibre material per ply |
|                        | Cut & Debur          | Length                   |
|                        | Hot forming          | Number of components     |
|                        |                      | Number of cycles         |
|                        | Cobond               | Area                     |
| Rib Web                | Hand lay-up          | Weight                   |
|                        |                      | % fibre material per ply |
| Rib Stiffeners         | Cut lightening holes | Length                   |
|                        |                      | Number of holes          |
|                        | Hot forming          | Number of components     |
|                        |                      | Number of cycles         |
|                        | Cure                 | Area                     |
|                        | Cut stringer cutouts | Number of cutouts        |
|                        | Lay-up               | Weight                   |
|                        |                      | % fiber material per ply |
|                        | Cut & Debur          | Length                   |
|                        | Hot forming          | Number of components     |
|                        |                      | Number of cycles         |
| Cobond                 | Area                 |                          |

Table 5.5: Manufacturing processes and drivers for composite wing box.



the cost factor estimation programme (see section 5.2.3.2 on page 135) cost factors were calculated for a datum wing model and then used for all the subsequent cost calculations. As mentioned earlier, the recurrent manufacturing cost is divided into three main cost categories: assembly cost, detailed manufacturing cost and material cost. Figure 5.8 shows the distribution of these cost categories for each component of the reference wing.

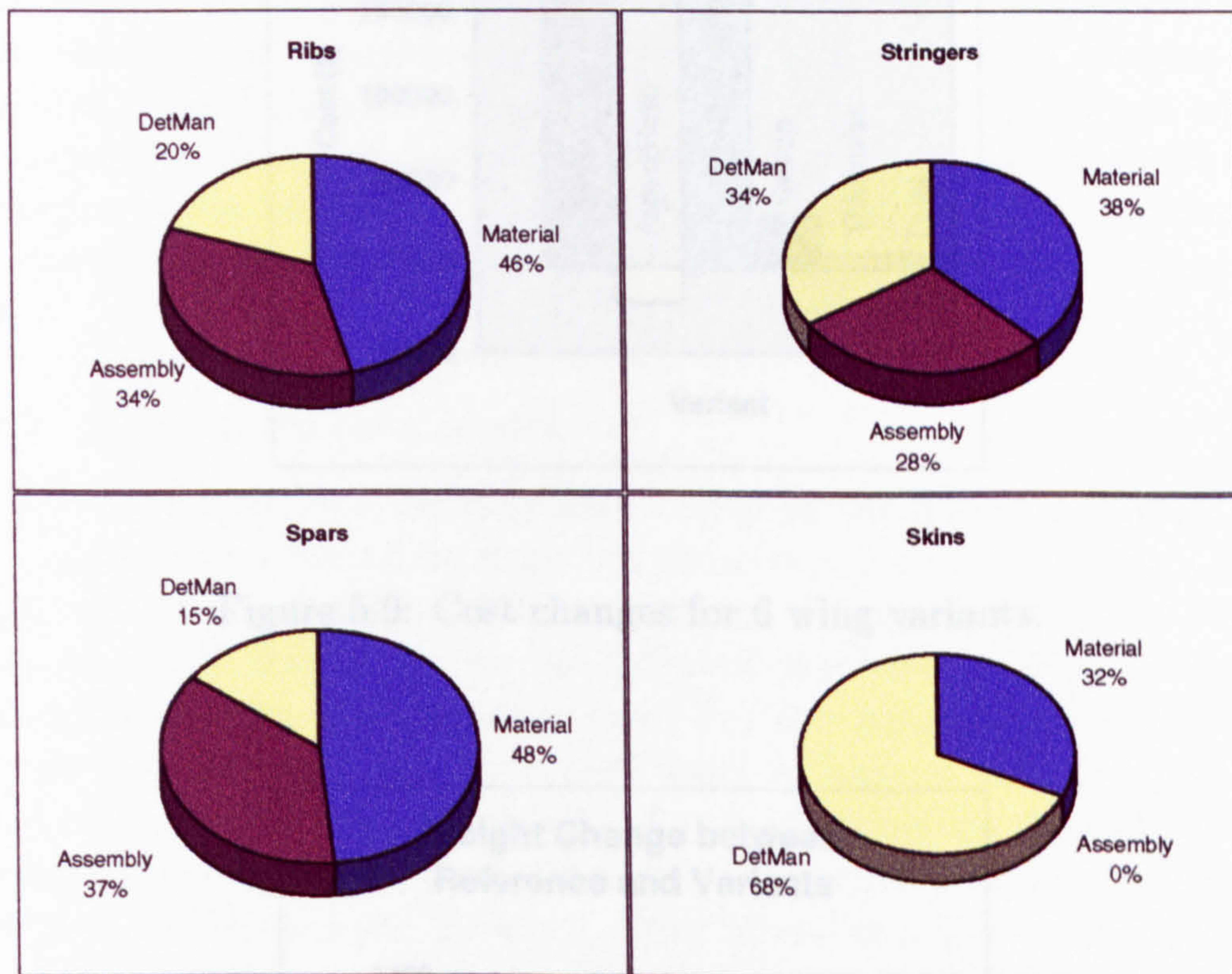


Figure 5.8: Cost breakdown reference model

These percentages can only be treated as indicative as they strongly depend on input values such as material usage factors and cost factors. For the skins there is no assembly cost, because the prototype model assumes that all assembly which involves skins is taken into account by the other components.

An example of how the prototype model can be used to predict cost trends is demonstrated for the six primary variants described in section 3.6.2 on page 65. Costs were calculated for a + 3% change in this variants. The optimised NASTRAN weights from table 4.7 on page 109 were used as input to the cost model. Figures 5.9 and 5.10 show the cost and weight changes of the variants with respect to those for the reference wing.

It can be noted that change in wing area and the rear spar position



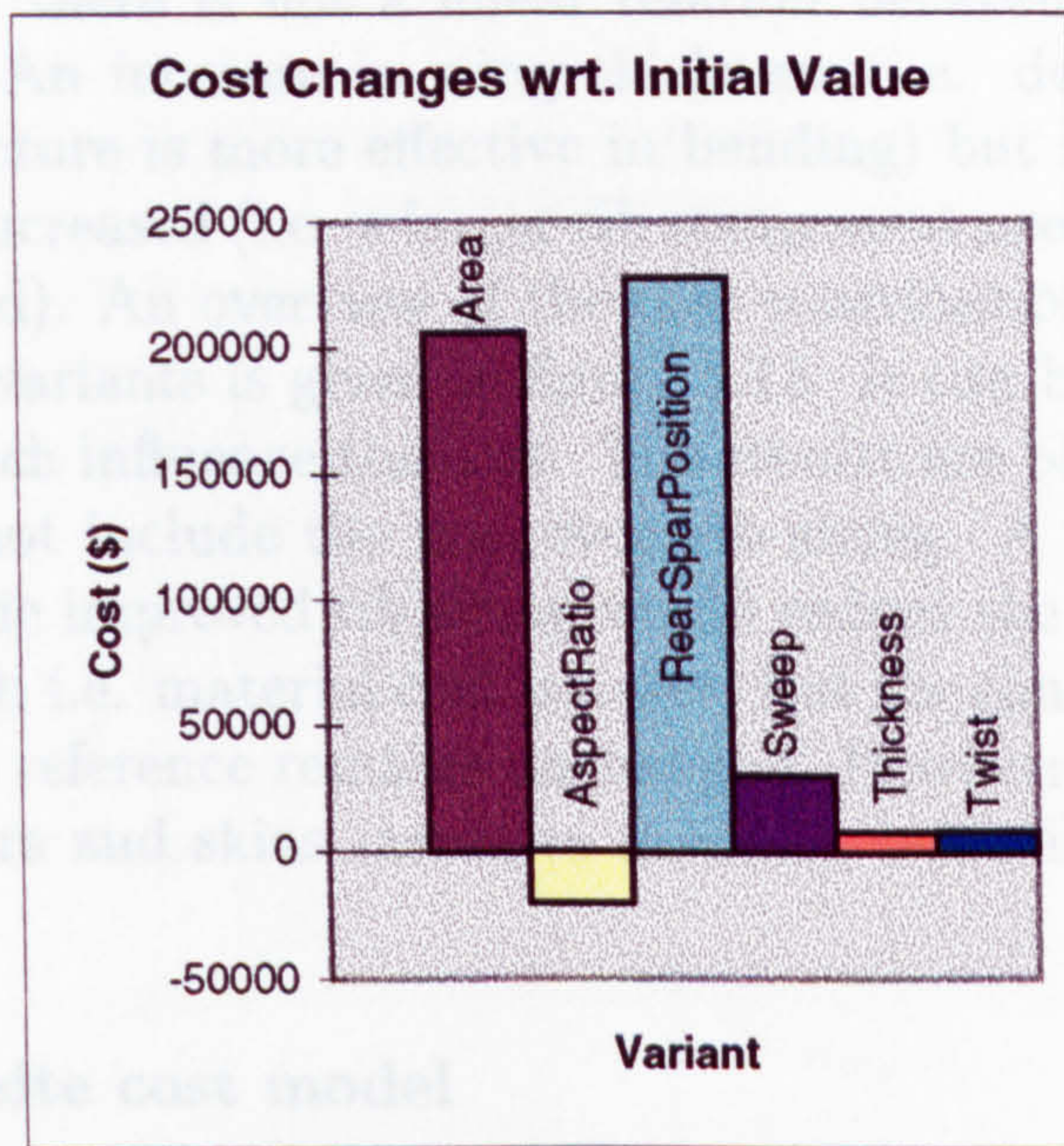


Figure 5.9: Cost changes for 6 wing variants.

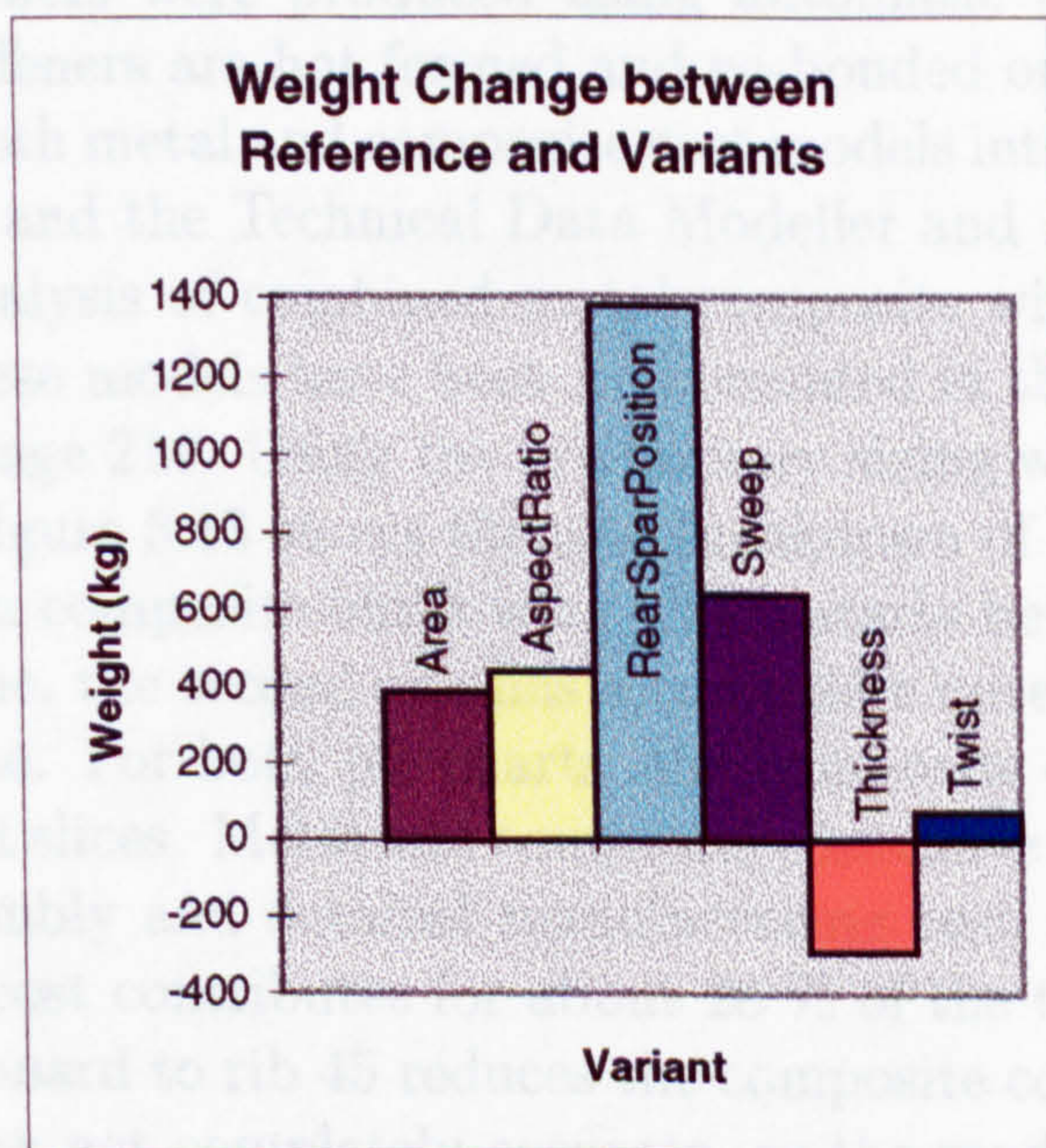


Figure 5.10: Weight changes for 6 wing variants.



have the biggest influence on cost. With respect to the thickness variant it is also clear that there is not a linear relation between weight and cost increase/decrease. An increase in wing thickness (i.e. depth) reduces the weight (as the structure is more effective in bending) but increases the cost as the rib area has increased (i.e. a larger rib component needs to be manufactured and assembled). An overview of the cost contribution per component, for the six aircraft variants is given in figure 5.11. It can be seen that it are mainly the ribs which influence the cost. The results are based on optimised weights which do not include the improved rib sizing. A brief analysis has shown that using the improved rib sizing would reduce the actual cost of the ribs (as their weight, i.e. material cost is lower) but the change of the rib cost with respect to the reference remains unchanged. However, the contribution from stringers, spars and skins increases in value, but still shows the same trends.

#### 5.2.4.2 Composite cost model

The composite model has initially been tested on the full aircraft wing. As described in section 5.2.3.5 on page 145 the composite model is much more process oriented and is able to calculate the cost of separate processes. Figure 5.12 shows a cost breakdown for the different processes involved in the detailed manufacturing of a composite spar. For this example it assumed that spar web and stiffeners were produced using automatic tape lay-up. The webs are cured, stiffeners are hot formed and co-bonded on the webs.

Integration of both metal and composite cost models into the Multi Model Generator (MMG) and the Technical Data Modeller and Browser (TDMB) allowed for cost analysis of combined metal-composite wings (hybrid). An overview of how these models have been implemented in the TDMB is given in appendix B on page 214. Using the preliminary sizing weights as input to both cost models, figure 5.13 shows the cost breakdown of two hybrid wings. The first wing has a composite outer wing which starts between the inboard and outboard engine, the second one has a composite outer wing starting at the outboard engine. For both pie charts, the composite cost contributions are shown as the cut slices. Metal and composite costs have been broken down into material, assembly and detailed manufacturing cost. With joint at rib 36, the composite cost contributes for about 28 % of the total cost. Moving the joint more outboard to rib 45 reduces the composite cost contribution to 18 %. The costs are not completely accurate, as the model did not include the costs and associated complexity of the actual metal-composite joint. In addition, if one would like to perform a detailed cost trade-off between metal and composites, one has to take into other cost issues such as maintenance



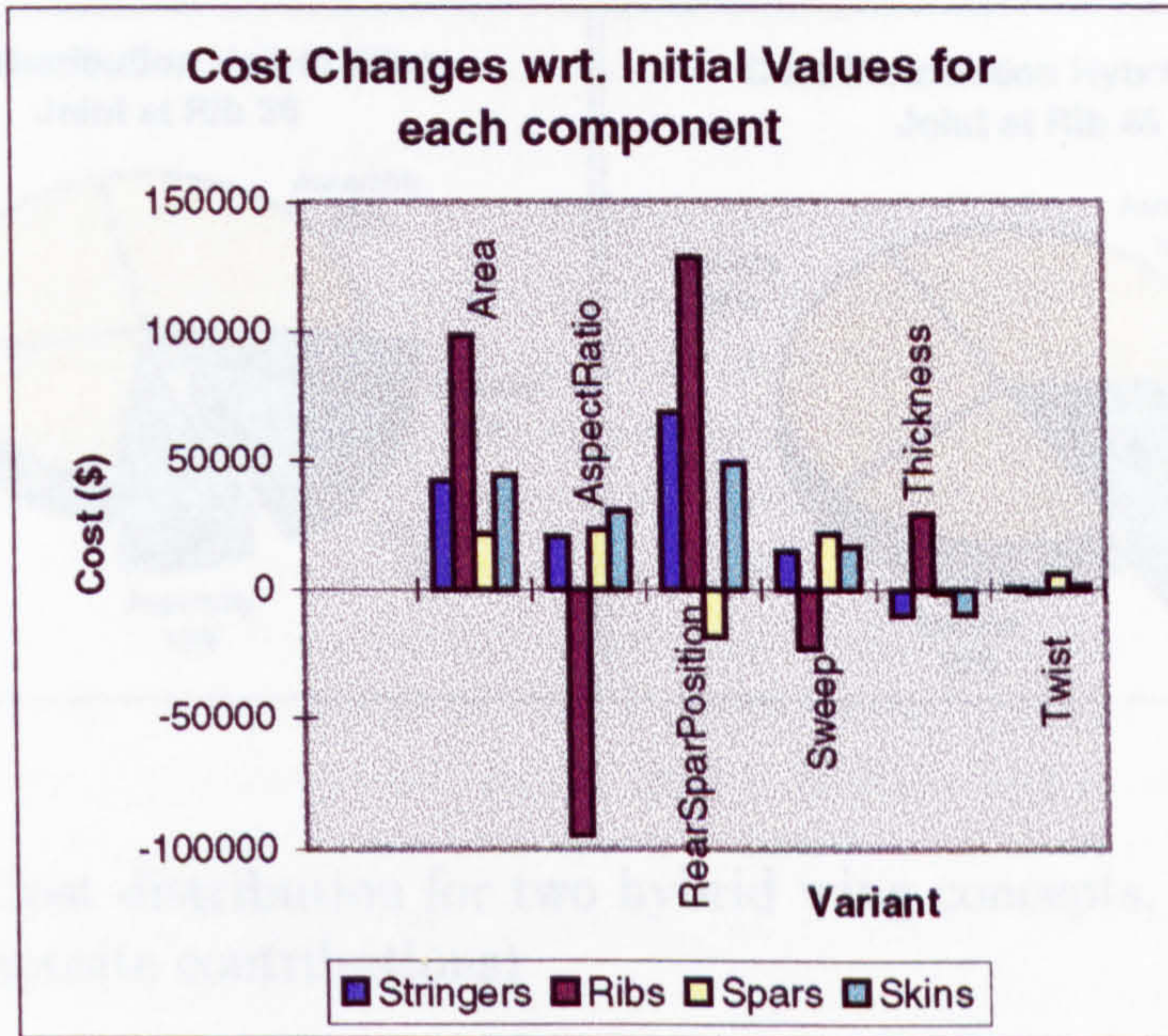


Figure 5.11: Cost changes per component for 6 wing variants.

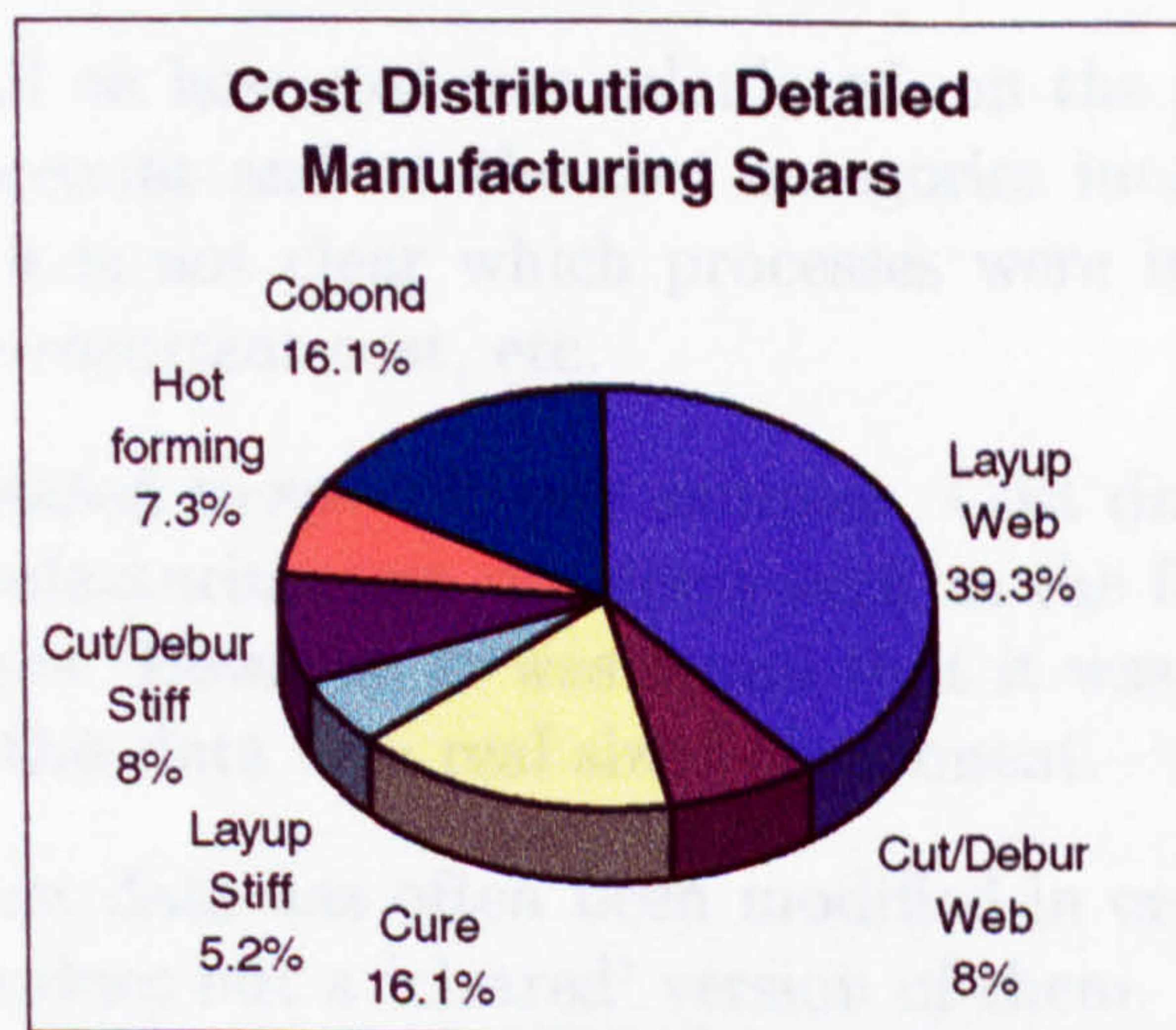


Figure 5.12: Cost distribution detailed manufacturing spar.



cost, repair cost etc.

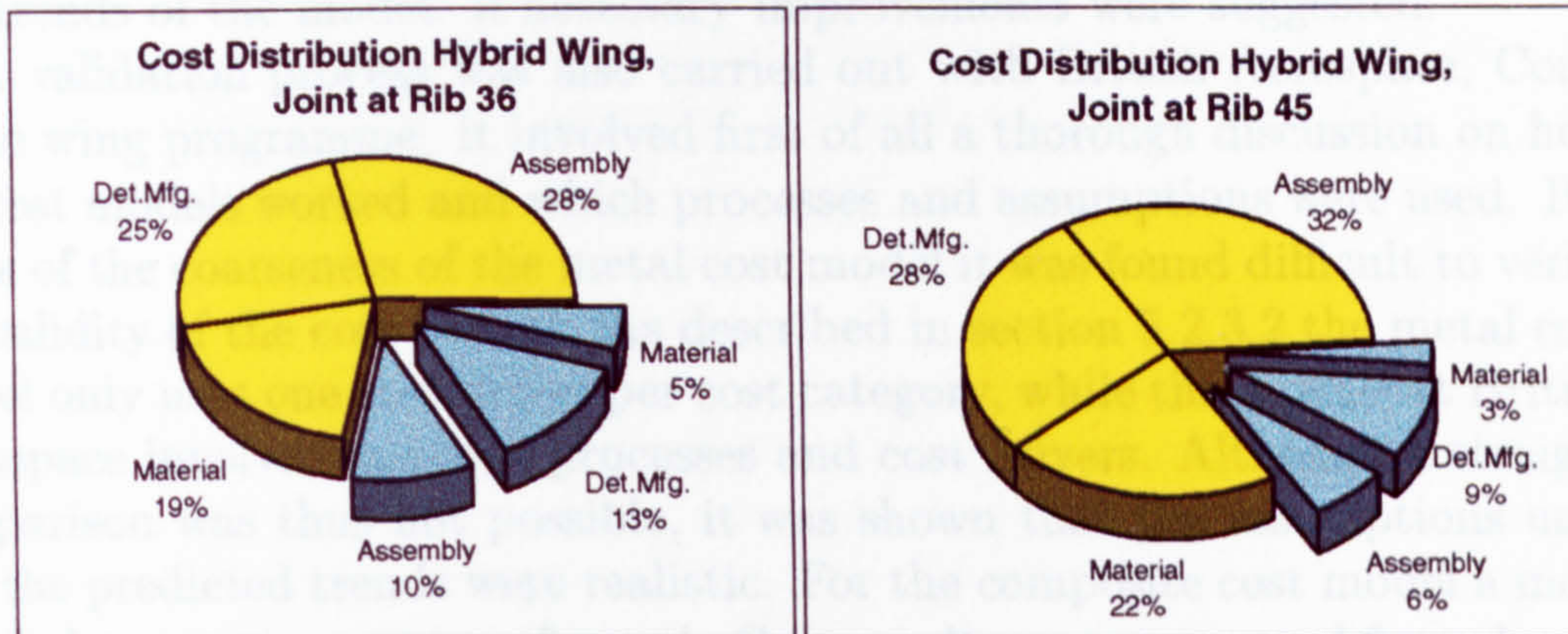


Figure 5.13: Cost distribution for two hybrid wing concepts. (sliced components are composite contributions)

### 5.2.5 Validation cost models

Because of the high confidential nature of cost, validation of the cost models was not a straightforward task. Although a fair amount of cost information was found in literature, it could often not directly be employed for validation purposes, because of the following reasons:

- Lack of detail on how cost was calculated, on the processes that were taken into account and on the cost categories involved. Cost data is shown, but it is not clear which processes were included, whether it includes non-recurrent cost, etc.
- Cost data related to specific test samples. Cost data can be found for specific manufacturing test problems such as the fabrication of a part of a skin panel. However, it was found that it was difficult to directly extrapolate this data to a real sized component.
- Published cost data has often been modified in order not to show the actual cost values but a 'cleared' version of them.
- Cost data and method applied is clearly explained but cannot be used directly as different assembly or manufacturing procedures were used.

A proper validation of the models would only be possible if the actual manufacturers were involved. The development of the composite cost model,



involved an iterative process with CASA. In this process cost results were sent to CASA and in return feedback was given on the general weaknesses and trends of the model. If necessary improvements were suggested.

A validation process was also carried out with British Aerospace, Composite wing programme. It involved first of all a thorough discussion on how the cost models worked and which processes and assumptions were used. Because of the coarseness of the metal cost model it was found difficult to verify the validity of the cost factors. As described in section 5.2.3.2 the metal cost model only uses one cost driver per cost category, while the models at British Aerospace involved multiple processes and cost drivers. Although a straight comparison was thus not possible, it was shown that the assumptions used and the predicted trends were realistic. For the composite cost model a more detailed comparison was performed. Cost results were generated for a datum problem using the composite cost model and a British Aerospace model and the results were compared. It became soon clear, however, that a direct comparison between both models was difficult, as different processes were part of different cost categories and because the British Aerospace model was of a higher level of detail. It includes many more processes such as preparing jigs, cleaning of components, lifting of components etc. and also uses different or more cost drivers. In the current composite cost model for example, the cost driver for the cut and debur process is the component length while in the British Aerospace model the cutting costs are a function of component length and component thickness. Another problem was that most of the British Aerospace data was based on A321. Nevertheless, it was possible to compare individual processes and to check whether parts of the cost data had the right order of magnitude. In a final validation exercise, British Aerospace cost factors (instead of those from CASA) were used as input for the composite cost model.

### 5.3 Conclusions

A prototype metal and more detailed composite model were developed. Although the Composite model is much more detailed than the metal model, as it takes into account different manufacturing processes, it is still of a high level nature. Validation of the models has shown that the overall cost predictions and especially the cost changes are meaningful.

The objective of this cost modelling effort was not to generate an extremely accurate low level cost model, but to demonstrate that cost models can be used in an MDO environment to:

- Give fast, realistic cost estimates.

- Visualise cost changes and trends to the designer.

Having this demonstrated, further work could be carried out in refining and expanding the current cost models to include more processes and to allow for investigation of different assembly and manufacturing concepts.



# Chapter 6

## A methodology for solving the global MDO-wing problem.

This chapter discusses an approach for solving the MDO wing problem by incorporating mass, drag and manufacturing cost. Based on the items discussed in previous chapters, it describes how the different disciplines can be integrated, raises problems and assumptions associated with this integration and gives an overview of results.

### 6.1 Aspects of MDO integration

As described in paragraph 3.6.1 on page 63 the full Direct Operating Cost Model, was not used as part of the MDO study, but was replaced by a simplified Direct Operating Cost Model. This model takes into account changes of weight and drag only. As discussed earlier the reduced DOC formulation is given by:

$$\Delta DOC = \sum_{i=1}^n w_1 \frac{\Delta M}{\Delta p_i} \Delta p_i + \sum_{i=1}^n w_2 \frac{\Delta D}{\Delta p_i} \Delta p_i \quad (6.1)$$

With:

$i$  = aircraft variant

$p_i$  = parameter being varied

$w_i$  = weighting factor

$\Delta M = (m_{opt} - m_{opt+i})$

$\Delta D = (drag - drag_i)$

The weighting factors were defined, based on data from similar sized aircraft (550 seats) and typical mission length (4000 nm; i.e. 7408 km) it was assumed that [56] [31]

1. A 1% reduction in drag is equivalent to a 5 tonne reduction in mass.
2. The drag of the complete configuration is of the order of 250 counts (count = accepted measure for drag), hence 1% of the drag is therefore approximately 2.5 counts.
3. There are two torsion boxes, so a 5 tonne reduction is a reduction of 2.5 tonnes in each box.

From assumption 1 it is clear that a 2.5 counts reduction in drag must have the same influence as a 2.5 tonne reduction in a single torsion box mass.

As equation 6.1 was used as datum during the MDO project, it was also used as starting point for the integration of manufacturing cost. However, integration of manufacturing cost into this DOC objective function is not so straight forward, as there are aspects of coupling, parameter classification, non-true partial derivatives and discrete cost changes.

### 6.1.1 Coupling aspects

When equation 6.1 was developed, it could only be used under the firm assumption of having a rigid wing, where a weak coupling between aerodynamics and structures is present. It was assumed that changes of the internal wing structure have a negligible effect on the aerodynamics.

For manufacturing cost, however, as described in chapter 5 one of the required inputs for the recurrent cost models is a mass breakdown. Combined with layout information and manufacturing rules, the mass breakdown forms an essential input to the cost calculation. Hence there is a strong coupling and dependency between cost and optimised mass which cannot be ignored. As the optimised mass is used, this indirectly means that the stresses in the structure have been considered and taken into account and that the cost has been calculated for a minimum mass problem with stress constraints.

### 6.1.2 Parameter classification: geometry and layout

Equation 6.1 involves mainly the six primary design parameters such as wing sweep, area, aspect ratio, etc. described in table 3.1 on page 38. However, two types of parameters which are used for design studies can be identified: layout and geometry parameters. The six major parameters are classified as geometric parameters. Stringer pitch and number of ribs are classified as layout parameters because they affect the internal layout of the wing box.

As described in chapter 5, three major cost contributions, material cost, assembly cost and detailed manufacturing cost were defined. While material



cost is driven by component mass, the assembly and detailed manufacturing cost changes are mainly affected by layout changes and less by geometry changes.

### 6.1.3 True partial derivatives

Because of the direct coupling between mass and cost and the fact that the cost calculation depends on weight, layout inputs and rules it is not possible to take a true partial derivative of cost which with respect to geometry and layout changes.

Consider, for example, an increase of wing depth as a parameter  $p_i$ . The cost of the metal wing is determined by the optimised mass ( $M$ ), component area ( $A$ ) and component length ( $L$ ). The change of cost with respect to the change of wing depth can be written as:

$$\frac{\partial c}{\partial p_i} \quad (6.2)$$

With:

$$c = f(M, A, L) \quad (6.3)$$

This could be written as:

$$\frac{\partial c}{\partial p_i} = \frac{\partial c}{\partial M} \frac{\partial M}{\partial p_i} + \frac{\partial c}{\partial A} \frac{\partial A}{\partial p_i} + \frac{\partial c}{\partial L} \frac{\partial L}{\partial p_i} \quad (6.4)$$

From equation 6.4 it is clear that because of the direct coupling one cannot isolate the change of cost with respect to the change of mass ( $\frac{\partial c}{\partial M}$ ) or the change of cost with respect to the change of Area ( $\frac{\partial c}{\partial A}$ ) etc. As shown in the results of section 5.2.4.1 on page 5.2.4.1 increasing the wing depth leads to an increase in rib cost because of the larger components to manufacture and a reduction in skin mass because the structure became more effective in bending.

### 6.1.4 Discrete cost changes

Optimisers are guided by gradient information to find an optimum solution. However, for manufacturing cost calculations one does not always have gradient information available, as the changes of manufacturing parameters can be discrete (e.g. tolerances, number of ribs, rules etc.). Depending on the

cost problem, equation 6.3 could include certain design and manufacturing rules which are discrete as input. An example of a design rule is the fact that the minimum stringer distance from rear and front spar has to be equal to half a stringer pitch. Stringers closer to the spars are not allowed. Hence, when increasing stringer pitch, one observes sudden changes in stringer mass and cost as a whole stringer is being removed because it passed the cut-off value of half a stringer pitch. Another example is the change to a finer tolerance, which causes a sudden step increase of the manufacturing cost as more intensive machining, better tools, more labour, more rigorous quality testing etc. is required.

## 6.2 MDO paradigm

The aspects of parameter classification, absence of true partials, direct coupling and the presence of discrete cost changes as discussed in the previous section, formed the motivation for not including manufacturing cost in the DOC equation 6.1 of page 156. It was decided not to treat manufacturing cost at the same level of mass and drag. Instead the following multi-level scheme which is shown in figure 6.1 was employed. Two main levels can be identified. The first one is the top-level which deals with the optimisation of aircraft direct operating cost, using the six primary geometry variants. The second level consists of the sub-levels which deal with specific contributors to the top-level. Thus, one of the component inputs is concerned with the ensuring that the optimiser takes into account the minimisation of structural mass and cost, the other sub-level minimises the aerodynamic drag. As can be seen it is assumed that the only cost parameter which can be varied at the top-level is mass, whereas cost is being dealt with as part of the structural sub-level. The following process is being performed:

- Top-level: minimisation of DOC by minimising mass and drag, subject to geometry changes.
- Sub-level 1: minimisation of cost for a given wing geometry, subject to layout changes. Input to the cost module is the minimum mass design for the changed layout.
- Sub-level 2: Calculation of drag.

Assume, for example, that sweep is the only geometry parameter which changes at the top-level. For a specific value of sweep, the structural sub-level (i.e. sub-level 1) performs a mass minimisation and subsequently examines



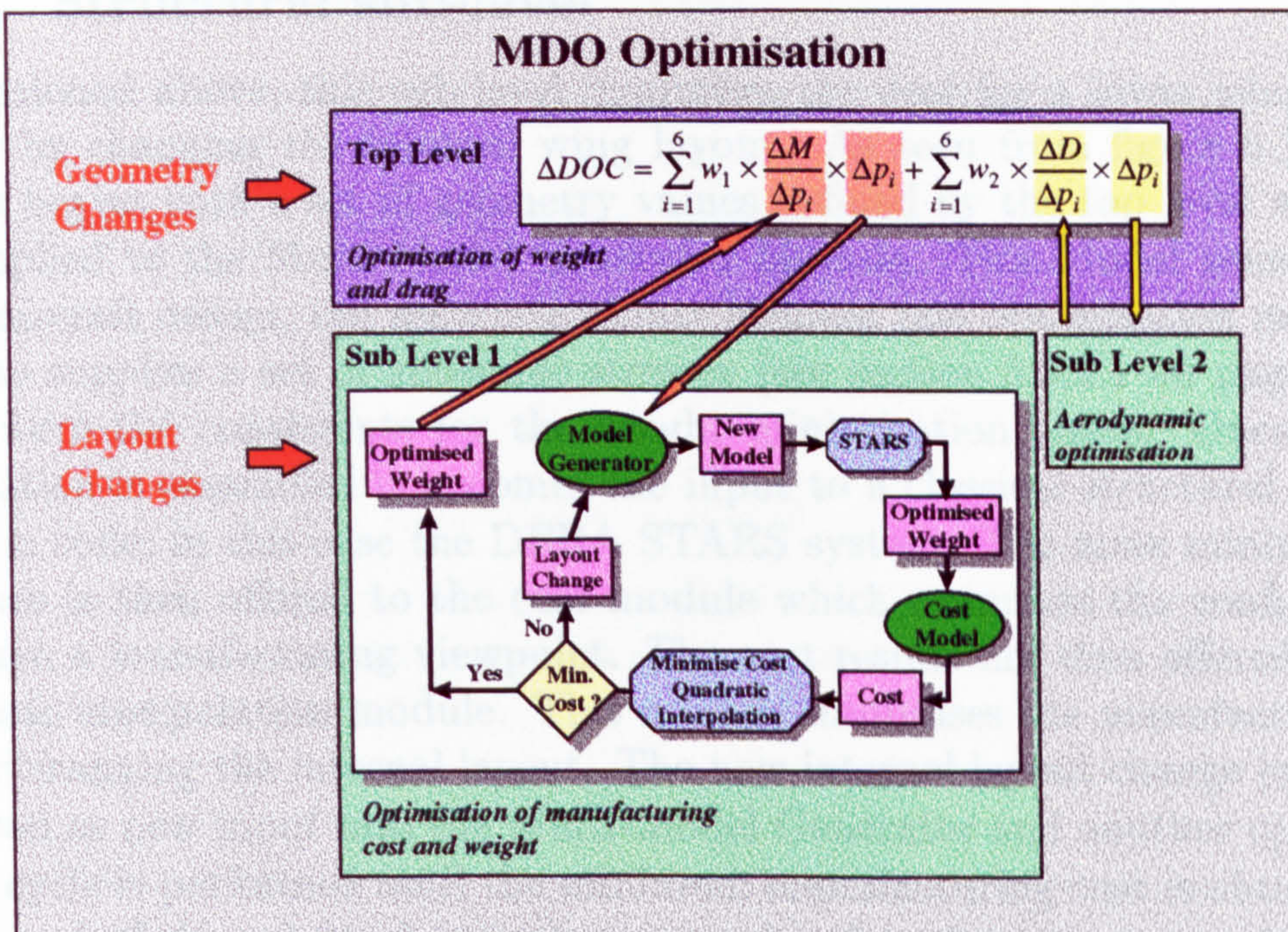


Figure 6.1: Multi-level optimisation scheme.

the configuration from a manufacturing viewpoint. It then makes changes to internal structural layout in order to minimise the manufacturing costs. At the end of this sub-level optimisation cycle, the weight and its derivative information which is associated with the minimum cost layout is handed over to the top-level. Based on the information obtained from the sub-levels, the upper level is now able to make a step in design parameter space to generate a new value for the sweep parameter. This new value can then be passed down to the sub-levels for re-optimisation and the creation of a new set of finite differences which can be passed back up to the top-level.

### 6.2.1 Solving DOC

To perform the top level minimisation of the Direct Operating Cost, a sequential quadratic programming algorithm, E04UCF (Mark 17) from the Numerical Algorithm Group (NAG) Fortran Library [86] is used. This algorithm uses a quadratic approximation for the objective function and uses linearised constraints. If first derivatives are not available, they are approximated by finite differencing (forward or central).



## 6.2.2 Structural sub-level

As mentioned above, this sub-level minimises the cost for a given wing geometry by changing the internal wing layout. As seen from figure 6.1 the process begins with a set of geometry values defined by the top level which are supplied to the Multi-Model Generator Module. This Model generates a new aircraft design, the associated finite element and optimisation model, and also supplies a set of allowable stresses (see section 3.5.2.1 on page 55) which form the constraints for the weight minimisation phase. Once this information is assembled it becomes the input to a classical structural optimisation code; in this case the DERA STARS system. The mass minimised structure is then offered to the cost module which examines the configuration from a manufacturing viewpoint. The cost results are then offered to a quadratic interpolation module. This module minimises the manufacturing cost by changing the internal layout. The new internal layout change is then imported as new input into the Multi-Model Generator and another optimisation cycle is performed until the minimum manufacturing cost is obtained. At the end of the sub-level optimisation cycle, the minimum mass which is associated with the minimum cost layout is handed over to the top level.

To reduce the computation time, the STARS fully stressing algorithm is used in place of the Newton method. For the quadratic interpolation module, algorithm E04ABF from the NAG Fortran Library [87] was implemented. The algorithm searches for a minimum in a given finite interval of a continuous function of a single variable. It requires function values only as input. The method is intended for functions which have a continuous first derivative, although it will usually work if the derivative has occasional discontinuities.

## 6.2.3 Aerodynamic sub-level

Given a set of geometry parameters, an aerodynamic model is generated by the Multi-Model Generator and CFD analysis can be performed. An overview of the different tools and methods which were used as part of the CFD analysis is given in section 3.4.2.1 on page 51. As mentioned in section 3.7 on page 67 it was not the scope of this thesis to perform CFD analysis. Instead, response surfaces for the CFD drag results were obtained from other partners and used as input for the MDO paradigm.



## 6.2.4 Process control

A process controller was developed which had to perform the following tasks:

- Link all the optimisers and MDO software
- Exchange and store input and output data
- Monitor the design progress
- Execute the proper tools at the proper time

This process controller was developed using the UNIX SHELL language and Fortran. A listing of all the programme routines can be found in appendix A on page 183.

## 6.3 Results

The MDO paradigm was developed and tested in several stages. In a first stage the focus was on the top level of figure 6.1 of page 160, a second stage looked at the structural sub-level and a third stage looked at the integration and results when all levels were combined and integrated.

### 6.3.1 Results top level

The top level was first of all developed and tested using response surface data for both the structure and aerodynamic sub-levels, i.e. no sub-level optimisation was performed. For reasons of comparison, it was decided to study only sweep and area changes, as there was project data available for these cases. The validation of this top level was also based on the wing models which did not take into account the improved rib sizing. Lower and upper bounds for the sweep changes were 31° and 35 °, the lower and upper bounds for the area changes were 625 m<sup>2</sup> and 825 m<sup>2</sup>. Figure 6.2 shows the wing layouts which correspond with these bounds of the design space. Because of an aircraft accessibility constraint, the wing span needs to be kept constant (i.e. 77.1 m). Thus, taking into account a constant wing span the following relationship between aspect ratio and wing area needs to be respected:

$$AspectRatio = \frac{77.1^2}{Area} \quad (6.5)$$

The DOC was minimised using equation 6.1 with a weighting factor of 1 for mass and 2 for drag which were obtained from British Aerospace. In



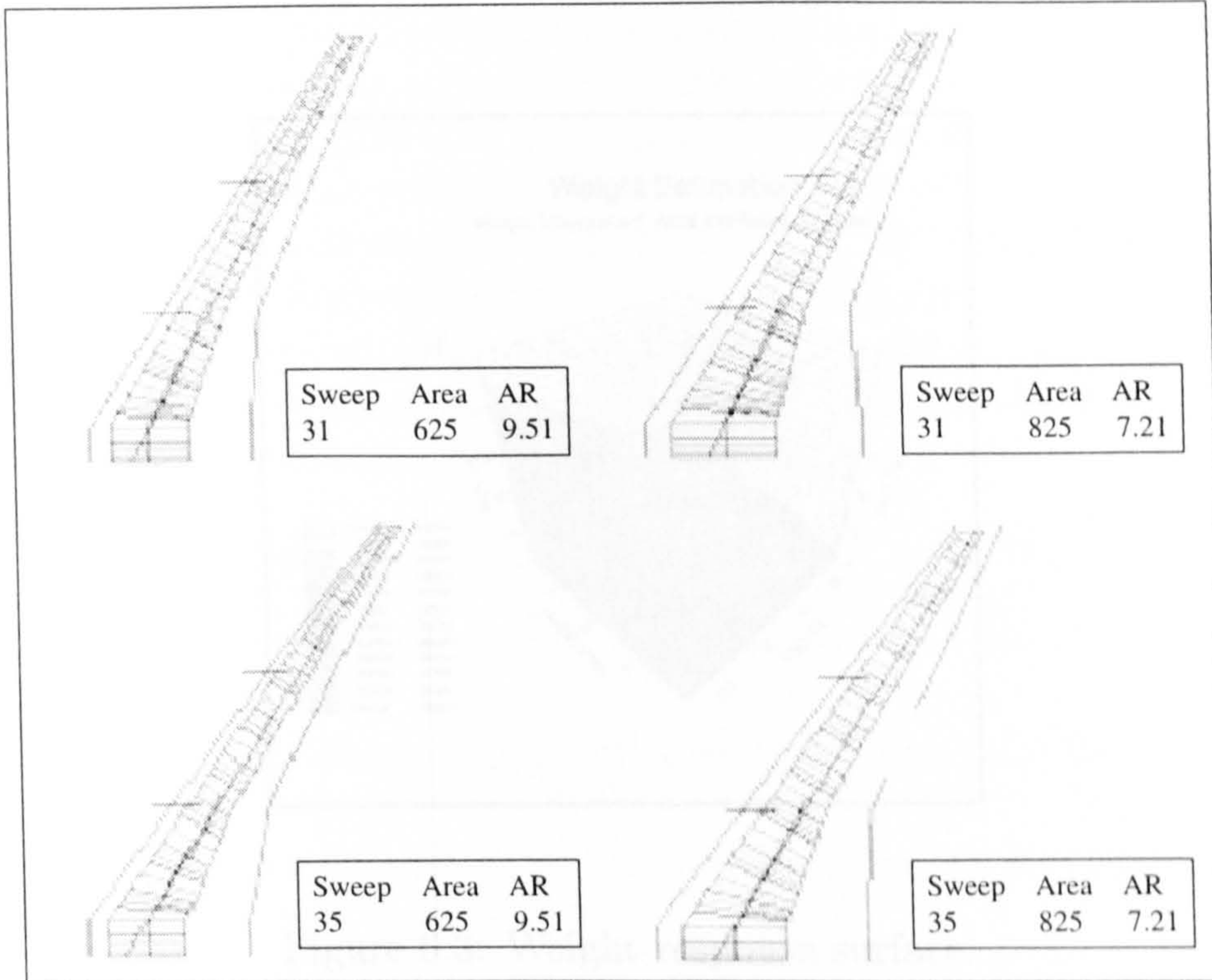


Figure 6.2: The 4-corners of the design space.

summary the following DOC formulation was solved for changes of wing sweep and area:

$$\Delta DOC = 1.0\Delta W + 2.0\Delta D \quad (6.6)$$

With the weight in tonnes and the drag in drag counts. The response surfaces which provide the mass and drag information are shown in figures 6.3 and 6.4.

Table 6.1 shows the optimised DOC results for this test case and the values which were available from project data [56].

|           | Sweep ( $^{\circ}$ ) | Area ( $m^2$ ) | $\Delta DOC(\%)$ |
|-----------|----------------------|----------------|------------------|
| Top-level | 34.98                | 690            | -6.2             |
| Project   | 35                   | 699            | -7               |

Table 6.1: Results DOC minimisation, comparison between top-level and MDO project results.

As can be seen from the table the same values and trends are predicted.



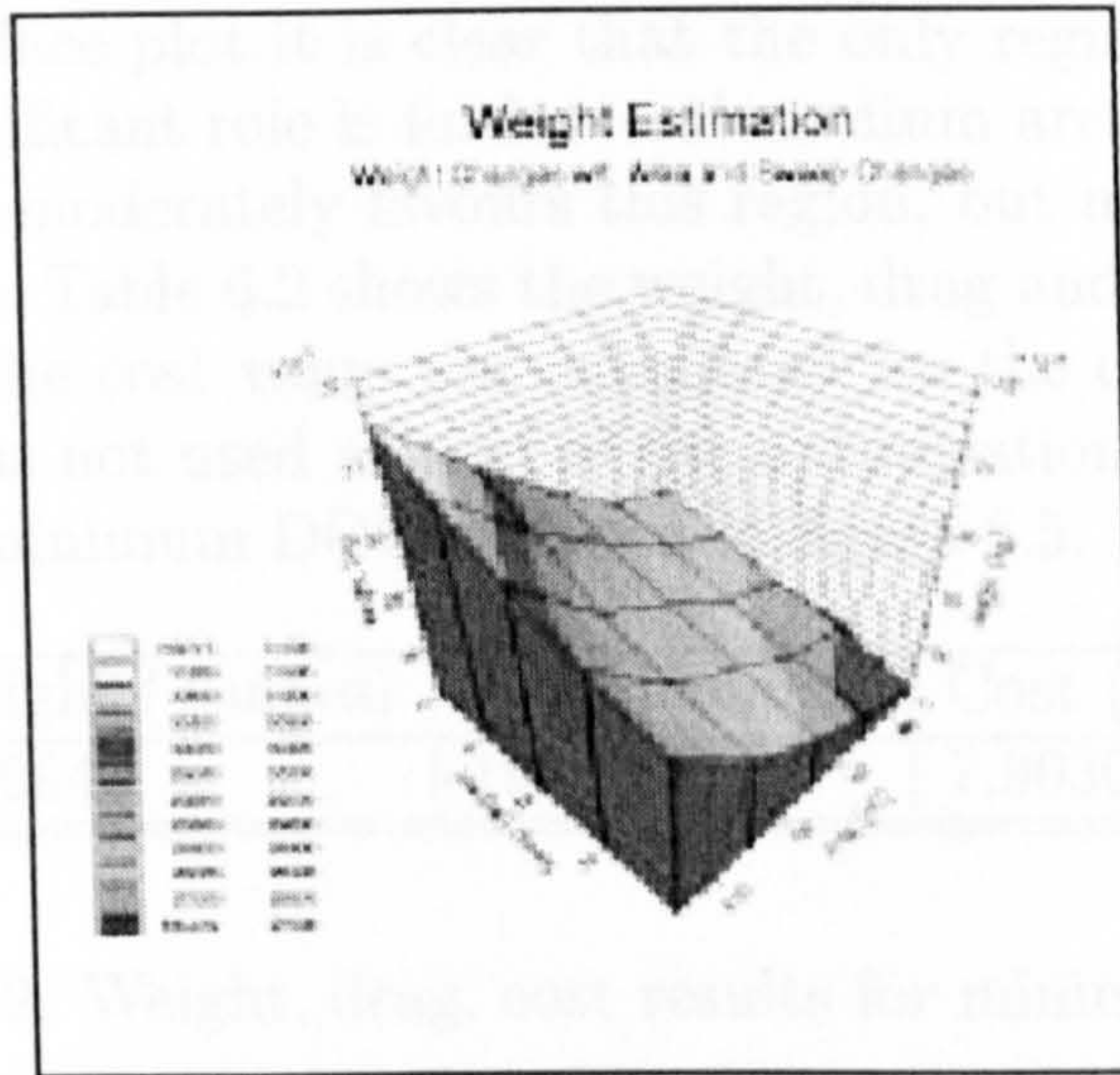


Figure 6.3: Weight response surface.

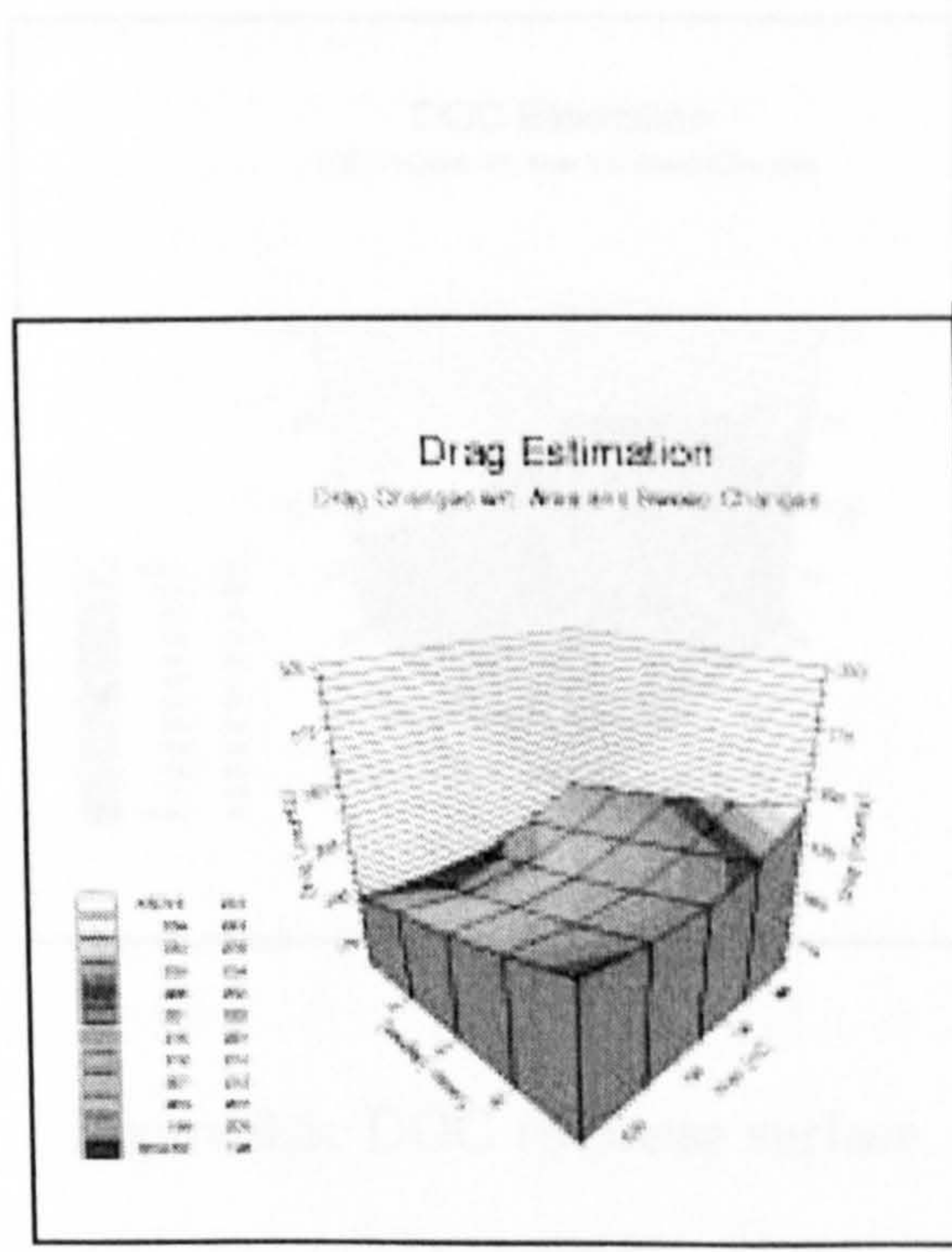


Figure 6.4: Drag response surface.



Slight differences occurred because of the very small drag changes at maximum sweep and near the area range of 700 m<sup>2</sup> (see figure 6.4). From the mass and drag surface plot it is clear that the only region where structural weight plays a significant role is for low and medium area high sweep wings. The aerodynamics moderately favours this region, but mass effects penalise the low area wings. Table 6.2 shows the weight, drag and cost values for this minimum DOC. The cost value was calculated for the optimum sweep and area values but was not used as part of the optimisation process. A surface plot showing the minimum DOC is shown in figure 6.5.

| Weight (Tonnes) | Drag (counts) | Cost (M\$) |
|-----------------|---------------|------------|
| 28.994          | 193.1225      | 7.903085   |

Table 6.2: Weight, drag, cost results for minimum DOC

Having developed and tested a top level optimisation process, the structure sub-level was developed.

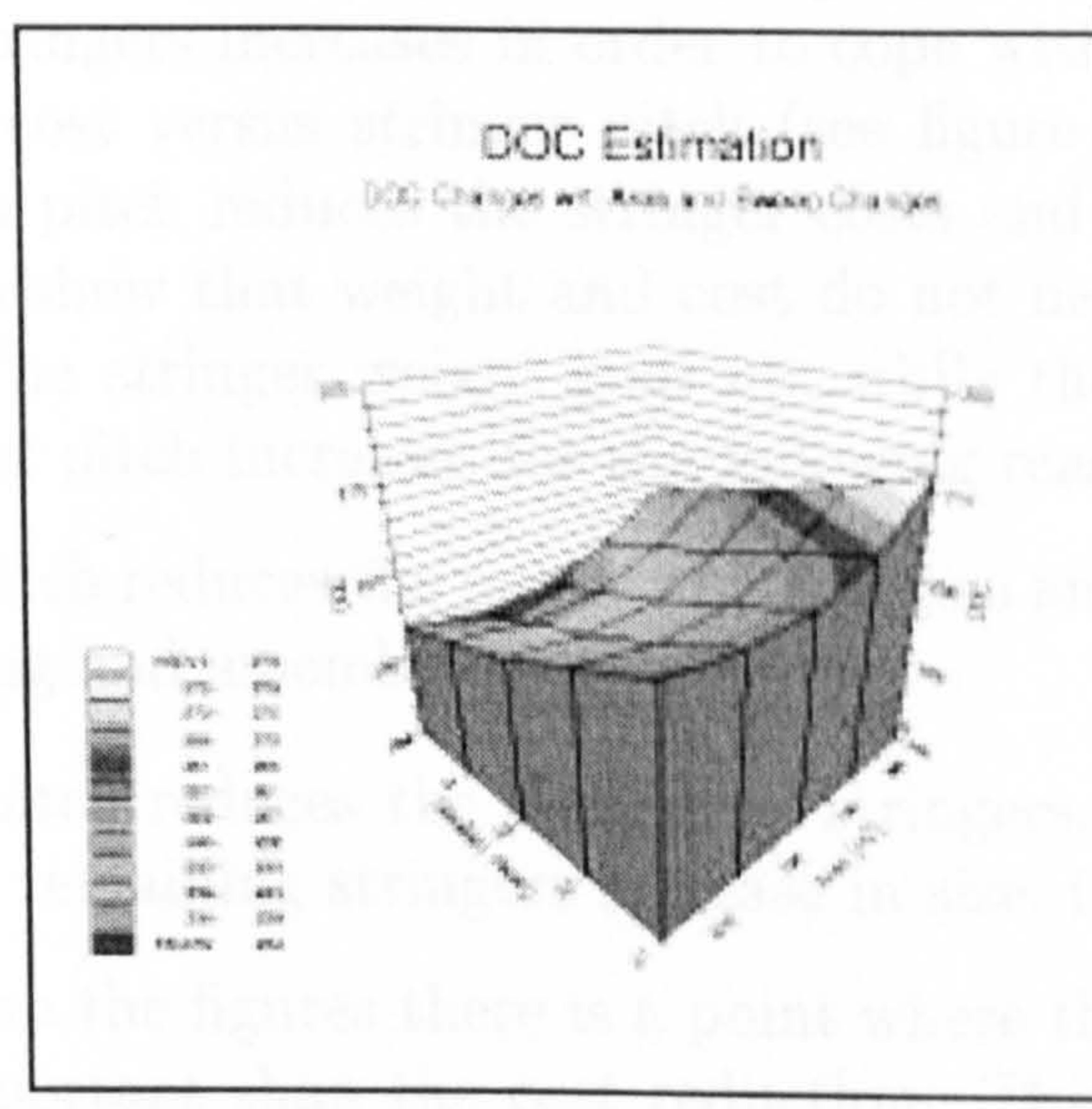


Figure 6.5: DOC response surface.

### 6.3.2 Results structure sub-level

As described in section 6.2.2 this sub-level minimises the cost for a given wing geometry by changing the internal wing layout. For a given wing area and



sweep, the stringer pitch was optimised for minimum cost. The improved model with rib sizing was used. Table 6.3 shows the results at optimum stringer pitch for a wing model which contains 69 and 41 ribs. The sub level optimisation process took 9 iterations for the 69 rib case and 10 iterations for 41 rib case. For each of these iterations, the optimised weight was obtained using seven iterations of the STARS fully stressing algorithm.

| Number of Ribs | Stringer Pitch (m) | Cost (\$) | Weight (kg) |
|----------------|--------------------|-----------|-------------|
| 69             | 0.250              | 6955487   | 25240       |
| 41             | 0.112              | 7676161   | 21106       |

Table 6.3: Stringer pitch and wing weight for minimum cost .

A lower number of ribs makes the skins more sensitive to buckling, to prevent this more stringers are required which explains the lower value for optimum stringer pitch with 41 ribs (see table 6.3).

Based on the 69 rib wing model, figure 6.6 shows how the weight varies when the stringer pitch is changed. As the stringer pitch increases, the weight of the skins and stringers increases in order to cope with buckling. Looking at the changes of cost versus stringer pitch (see figure 6.7) it can be seen that an increase in pitch reduces the stringer costs and increases skin cost.

These two figures show that weight and cost do not necessary move in the same direction. The stringer weight goes up, while the stringer cost goes down when stringer pitch increases for the following reasons:

- Increase in pitch reduces the number of stringers and hence the detailed manufacturing and assembly cost.
- Increase in pitch reduces the number of stringers, hence to cope with buckling the remaining stringers increase in size, i.e. weight.

As can be seen from the figures there is a point where the increase in weight becomes more important than the cost reduction. This trends can also be observed in figure 6.8 which shows the change of total manufacturing cost with respect to stringer pitch changes. The cost is also broken down in the three cost categories: assembly, detailed manufacturing and material cost. Pitch increase reduces detailed manufacturing and assembly costs, while the material costs increase. Mass and manufacturing cost trading, however, are influenced by the assumptions made relating to material cost and the structural behaviour.

Figures 6.6 and 6.7 also show an increase in weight and cost for the spars. Normally the spars should not be so sensitive to changes in stringer pitch.



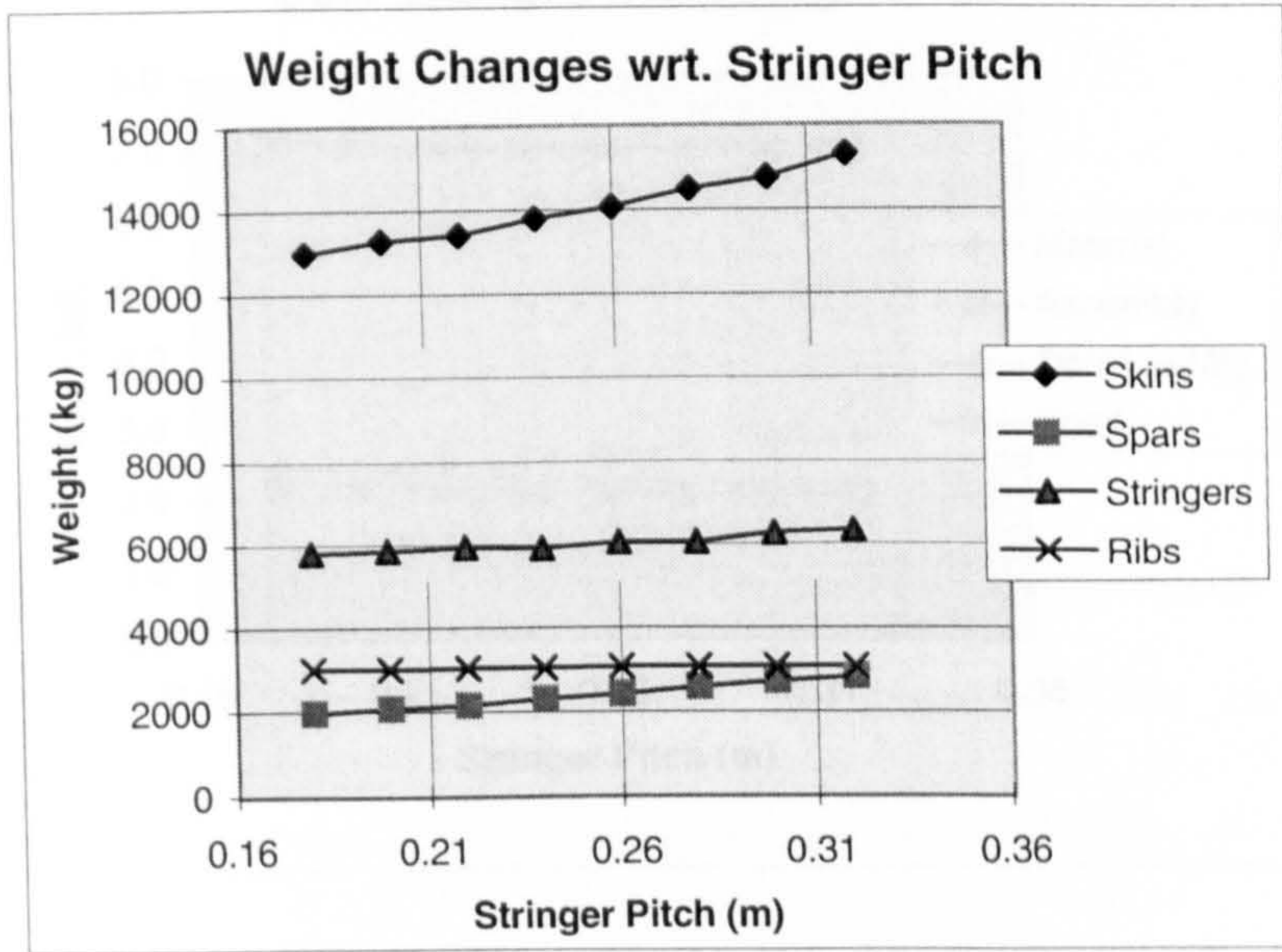


Figure 6.6: Weight changes wrt. stringer pitch changes.

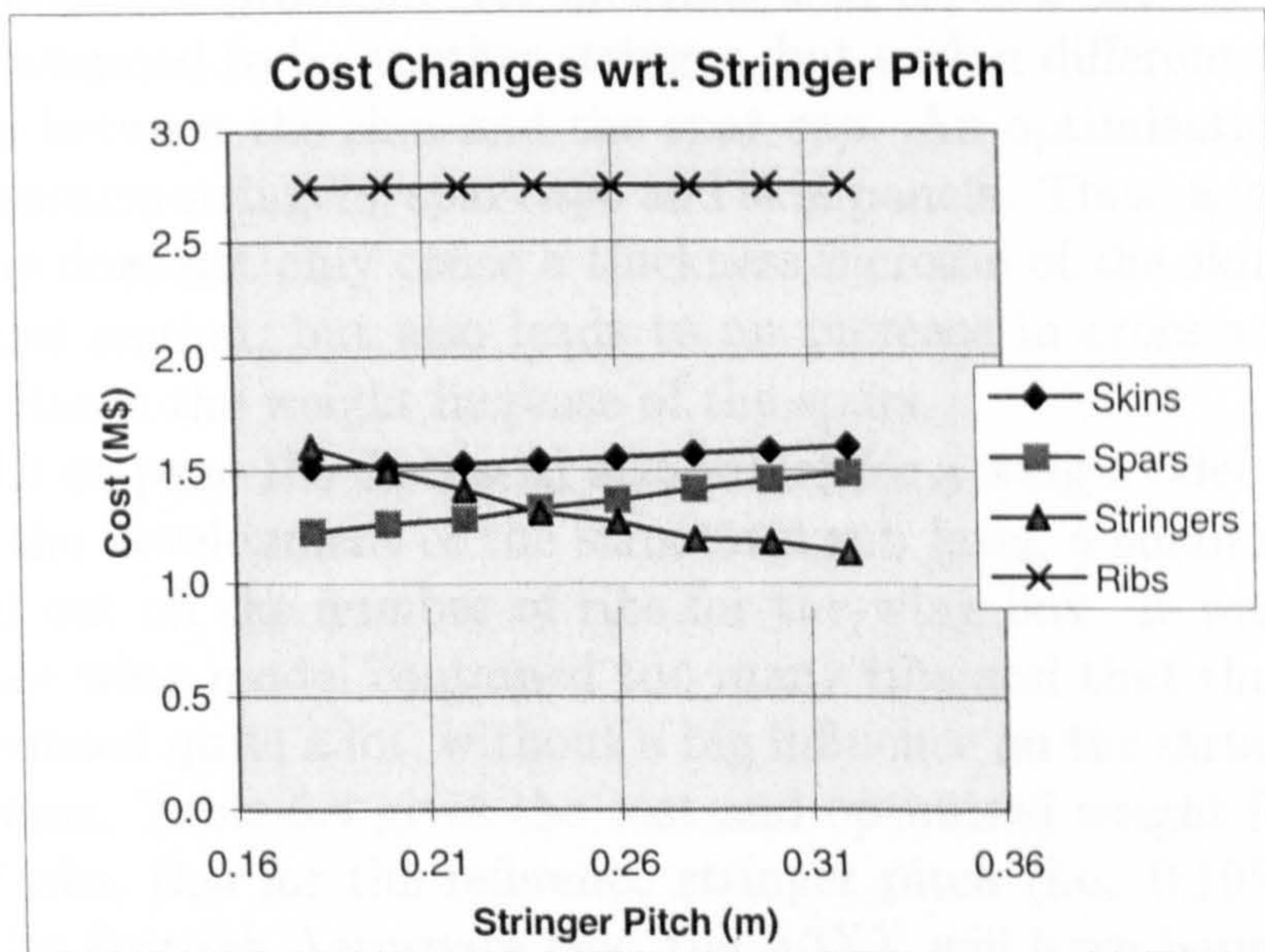


Figure 6.7: Cost changes wrt. stringer pitch changes, breakdown per component.



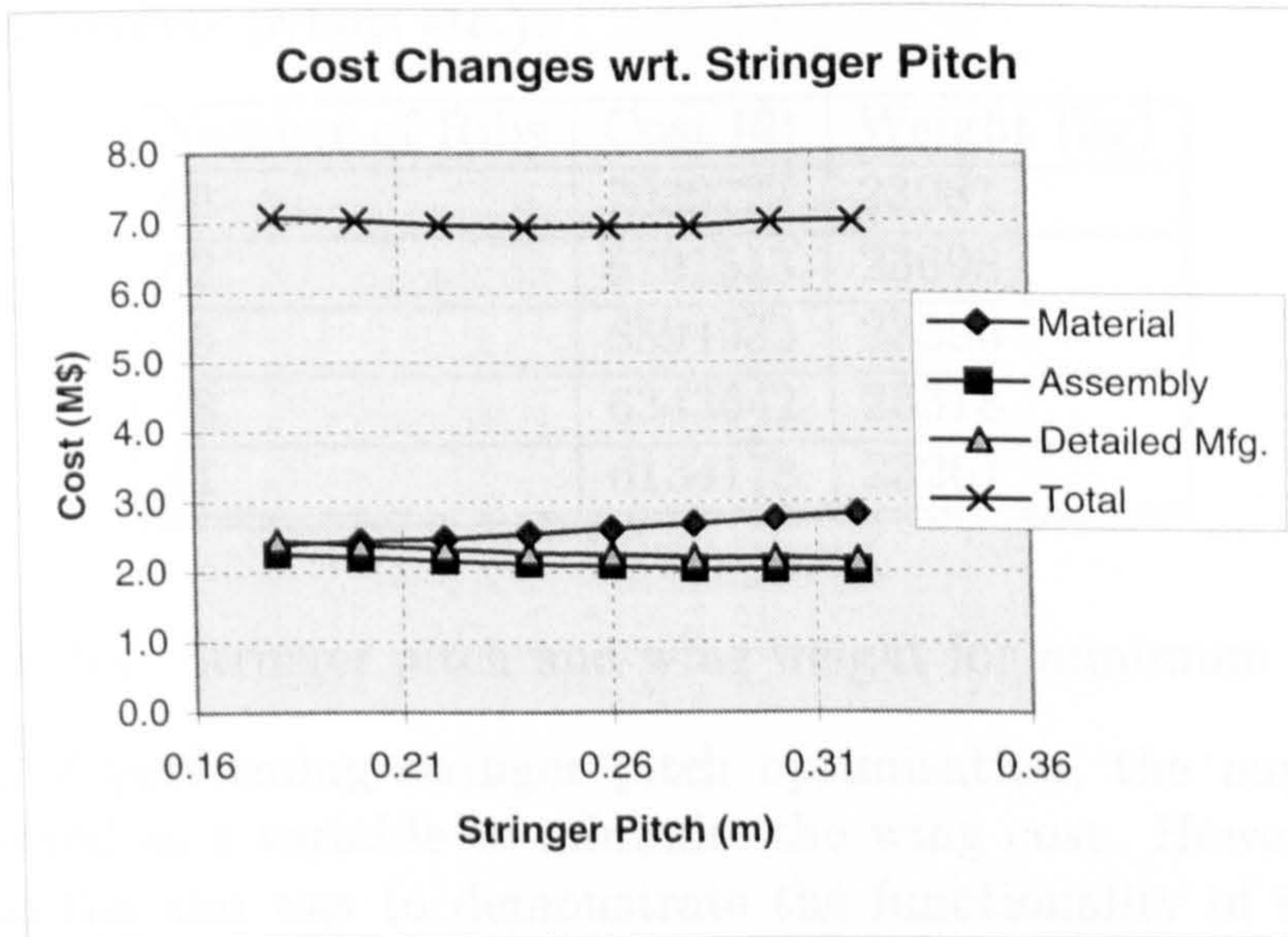


Figure 6.8: Cost changes wrt. stringer pitch changes, breakdown per cost category.

However, this behaviour can be explained as it is related to the definition of the optimisation problem. As mentioned in section 4.1 on page 71 the spar cap is assumed to be another stringer, but with a different proportional relationship between the skin and the spar cap. An optimisation zone near the spars contains stringers, spar caps and skin panels. Thus, a value increase for this zone does not only cause a thickness increase of the skin panels and stringer cross section, but also leads to an increase in cross section of the spar caps. Hence the weight increase of the spars.

Table 6.2 on page 165 discussed also values for a wing model with 41 ribs. As part of the development of the structural sub level, a small investigation was carried out on the number of ribs for the wing box. It was found that the reference wing model contained too many ribs and that the rib number could be reduced quite a lot, without a big influence on the structural weight and behaviour. Table 6.4 gives the cost and optimised weight for a reduced number of ribs, this for the reference stringer pitch (i.e. 0.195 m). It was confirmed by British Aerospace that the A3XX will have between 40 to 45 ribs. A further rib reduction of less than 40 ribs was not investigated, as it was felt that a bigger reduction would not be realistic from structural point of view (because of the assumptions in the structural optimisation model and the reduced number of load cases considered) and wing layout point of view (i.e. need enough ribs to provide attachment points to secondary wing



structure and engine pylons etc.).

| Number of Ribs | Cost (\$) | Weight (kg) |
|----------------|-----------|-------------|
| 69             | 7039727   | 23981       |
| 62             | 6791513   | 23698       |
| 55             | 6594935   | 23596       |
| 48             | 6343642   | 23318       |
| 41             | 6134178   | 23261       |

Table 6.4: Stringer pitch and wing weight for minimum cost .

Instead of performing stringer pitch optimisation, the number of ribs can be selected as a variable to minimise the wing cost. However, this was not done as the aim was to demonstrate the functionality of the approach developed in this sub level.

### 6.3.3 Results MDO paradigm

Having tested and evaluated the results of both the top level and the structural sub-level, all the different software components were integrated to form the MDO paradigm. Starting from the reference aircraft, with a sweep of  $33^\circ$ , an area of  $725 \text{ m}^2$  and a stringer pitch of  $0.195 \text{ m}$  the MDO paradigm delivered the optimised design which is shown in table 6.5. As can be seen from the table, the DOC reduction is about 2% more than that of combined mass and drag optimisation without taking into account manufacturing cost (see table 6.1). Comparing the results with table 6.1 it can also be observed that the sweep is similar, but that the wing area has reduced more. This, for the reason that because of internal layout changes (i.e. stringer pitch) the penalising mass effects have reduced. (Recall from section 6.3.1 that it was the mass which prevented low area high sweep wings.) Figure 6.9 shows the actual wing layout for this optimal design in comparison with the reference wing layout.

Solving this paradigm involved 21 iterations of the top-level. For each of these iterations, on average 10 structural sub-level iterations are performed, which in their turn include each 7 iterations of fully stressing. Due to this high level of iterations, the paradigm has proved to be very computationally demanding. By applying more efficient programming approaches the computational efficiency might be improved a lot. However, as it was the aim of this thesis to develop and demonstrate a novel approach to integrate manufacturing cost in an MDO environment, computational efficiency was not a prime objective.



|                      | Paradigm |
|----------------------|----------|
| Sweep ( $^{\circ}$ ) | 34.96    |
| Area ( $m^2$ )       | 625.1    |
| Pitch (m)            | 0.242    |
| Weight (kg)          | 24071    |
| Drag (counts)        | 197.2478 |
| Cost (\$)            | 6307558  |
| $\Delta DOC(\%)$     | -8.1     |

Table 6.5: Results MDO paradigm.

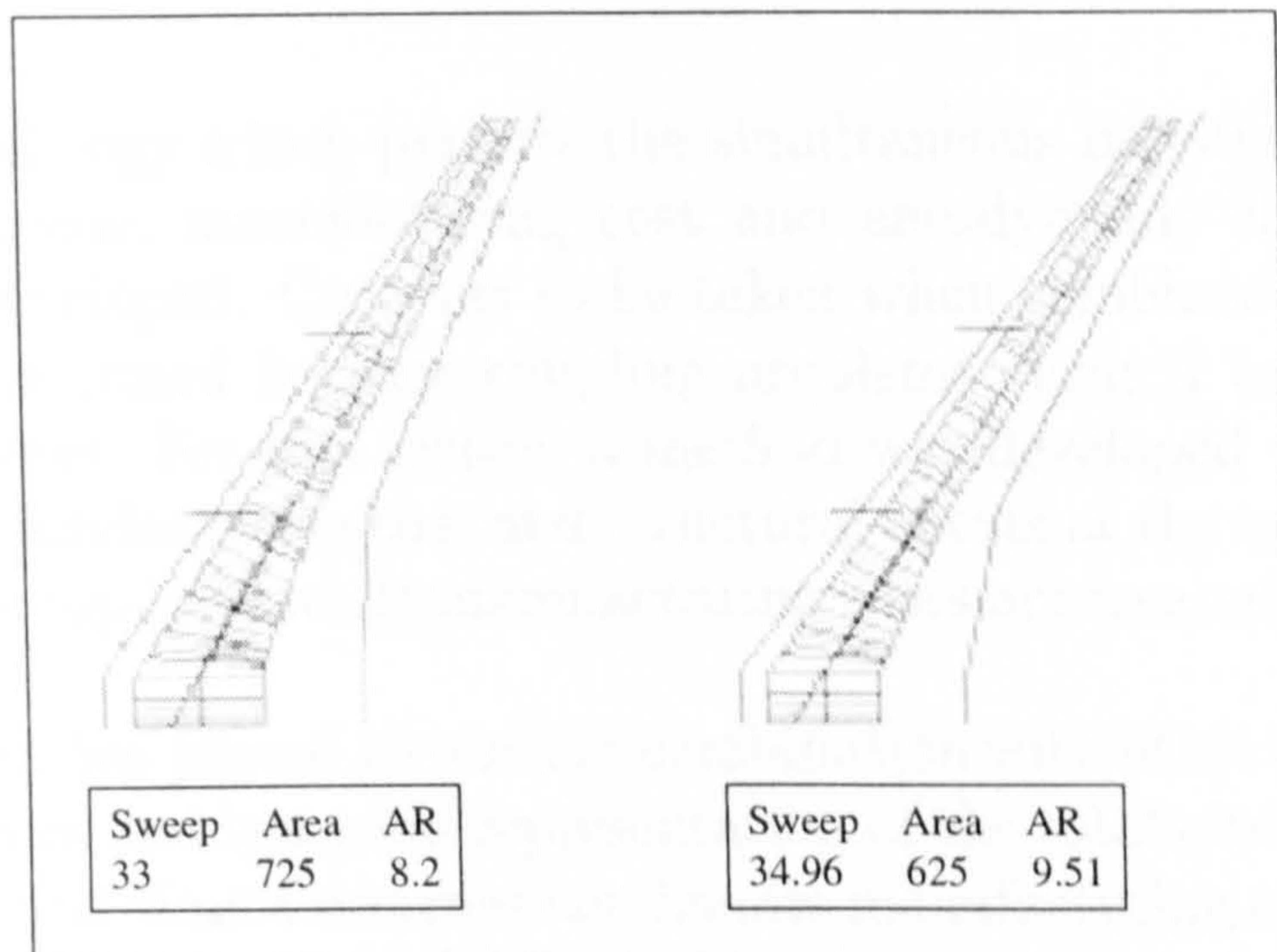


Figure 6.9: Reference and optimal wing layout.

## 6.4 Conclusions

A method to integrate recurrent manufacturing cost into an MDO process has been developed. It allowed for trade-off studies between cost and mass on a sub-level. This, for internal layout changes such as stringer pitch or number of ribs. Optimising stringer pitch for minimum cost and weight has lead to a pitch increase from 0.195 to 0.250. A similar investigating on the number of ribs has shown that the current number of 69 ribs is far too high and that a wing box with about 40 ribs is more realistic.



# Chapter 7

## Conclusions

A new methodology which permits the simultaneous investigation of structural optimisation, manufacturing cost and aerodynamic effects has been successfully developed. Care has to be taken when combined mass and cost studies are performed because coupling problems occur if both are treated at the same level. For this reason, a method was developed which does not incorporate manufacturing cost and structural effects at the same level: mass is treated at a top-level while manufacturing costs are treated at a structural sub-level.

The system has proved to work in combination with other disciplines such as aerodynamics and is a good representation of the total aircraft design process. For the first time a designer can handle manufacturing cost information simultaneously with mass and drag, and is able to quickly assess the effect of cost changes with respect to design changes. By combining both top and bottom levels, a highly flexible system was developed which allows companies to import their own results or cost data and perform cost studies based on historical data or highly novel processes.

Although an automated Multi-Disciplinary Design and Optimisation system was created, it is clear that engineering experience is still required for the relevant technology areas.

Within this overall framework, the following objectives and conclusions were achieved:

- Structural optimisation was performed using MSC/NASTRAN and STARS on a the wing of a large civil airliner. A model with 156 design variables was used and an optimum weight of 24000kg was achieved. However, when taking into account other wing box structure, the weight is expected to increase by approximately 15 %.
- Optimisation of a combined metal/composite wing, using a simplified



composite model, lead to an additional weight reduction of about 1.6%.

- An early comparison with data from other optimisation packages such as OPTSYS, ELFINI has shown that the results closely agreed.
- Although it was presumed that structural optimisation would be a fairly straight forward process, it required a lot of attention in order to achieve good convergence. It was found that the convergence was strongly influenced by the size of the design problem, definition of the optimisation model, selection of the solution algorithm, proper selection of optimisation control options, user experience etc. This lead to the conclusion that structural optimisation is not as mature as originally expected. It still requires an iterative process and user experience which makes it far from 'plug and play'. However, once the optimisation process is under control it can be run multiple times for small changes in aircraft variants.
- Application of a Fully stressing algorithm has proven to give good and fast indications of where the optimum is situated. Although it does not give the proper optimum and weight distribution, the global optimum weight was always found within less than 3% of the actual optimum.
- The Dual bound was found to be a useful feature to track how close the solution was to the optimum, especially at the start of the thesis when there were many uncertainties on the value of the obtained optima.
- A prototype metal recurrent cost model and a composite recurrent cost model which employ a feature based approach, have been developed. Although they are of high level nature, they have proved to be a powerful tools for a designer to quickly investigate and visualise cost changes with respect to design changes. Validation has shown that the predicted cost changes are meaningful. The models allow the investigation of cost on both internal layout changes and external geometry changes. Cost studies of hybrid wings made of both metal and composite have also been implemented.
- Integration of cost and structural optimisation into an MDO process has allowed cost and mass trade-off studies. Studies have been carried out on stringer pitch and number of ribs. Optimising stringer pitch for minimum cost and weight has lead to a pitch increase from 0.195 to 0.250. A similar investigating on the number of ribs has shown that the current number of 69 ribs is far too high and that a wing box with about 40 ribs is more realistic.



- An MDO paradigm was generated to perform combined weight, drag and cost optimisation. In this paradigm, weight and drag were dealt with at a top-level, while manufacturing cost was treated at the structural sub-level. Although computationally expensive, it was demonstrated that the paradigm worked and can be used to give a cost, weight and drag efficient solution.
- Interaction between the thesis research and the European MDO project has not only led to exchange of results and data but gave an invaluable experience in practical application of MDO, communication and presentation skills, workshop practise, etc. A better overview of how MDO and Concurrent Engineering can work in the real world of industry could not be given by any text book or publication!

The research has identified many areas for further improvement and recommendations:

- Investigation into the fidelity of the finite element model.
- Structural optimisation which includes the ribs as design variables.
- Improve cost models, especially the prototype metal cost model so they are of a lower, more detailed level.
- Include other/novel production concepts in the cost models.
- Include non-recurrent cost into the cost models and investigate the interaction of cost at the preliminary and conceptual design change.
- Use tools such as ICAD to define the aircraft geometry. From this model the Finite Element model and the cost driver data can be generated. Currently the project software only defines a Finite Element model. All cost data is derived by interpreting the Finite Element data, which involves many Fortran routines and iterations. By using ICAD, a much more flexible approach towards model generation will be applied.
- Integrate the cost model into ICAD. This will make the models more efficient, generic and user friendly. It will allow the designer to faster investigate different production concepts.
- In order to make optimisation more accessible and a wider applied tool, a lot of improvement could still be done in developing more user friendly interfaces and clearer user manuals. Currently for large optimisation problems the definition of design variables, gauges, and constraints is



still an elaborate task, which is not supported by most pre-processors. The user is often required to write his own Fortran code to this task, however this does not give a lot of model flexibility. User manuals and interface with Artificial Intelligence tools could improve the quality of initial model definition and could assist a less experienced user with the procedures to obtain a good converging solution.

- Improve visualisation of optimisation results. Visualisation of design variable changes, design variable history, stresses etc. by post-processors is still in an early stage. The user is often left to write his own post-processing code. However, a good post processing capability would improve the accessibility to optimisation in general and would assist the designer in quickly identifying what is going wrong in case of convergence problems. This is very important for large scale realistic optimisation problems which produce a huge amount of results.



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# Appendix A

## Software description

This appendix contains the fortran and shell script files which were created for the MDO paradigm. The following files are described below:

- MDOParadigm.f
- SubLevel1.f
- SubLevel2.f
- mmgsript
- mmgsript2
- runstars

All the scripts and programs can be found and run in the MDO/MDO\_Kristof directory on the College of Aeronautics SUN system.



## A.1 MDOParadigm.f

This is the main coordinating program. It represents the top level which is shown in figure 6.1 on page 160. To perform the top level minimisation of the Direct Operating Cost, a sequential quadratic programming algorithm, E04UCF (Mark 17) from the Numerical Algorithm Group (NAG) Fortran Library [86] is used.

```
*****
* PROGRAM DOC
*
* Produced by Kristof Gantois, Cranfield College of Aeronautics
*
* 11 July 1998
*
* Purpose: MDO paradigm: Toplevel DOC minimisation for mass and drag
*           Sublevel Cost minimisation
*
* WEIGHTINGS Top Level: Mass*1+Drag*2
*
* Input/Output with:
* - Mass calculation using DERA STARS
*
* - BAe drag count calculation through response surface:
*   DragResSurfBAe.f (economic cruise)
*   Drag calculated for changes of the following parameters:
*   Area and Sweep (with adaption of Aspect Ratio)
*
* - Cranfield Cost Calculation
*
*   - E04UCF SQP Program from NAG library
*     Mark 16 Release. NAG Copyright 1993.
*
*   - Output to files: MDOPmasscheck.dat
*                       MDOP\_mass\_drag\_cost.dat
*                       MDOP\_DOC\_Var.dat
*
* Variables: Sweep, Area, ARatio (Aspect Ratio), Ptich, Mass, Cost, Drag
```



```

*
* Variables to control optimisation process:
*           X1LB : Lower bound variable x1 (i.e. sweep)
*           X1UB : Upper bound variable x1 (i.e. sweep)
*           X2LB : Lower bound variable x2 (i.e. Area)
*           X2UB : Upper bound variable x1 (i.e. Area)
*           N : Number of variables
*           NCLIN : number of linear constraints
*           NCNLN : number of nonlinear constraints
*           OBJF : Objective function
*

```

\*\*\*\*\*

```

* .. Parameters ..

```

```

INTEGER          NIN, NOUT
PARAMETER        (NIN=5,NOUT=6)
INTEGER          NMAX, NCLMAX, NCNMAX
PARAMETER        (NMAX=10,NCLMAX=10,NCNMAX=10)
INTEGER          LDA, LDCJ, LDR
PARAMETER        (LDA=NCLMAX,LDCJ=NCNMAX,LDR=NMAX)
INTEGER          LIWORK, LWORK
PARAMETER        (LIWORK=100,LWORK=1000)

```

```

* .. Local Scalars ..

```

```

DOUBLE PRECISION OBJF
INTEGER          I, IFAIL, ITER, J, N, NCLIN, NCNLN

```

```

* .. Local Arrays ..

```

```

DOUBLE PRECISION A(LDA,NMAX), BL(NMAX+NCLMAX+NCNMAX),
+               BU(NMAX+NCLMAX+NCNMAX), C(NCNMAX),
+               CJAC(LDCJ,NMAX), CLAMDA(NMAX+NCLMAX+NCNMAX),
+               OBJGRD(NMAX), R(LDR,NMAX), USER(1), WORK(LWORK)

```

```

INTEGER          ISTATE(NMAX+NCLMAX+NCNMAX), IUSER(1),
+               IWORK(LIWORK)

```

```

DOUBLE PRECISION Sweep,Area,ARatio,X(NMAX)

```

```

REAL             Swout,Arout,ARaout

```

```

REAL             Pitch,Cost

```

```

* ..Parameters and arrays for time monitoring..

```

```

REAL             TIMEARRAY(2)

```

```

REAL             ELAPSE

```

```

REAL             DELAPSE

```



```

INTEGER          HMS(3)
INTEGER          DMY(3)

*   .. External Subroutines ..
EXTERNAL          CONFUN, E04UCF, OBJFUN
*   DOUBLE PRECISION Cost

CHARACTER*14     CMD,text

*   .. Executable Statements ..
WRITE (NOUT,*) 'E04UCF Example Program Results'

open(7,file='MDOPmasscheck.dat',form='formatted')
open(23,file='MDOP\_mass\_drag\_cost.dat',form='formatted')
open(9,file='MDOP\_DOC\_Var.dat',form='formatted')

print*, 'START DOC: Weight, Drag and Cost optimisation'
print*, 'Data from response surfaces'
print*, 'Weigting factors: 1*M,2*D'

*
*Specification of values for LP
*

* Number of variables

      N=2

* Number of Linear/nonlinear Constraints

      NCLIN=0
      NCNLN=0

*Upper/lower bounds for design variables X1=Sweep,X2=Area

      X1LB=31.05
      X1UB=34.96

```



```
X2LB=625.1
X2UB=824.9
```

```
* Start values
```

```
Sweep=33
Area=725
```

```
* Aspect Ratio = (Span)**2/Area
ARatio=((77.1038)**2/Area)
```

```
* Read/assign start values so they can be used by NAG routine
IF (N.LE.NMAX .AND. NCLIN.LE.NCLMAX .AND. NCNLN.LE.NCNMAX) THEN
```

```
BL(1)=X1LB
BL(2)=X2LB
```

```
BU(1)=X1UB
BU(2)=X2UB
```

```
X(1)=Sweep
X(2)=Area
```

```
print*, 'Update input file for start optimisation '
```

```
Swout=Sweep
Arout=Area
ARaout=ARatio
```

```
* initiate output file
```

```
write(23,*) , ' OBJ ' , ' Mass ' ,
+ ' Drag ' , ' Cost'
```

```
write(9,*) , ' OBJ ' , ' Sweep ' ,
+ ' Area ' , ' Pitch'
```



```

* Optional parameters for NAG routine
*
*   Use finite differencing:
*   CALL E04UEF ('Derivative level = 0')
*
*   Maximum number of iterations:
*   CALL E04UEF (ITERation Limit = 2')
*
*   Solve the problem
*
*   IFAIL = -1
*
*   print*, 'Start NAG SQP algorithm to minimise DOC'
*
*   CALL E04UCF(N,NCLIN,NCNLN,LDA,LDCJ,LDR,A,BL,BU,CONFUN,OBJFUN,
+           ITER,ISTATE,C,CJAC,CLAMDA,OBJF,OBJGRD,R,X,IWORK,
+           LIWORK,WORK,LWORK,IUSER,USER,IFAIL)
*
*   END IF
*
*   close(7)
*   close(23)
*   close(9)
*
*   STOP
*   END
*
*   SUBROUTINE OBJFUN(MODE,N,X,OBJF,OBJGRD,NSTATE,IUSER,USER)
*   Routine to evaluate objective function and its 1st derivatives.
*   .. Parameters ..
*   DOUBLE PRECISION  ONE, TWO
*   PARAMETER          (ONE=1.0D0,TWO=2.0D0)
*   .. Scalar Arguments ..
*   DOUBLE PRECISION  OBJF
*   INTEGER           MODE, N, NSTATE
*   .. Array Arguments ..

```



```
DOUBLE PRECISION OBJGRD(N), USER(*)
INTEGER          IUSER(*)
```

```
DOUBLE PRECISION Sweep,Area,ARatio,X(N),Mass,Drag
```

```
REAL            Cost,Pitch
CHARACTER*14    CMD,text
```

```
REAL            Swout,AROUT,ARaout
```

```
* ..Parameters and arrays for time monitoring..
```

```
REAL            TIMEARRAY(2)
REAL            ELAPSE
REAL            DELAPSE
```

```
Sweep=X(1)
Area=X(2)
```

```
* Correction for Aspect Ratio
```

```
ARatio=((77.1038)**2/Area)
```

```
write(7,*)'Sweep',Sweep
write(7,*)'Area',Area
write(7,*)'ARatio',ARatio
```

```
*
```

```
* Run MMG with new geometry values and get updated cost
```

```
*
```

```
* Create new updated input file for MMG
```

```
print*, 'Update Geometry Parameters for MMG'
```

```
Swout=Sweep
AROUT=Area
ARaout=ARatio
```



```

*****
*           *
* SUB LEVEL 1 *
*           *
*****
*** Start Mass Estimate, by activating structural sub level
*** this sublevel takes the new Geometry parameters as fixed datum and
*** will minimise the cost by changing internal Layout parameters.
*** After minimum cost is obtained, the minimum mass associated with the
*** minimum cost is handed over to the top level

      call SubLevel1(Sweep,Area,ARatio,Mass,Cost,Pitch)

*****
*           *
* SUB LEVEL 2 *
*           *
*****
*** Start Drag Estimate

      call SubLevel2(Sweep,Area,Drag)

*** write results from both sub-levels

      write(7,*),'Mass',Mass
      write(7,*),'Drag',Drag
      write(7,*),'Cost',Cost
      write(7,*),'Pitch',Pitch

*****
*           *
* Top Level   *
*           *
*****
*** Objective Function

      OBJF=Mass+(2*Drag)

```



```

write(7,*), 'OBJ', OBJF
write(23,*), OBJF, Mass, Drag, Cost
write(9,*), OBJF, Sweep, Area, Pitch
print*, 'OBJ', OBJF, 'Mass', Mass, 'Drag', Drag, 'Cost', Cost

```

```

RETURN
END

```

```

SUBROUTINE CONFUN(MODE, NCNLN, N, LDCJ, NEEDC, X, C, CJAC, NSTATE, IUSER,
+
USER)

```

```

* Routine to evaluate the nonlinear constraints and their 1st
* derivatives.

```

```

* .. Parameters ..

```

```

DOUBLE PRECISION ZERO, TWO

```

```

PARAMETER (ZERO=0.0D0, TWO=2.0D0)

```

```

* .. Scalar Arguments ..

```

```

INTEGER LDCJ, MODE, N, NCNLN, NSTATE

```

```

* .. Array Arguments ..

```

```

DOUBLE PRECISION C(*), CJAC(LDCJ,*), USER(*), X(N)

```

```

INTEGER IUSER(*), NEEDC(*)

```

```

* .. Local Scalars ..

```

```

INTEGER I, J

```

```

* .. Executable Statements ..

```

```

RETURN

```

```

END

```



## A.2 SubLevel1.f

This sub-level minimises the cost for a given wing geometry by changing the internal wing layout (i.e. stringer pitch) Geometry variables, sweep and area are specified by the top-level and remain fixed at this level. As described in section 6.2.2 on page 161 the following steps are performed by this programme:

1. Change internal layout (i.e. stringer pitch)
2. Create MMG input file 'mmg\_in.dat' with new value for internal layout.
3. Run MMG using the script 'mmgscript' to generate FE model
4. Run STARS optimisation using the script 'runstars'
5. Run script 'weight\_script' to retrieve optimised weight results from STARS and write into file 'WeightSTARS.dat'
6. Read optimised weight results from file 'WeightSTARS.dat'
7. Create MMG input files 'mmg\_in.dat' and 'cost\_in.dat'
8. Run MMG using the script 'mmgscript2' to perform cost analysis
9. Read calculated manufacturing cost from the file 'Summary\_Weight\_Cost.dat'
10. Update objective function with new cost result and asses the new objective.
11. If minimum has not been reached, go to 1.

```
*****
* Subroutine SubLevel1(Sweep,Area,ARatio,Mass,Cost)
*****
* PROGRAM MassQInt\_sun
*
* Produced by Kristof Gantois, Cranfield College of Aeronautics
*
* 20 May 1998
*
* Purpose: Minimisation of Cost by changing Layout: Stringer pitch
*
* - uses quadratic interpolation program from NAG Library:
```



```

*           E04ABF Program.
*           Mark 17 Revised.  NAG Copyright 1995
*
* Input/Output with:
*
*   - MDOParadigm.f
*     exchanged variables: Sweep, Area, ARatio, Mass, Cost
*
*   - MDO Software MMG/TDMB
*     and Cranfield Cost model
*
*   - input from files: Summary\_Weight\_Cost.dat
*                       GeoVariants.dat
*                       WeightSTARS.dat
*
*   - output to files: GeoVariants.dat
*                       mmg\_in.dat
*                       cost\_in.dat
*                       CostQIntTUD\_Results.dat
*
* Global Variables: Sweep, Area, ARatio (Aspect Ratio), Mass, cost
*
* Local Variables:
*           Pitch : variable for quatdratic interpolation (i.e. X)
*           A : Lower bound on variable X
*           B : Upper bound on variable X
*           MAXCAL : maximum number of iterations
*           F, FC : Function values
*
* MMG Settings:
*
*   - TUD\_STRESSES = YES
*   - Rib\_Sizing = Yes
*
*
*****
SUBROUTINE SubLevel1(Sweep,Area,ARatio,Mass,Cost,Pitch)

* .. Parameters ..

```



```

INTEGER          NOUT
PARAMETER        (NOUT=6)
*   .. Local Scalars ..
DOUBLE PRECISION A, B, EPS, F, T, X,
+ Sweep, Area, Mass, ARatio
INTEGER          IFAIL, MAXCAL
*   .. External Subroutines ..
EXTERNAL         E04ABF, FUNCT
*   .. Executable Statements ..
REAL            Cost,Pitch

print*, 'Start CostQIntTUD'

****write Sweep, Area, ARatio and Mass in an input file
open(20,file='GeoVariants.dat',form='formatted')
WRITE (20,'(F15.12)')Sweep
WRITE (20,'(F16.12)')Area
WRITE (20,'(F14.12)')ARatio
WRITE (20,'(F8.2)')Mass
close(20)

open(8,file='CostQIntTUD\_Results.dat',form='formatted')
WRITE (8,*) 'Results cost minimisation by changing
+ stringer pitch'
WRITE (8,*) '69 ribs: 5;21;18;24'

*   EPS and T are set to zero so that E04ABF will reset them to
*   their default values
EPS = 0.0D0
T = 0.0D0
*   The minimum is known to lie in the range (0.100, 0.400)
A = 0.100D0
B = 0.400D0
*   Allow 30 calls of FUNCT
*   MAXCAL = 30
MAXCAL = 10

IFAIL = 1

*
* Start Quadratic interpolation
*
```



```
CALL E04ABF(FUNCT, EPS, T, A, B, MAXCAL, X, F, IFAIL)
```

```
*  
WRITE (NOUT,*)  
IF (IFAIL.EQ.1) THEN  
  WRITE (NOUT,*) 'Parameter outside expected range'  
ELSE  
  IF (IFAIL.EQ.2) THEN  
    WRITE (NOUT,*)  
+    'Results after MAXCAL function evaluations are'  
    print*,  
+    'Results after MAXCAL function evaluations are'  
    WRITE (NOUT,*)  
    print*  
  END IF  
  WRITE (NOUT,99999) 'The minimum lies in the interval ', A,  
+  ' to ', B  
  WRITE (NOUT,99999) 'Its estimated position is ', X, ', '  
  WRITE (NOUT,99998) 'where the function value is ', F  
  WRITE (NOUT,99997) MAXCAL, 'function evaluations were required'  
  print*, 'The minimum lies in the interval ', A,  
+  ' to ', B  
  print*, 'Its estimated position is ', X, ', '  
  print*, 'where the function value is ', F  
  print*, MAXCAL, 'function evaluations were required'  
END IF  
  
close (8)
```

```
***Update New Mass
```

```
print*, 'Read Updated Mass which will be used by Top Level'
```

```
open(20, file='GeoVariants.dat', form='formatted')
```

```
READ (20, '(F15.12)') Sweep
```

```
READ (20, '(F16.12)') Area
```

```
READ (20, '(F14.12)') ARatio
```

```
READ (20, '(F8.2)') Mass
```

```
close(20)
```

```
print*, 'Updated Mass', Mass
```

```
* update cost value
```



```
Cost=F
Pitch=X
```

```
RETURN
```

```
*
99999 FORMAT (1X,A,F10.8,A,F10.8)
99998 FORMAT (1X,A,F7.4)
99997 FORMAT (1X,I2,1X,A)
END
```

```
*
```

```
SUBROUTINE FUNCT(XC,FC)
```

```
* Routine to evaluate F(x) at any point in (A, B)
```

```
* .. Scalar Arguments ..
```

```
DOUBLE PRECISION FC, XC
```

```
DOUBLE PRECISION Cost,Total,MP1skin,MP1spar,MP1string,
+ MP1rib,Sweep,Area,Mass
CHARACTER*30 CMD,text
```

```
*
```

```
* Run MMG with new variant values and get updated cost
```

```
*
```

```
* Read new top level Sweep, Area and Aspect Ratio value
```

```
print*, 'Read top level geometry variables'
open(20,file='GeoVariants.dat',form='formatted')
READ (20,'(F15.12)')Sweep
READ (20,'(F16.12)')Area
READ (20,'(F14.12)')ARatio
close(20)
print*, 'Top level Sweep',Sweep
print*, 'Top level Area',Area
print*, 'Top level Aspect Ratio',ARatio
```



```

* Create new updated input file for MMG
  print*, '*****New Cost function evaluation*****'
  print*, 'Update MMG input file with new stringer pitch'
  print*, 'Striger pitch = ', XC
  write(8,*)
+ '*****New Cost function evaluation*****'
  write(8,*) 'Striger pitch = ', XC

Open(10, file='mmg\_in.dat', form='formatted')
write(10, '(A4)') 'SLLS'
write(10, '(A9)') 'slls-comp'
write(10, '(A3)') 'FEG'
write(10, '(A8)') 'feg-comp'
write(10, '(A6)') 'Costs?'
write(10, '(A2)') 'No'
write(10, '(A13)') 'CompositeWing'
write(10, '(A2)') 'No'
write(10, '(A11)') 'TUDStresses'
write(10, '(A3)') 'Yes'
write(10, '(A9)') 'RibSizing'
write(10, '(A3)') 'Yes'
write(10, '(A12)') 'Wing QCSweep'
write(10, '(F15.12)') Sweep
write(10, '(A9)') 'Wing Area'
write(10, '(F16.12)') Area
write(10, '(A16)') 'Wing AspectRatio'
write(10, '(A3)') ARatio
write(10, '(A12)') 'Wing CrankTc'
write(10, '(A4)') '0.10'
write(10, '(A21)') 'Wing TipSection Twist'
write(10, '(A6)') '-1.638'
write(10, '(A14)') 'Wing CrankRSXc'
write(10, '(A4)') '0.65'
write(10, '(A11)') 'Wing RootTc'
write(10, '(A4)') '0.14'
write(10, '(A16)') 'TopStringerPitch'
write(10, '(F8.6)') XC
write(10, '(A16)') 'BotStringerPitch'
write(10, '(F8.6)') XC
write(10, '(A11)') 'NRibscentre'

```



```
write(10,'(A1)')'5'  
write(10,'(A8)')'NRibsIbd'  
write(10,'(A2)')'21'  
write(10,'(A8)')'NRibsObd'  
write(10,'(A2)')'18'  
write(10,'(A8)')'NRibsTip'  
write(10,'(A2)')'24'  
Close(10)
```

\* Run MMG

```
print*,'Start MMG'
```

```
CMD='mmgscript'  
I=SYSTEM(CMD)
```

```
print*,'End MMG'
```

\* Start STARS Fully Stressing OPTIMISATION

```
print*,'BEGIN script runstars for STARS Fully Stressing  
+ Optimisation'  
CMD='runstars'  
I=SYSTEM(CMD)  
print*,'END script runstars and STARS optimisation'
```

```
print*,'Calculate weight breakdown from STARS results'  
CMD='weight\_script'  
I=SYSTEM(CMD)  
print*,'End weight breakdown calculation'
```

\* read weight breakdown

```
open(289,file='WeightSTARS.dat',form='formatted')  
read(289,*)  
read(289,'(14X,F8.2)')MP1skin  
read(289,'(14X,F8.2)')MP1spar  
read(289,'(14X,F8.2)')MP1string  
read(289,'(14X,F8.2)')MP1rib
```



```

read(289,'(14X,F8.2)')Total
close(289)
write(8,*)'TOTAL Weight =',Total

```

\* Prepare MMG input file to initiate cost analysis  
 print\*, 'Set input file MMG to do Cost analysis with  
 + STARS weight breakdown '

```

Open(10,file='mmg\_in.dat',form='formatted')
write(10,'(A4)')'SLLS'
write(10,'(A9)')'slls-comp'
write(10,'(A3)')'FEG'
write(10,'(A8)')'feg-comp'
write(10,'(A6)')'Costs?'
write(10,'(A2)')'Yes'
write(10,'(A13)')'CompositeWing'
write(10,'(A2)')'No'
write(10,'(A9)')'RibSizing'
write(10,'(A3)')'Yes'
write(10,'(A11)')'TUDStresses'
write(10,'(A3)')'Yes'
write(10,'(A12)')'Wing QCSweep'
write(10,'(F15.12)')Sweep
write(10,'(A9)')'Wing Area'
write(10,'(F16.12)')Area
write(10,'(A16)')'Wing AspectRatio'
write(10,'(A3)')ARatio
write(10,'(A12)')'Wing CrankTc'
write(10,'(A4)')'0.10'
write(10,'(A21)')'Wing TipSection Twist'
write(10,'(A6)')'-1.638'
write(10,'(A14)')'Wing CrankRSXc'
write(10,'(A4)')'0.65'
write(10,'(A11)')'Wing RootTc'
write(10,'(A4)')'0.14'
write(10,'(A16)')'TopStringerPitch'
write(10,'(F8.6)')XC
write(10,'(A16)')'BotStringerPitch'
write(10,'(F8.6)')XC
write(10,'(A11)')'NRibscentre'

```



```
write(10,'(A1)')'5'  
write(10,'(A8)')'NRibsIbd'  
write(10,'(A2)')'21'  
write(10,'(A8)')'NRibsObd'  
write(10,'(A2)')'18'  
write(10,'(A8)')'NRibsTip'  
write(10,'(A2)')'24'  
Close(10)
```

```
open(29,file='cost\_in.dat',form='formatted')  
write(29,'(A1)')'4'  
write(29,'(A1)')'2'  
write(29,'(A2)')'99'  
Close(29)
```

\* put weight data in pegasus, delete stars data and rerun mmg

```
print*,'cp WeigthSTARS, mmg\_in.dat and cost\_in.dat;  
+ Start MMG'
```

```
CMD='mmgscript2'  
I=SYSTEM(CMD)
```

\* Read updated cost value in sqp program

```
print*,'Get Manufacturing Cost from Summary\_Weight\_Cost.dat'
```

```
70 open(11,file='Summary\_Weight\_Cost.dat',form='formatted')  
READ (11,'(19x,A19)')text  
if (text.eq.'DManCostTotalMetal=') then  
READ (11,'(23x,A15,11x,F12.3)')text,Cost  
print*,Cost  
write(8,*)'Cost =',Cost  
write(8,*)'STARS Weight breakdown'  
write(8,*)'Skin weight =',MP1skin  
write(8,*)'Spar weight =',MP1spar  
write(8,*)'Stringer weight =',MP1string  
write(8,*)'Rib weight =',MP1rib  
else
```



```
goto 70
endif
close(11)
```

```
* Write Updated Mass to file
  open(20,file='GeoVariants.dat',form='formatted')
  WRITE (20,'(F15.12)')Sweep
  WRITE (20,'(F16.12)')Area
  WRITE (20,'(F14.12)')ARatio
  WRITE (20,'(F8.2)')Total
  close(20)
```

```
*** End Cost Estimate
```

```
* Update function with new total cost.
```

```
FC = Cost
```

```
RETURN
END
```



### A.3 SubLevel2.f

Response surfaces data for the CFD drag results was used as input to the paradigm. Hence, this sub-level does not actually contain a CFD solver but only a routine to interpolate between the provided CFD data points.

```
*****
* Subroutine SubLevel2(Sweep,Area,Drag)
*****
*
* Produced by Kristof Gantois, Cranfield College of Aeronautics
*
* 1 May 1998
*
* Purpose: Interpolate between Drag data points for given sweep and area
*
*   - Interpolation program from NAG Library:
*     E01DAF Program: Interplating functions, fitting bicubic spline,
*     data on rectangular grid
*     Mark 17 Revised.  NAG Copyright 1995
*
* Input/Output with:
*
*   - MDOParadigm.f
*     exchanged variables: Sweep, Area, Drag
*
*   - input from files: DragResSurfBAe.dat (This file contains drag resu
*     and control values for the NAG programmo)
*
* Global Variables: Sweep, Area, Drag
*
*
*****
      subroutine SubLevel2(Sweep,Area,Drag)

*   E01DAF Example Program Text
*   Mark 14 Release.  NAG Copyright 1989.
*   .. Parameters ..
      INTEGER          NIN, NOUT
      PARAMETER        (NIN=5,NOUT=6)
```



```

INTEGER          MXMAX, MYMAX
PARAMETER        (MXMAX=20,MYMAX=MXMAX)
INTEGER          LIWRK, LWRK
PARAMETER        (LIWRK=MXMAX+2*(MXMAX-3)*(MYMAX-3),LWRK=(MXMAX+6)
+                *(MYMAX+6))
*
* .. Local Scalars ..
DOUBLE PRECISION STEP, XHI, XLO, YHI, YLO
DOUBLE PRECISION Sweep,Area,Drag
INTEGER          I, IFAIL, J, MX, MY, NX, NY, PX, PY
*
* .. Local Arrays ..
DOUBLE PRECISION C(MXMAX*MYMAX), F(MXMAX*MYMAX), FG(MXMAX*MYMAX),
+                LAMDA(MXMAX+4), MU(MYMAX+4), TX(MXMAX),
+                TY(MYMAX), WRK(LWRK), X(MXMAX), Y(MYMAX)
INTEGER          IWRK(LIWRK)
CHARACTER*10     CLABS(MYMAX), RLABS(MXMAX)
*
* .. External Subroutines ..
EXTERNAL         E01DAF, E02DFF, X04CBF
*
* .. Intrinsic Functions ..
INTRINSIC        MAX, MIN
*
* .. Executable Statements ..

open(NIN,file='DragResSurfBAe.dat',form='formatted')

print*, 'Sublevel2'

WRITE (NOUT,*) 'DragResSurfBAe Program Results'
*
* Skip heading in data file
READ (NIN,*)
*
* Read the number of X points, MX, and the values of the
* X co-ordinates.
READ (NIN,*) MX
READ (NIN,*) (X(I),I=1,MX)
*
* Read the number of Y points, MY, and the values of the
* Y co-ordinates.
READ (NIN,*) MY
READ (NIN,*) (Y(I),I=1,MY)
*
* Read the function values at the grid points.
DO 20 J = 1, MY
    READ (NIN,*) (F(MY*(I-1)+J),I=1,MX)
20 CONTINUE
    IFAIL = 0

```



```

*
* Generate the (X,Y,F) interpolating bicubic B-spline.
CALL E01DAF(MX,MY,X,Y,F,PX,PY,LAMDA,MU,C,WRK,IFAIL)
*
* Print the knot sets, LAMDA and MU.
WRITE (NOUT,*)
WRITE (NOUT,*)
+ '          I      Knot LAMDA(I)          J      Knot MU(J)'
DO 40 J = 4, MAX(PX,PY) - 3
  IF (J.LE.PX-3 .AND. J.LE.PY-3) THEN
    WRITE (NOUT,99997) J, LAMDA(J), J, MU(J)
  ELSE IF (J.LE.PX-3) THEN
    WRITE (NOUT,99997) J, LAMDA(J)
  ELSE IF (J.LE.PY-3) THEN
    WRITE (NOUT,99996) J, MU(J)
  END IF
40 CONTINUE
* Print the spline coefficients.
WRITE (NOUT,*)
WRITE (NOUT,*) 'The B-Spline coefficients:'
WRITE (NOUT,99999) (C(I),I=1,MX*MY)
WRITE (NOUT,*)
* Evaluate the spline on a regular rectangular grid at NX*NY
* points over the domain (XLO to XHI) x (YLO to YHI).
READ (NIN,*) NX
READ (NIN,*) NY

XLO=Sweep-0.01
XHI=Sweep
YLO=Area-0.01
YHI=Area

IF (NX.LE.MXMAX .AND. NY.LE.MYMAX) THEN
  STEP = (XHI-XLO)/(NX-1)
  DO 60 I = 1, NX
*   Generate NX equispaced X co-ordinates.
    TX(I) = MIN(XLO+(I-1)*STEP,XHI)
*   Generate X axis labels for printing results.
    WRITE (CLABS(I),99998) TX(I)
60 CONTINUE
  STEP = (YHI-YLO)/(NY-1)

```



```

      DO 80 I = 1, NY
        TY(I) = MIN(YLO+(I-1)*STEP, YHI)
        WRITE (RLABS(I),99998) TY(I)
80    CONTINUE
*
*    Evaluate the spline.
      CALL E02DFE(NX,NY,PX,PY,TX,TY,LAMDA,MU,C,FG,WRK,LWRK,IWRK,
+         LIWRK,IFAIL)

* print interpolation result for point of interest = 4th point of mesh

      print*, 'Interpolation for Sweep =', TX(2), 'and Area =', TY(2), ':'
      print*, 'Drag =', FG(4)
      Drag=FG(4)

*
*    Print the results.
      CALL X04CBF('General', 'X', NY, NX, FG, NY, 'F8.3',
+         'Spline evaluated on a regular mesh (X across, Y down):'
+         , 'Character', RLABS, 'Character', CLABS, 80, 0, IFAIL)
*
      END IF
      close(NIN)
      return

*
99999 FORMAT (1X,8F9.4)
99998 FORMAT (F5.2)
99997 FORMAT (1X,I16,F12.4,I11,F12.4)
99996 FORMAT (1X,I39,F12.4)
      END

```



## A.4 mmg\_in.dat

This file is the main input file for the Multi-Model Generator (MMG). Depending on which script is used to run MMG, user interaction or an input file is required. For the MDO paradigm, all information for the MMG is given through input files. As can be seen directly below, the file mmg\_in.dat contains a list of control/design variables and their values:

\*\*\*\*\*

```
SLLS
slls-comp
FEG
feg-comp
Costs?
No
CompositeWing
No
TUDStresses
Yes
RibSizing
Yes
Wing QCSweep
34.0
Wing Area
625.0
Wing AspectRatio
8.2
Wing CrankTc
0.10
Wing TipSection Twist
-1.638
Wing CrankRSXc
0.65
Wing RootTc
0.14
TopStringerPitch
0.243
BotStringerPitch
0.243
NRibscentre
```



5  
NRibsIbd  
21  
NRibsObd  
18  
NRibsTip  
24



## A.5 mmgscript

This script performs the following tasks:

1. Copy file mmg\_in.dat to the directory where the Multi-Model Generator (MMG) will be run.
2. Go to the MMG directory.
3. Set environment variable to enable execution of MMG.
4. Run MMG by activating the script 'mmgkg'
5. Change names of input files for STARS.
6. Copy all the relevant files to the directory where STARS and the MDO paradigm are run.

```
*****
```

```
#!/bin/csh -f
```

```
echo " stars mmgscript"
```

```
cp mmg\_in.dat ../../../MDO-Software/data
```

```
cd ../../../MDO-Software/data
```

```
echo "define environment variable"  
source csh\_login
```

```
echo "run mmg"
```

```
mmgkg
```

```
echo "mmg terminated"
```

```
echo "change name of STARS input files"  
mv fegnsin.dat MDOuain.dat  
mv fegcons.dat MDOcons.dat
```



```
echo "copy files"
cp Summary\_Weight\_Cost.dat ../../UAI/DERA/MDOParadigm
cp RibID.dat ../../UAI/DERA/MDOParadigm/Stars
cp SkinID.dat ../../UAI/DERA/MDOParadigm/Stars
cp SparID.dat ../../UAI/DERA/MDOParadigm/Stars
cp StringerID.dat ../../UAI/DERA/MDOParadigm/Stars
cp MDOcons.dat ../../UAI/DERA/MDOParadigm/Stars
cp MDOuain.dat ../../UAI/DERA/MDOParadigm/Stars

echo "End script login test"
```

## A.6 mmgscript2

This script is used to run the cost model of MMG. It is nearly the same as 'mmgscript' apart from two additional files which also need to be transferred to the directory where MMG is executed. The file 'cost\_in.dat' contains settings for the cost model and 'WeightSTARS.dat' contains the optimised component weight breakdown for the cost model. In addition, to enable the cost model the parameter 'Cost?' is set equal to 'YES' in the 'mmg\_in.dat' file.

```
*****
```

```
#!/bin/csh -f
```

```
echo " stars mmgscript"
```

```
cp mmg\_in.dat ../../MDO-Software/data
cp cost\_in.dat ../../MDO-Software/data
cp WeightSTARS.dat ../../MDO-Software/data
```

```
cd ../../MDO-Software/data
```

```
echo "define environment variable"
source csh\_login
```

```
echo "run mmg"
```

```
mmgkg
```

```
echo "mmg terminated"
```

```
echo "change name of STARS input files"
mv fegnsin.dat MDOuain.dat
mv fegcons.dat MDOcons.dat
```

```
echo "copy files"
cp Summary\_Weight\_Cost.dat ../../UAI/DERA/MDOParadigm
cp RibID.dat ../../UAI/DERA/MDOParadigm/Stars
cp SkinID.dat ../../UAI/DERA/MDOParadigm/Stars
```



```
cp SparID.dat ../../UAI/DERA/MDOParadigm/Stars
cp StringerID.dat ../../UAI/DERA/MDOParadigm/Stars
cp MDOcons.dat ../../UAI/DERA/MDOParadigm/Stars
cp MDOuain.dat ../../UAI/DERA/MDOParadigm/Stars

echo "End script "
```

## A.7 runstars

Script to setup and execute STARS. The following steps are carried out:

1. Go into STARS directory.
2. Remove old STARS files.
3. Run STARS, i.e. specify ID (MDO), give the 'run' command and quit STARS when analysis is terminated.

```
*****
```

```
#!/bin/csh -f
```

```
echo "Remove old STARS files"
```

```
# Move, Delete files from stars & mmg run into appropriat directory  
@ count = 1
```

```
cd Stars/
```

```
rm MDO\_STARS.00  
rm MDO\_STARS.EDB  
rm MDOdust.dat  
rm MDOfile.dat  
rm MDOgraf.dat  
rm MDOnast.dsp  
rm MDOnast.ept  
rm MDOnast.esm  
rm MDOnast.kdc  
rm MDOuast.bd*  
rm MDOfeg.out  
rm MDOrslt.dat  
rm MDOuast.out  
rm MDOwarm.dat  
rm MDOuast.prt  
rm MDOsave.dat
```

```
echo "Start STARS run"
```



```
/applics/stars/com/stars << "eof"  
MDO  
run  
quit  
"eof"  
  
echo "STARS run terminated"
```

# Appendix B

## Manufacturing Modules in TDMB

This appendix describes the manufacturing modules which have been added to the Technical Data Modeller and Browser (TDMB) in order to incorporate the cost models into the MDO software. Detailed information on how to run the TDMB and MMG can be found in the online help pages.

### B.1 TDMB Specification Section

Figure B.1 shows the main menu which is obtained when TDMB is run. As part of the aircraft specifications there is a manufacturing cost node which contains all the required reference cost and manufacturing data which is required for the cost models

Under this manufacturing node, six sub-nodes are present:

- **RefCostBreakdown**  
Contains the
- **FeatureCost**  
Contains the cost factors for the metal cost model. These factors can be typed in directly or calculated with the cost factor estimation programme.
- **ProductCost**  
Contains a breakdown of all the processes for the composite cost model.
- **RefManInfoMetal**  
Contains reference manufacturing information for metal wing. (i.e. fastener diameters, etc)



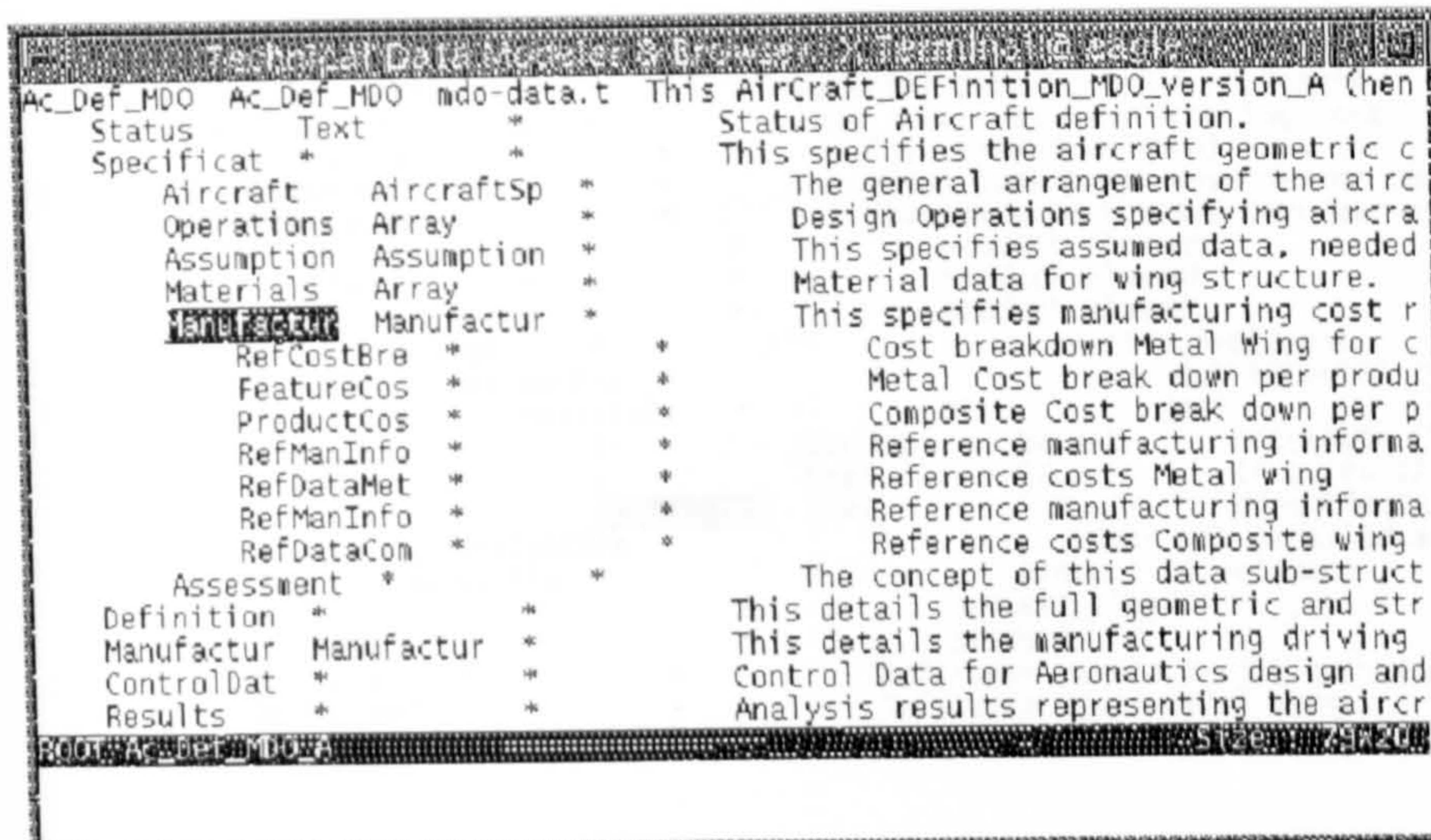


Figure B.1: Main menu in TDMB.

- **RefDataMetal**  
Contains reference costs for metal cost model, such as material costs.
- **RefManInfoComposite**  
Contains reference manufacturing information for composite wing. (i.e. fastener diameters, number of stiffeners per rib, etc)
- **RefDataComposite**  
Contains reference costs for composite cost model, such as material costs, machine and labour costs.

Looking first at data for the composite cost model, expanding the node 'ProductCost' gives a list of the different composite components (see figure B.2). Each component can be broken down into material cost (MatCost-Breakdown) and detailed manufacturing cost (DetCostBreakdown). An example is given in figure B.2 for the spar web. 'Material1' is the lowest level to which the material cost can be broken down. To the node 'Material1' three parameters are defined: a feature (i.e. weight), a cost factor and a weighting factor. The weighting factor can vary from 0 to 1.

When expanding the detailed manufacturing cost node, a list of detailed manufacturing processes which are associated to the component in question appears (see figure B.3). Each process contains again information on the features, cost factors and weight factor associated with the process. As can be seen from the figure, the 'Debug' process which is part of the curing has



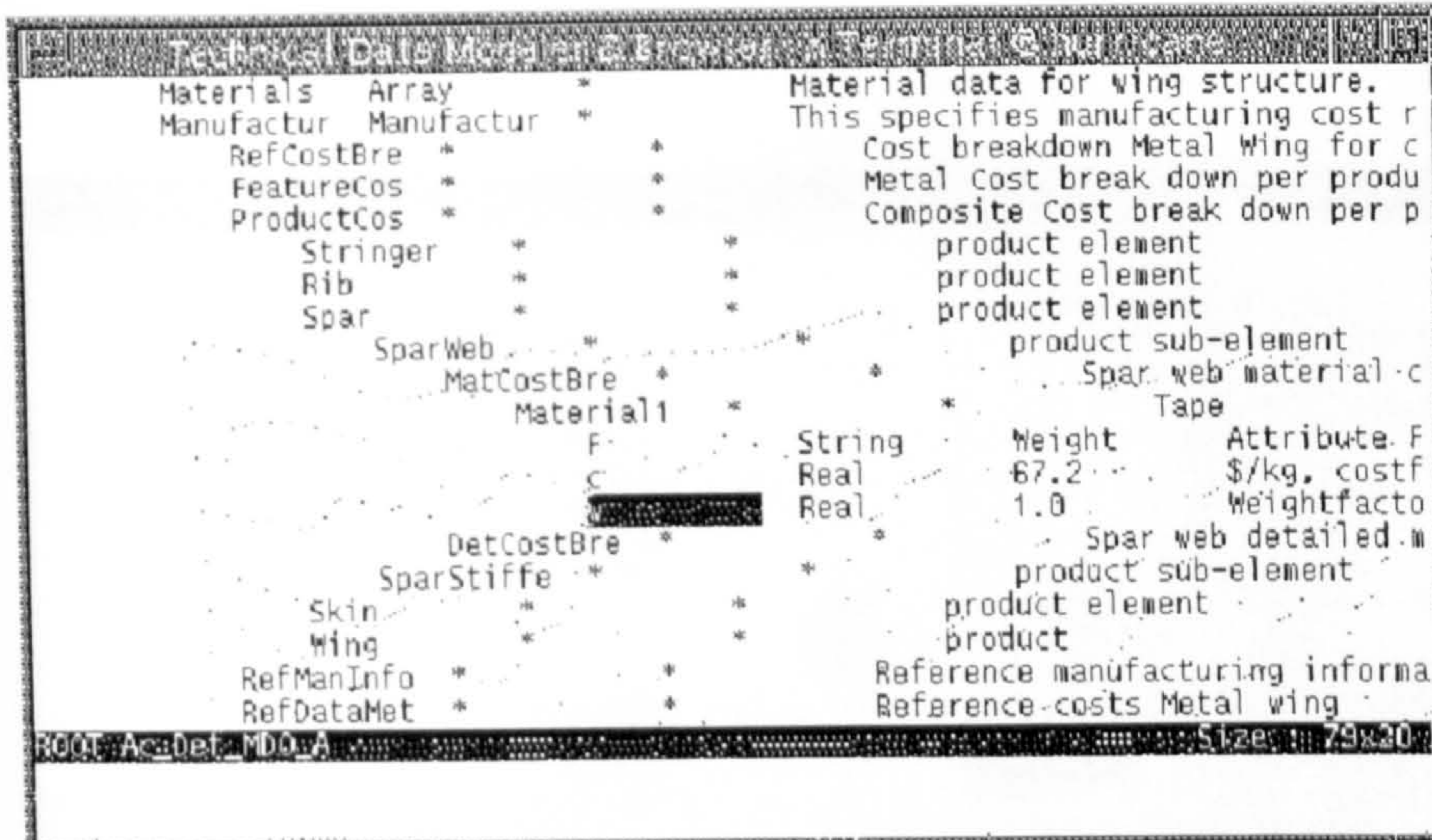


Figure B.2: Breakdown of composite components, spar web.

the component area as geometry feature and a cost factor for this process which is defined in  $h/m^2$ .

For the calculation of the composite costs, additional cost information is needed on material cost, cost for tape lay-up, cobonding, curing etc. This cost data is being defined under the section 'RefDataComposite' and is displayed in figure B.4.

The specifications for the metal cost modal are defined and structured in a different way. As described in chapter 5 a cost factor estimation programme was developed to define metal cost factors. Figure B.5 shows the 'Reference Cost Breakdown' section where the initial cost distributions for material, assembly and detailed manufacturing cost are defined. As discussed in chapter 5 these are used for the cost factor estimation. This distribution is defined for each component. The cost drivers for the metal cost model are defined in the 'Feature Cost' section. As was explained, the prototype metal cost model only uses one key cost driver for each component and does not have a detailed breakdown of all the manufacturing processes. This can also be seen in figure B.5. The detailed manufacturing cost of a stringer, for example, is driven by the component length (i.e. Parameter1 in the figure). For this cost driver, a cost factor and weight factor have been allocated (see figure B.5). The cost factor is calculated using the cost factor estimation programme or can be defined by the user.



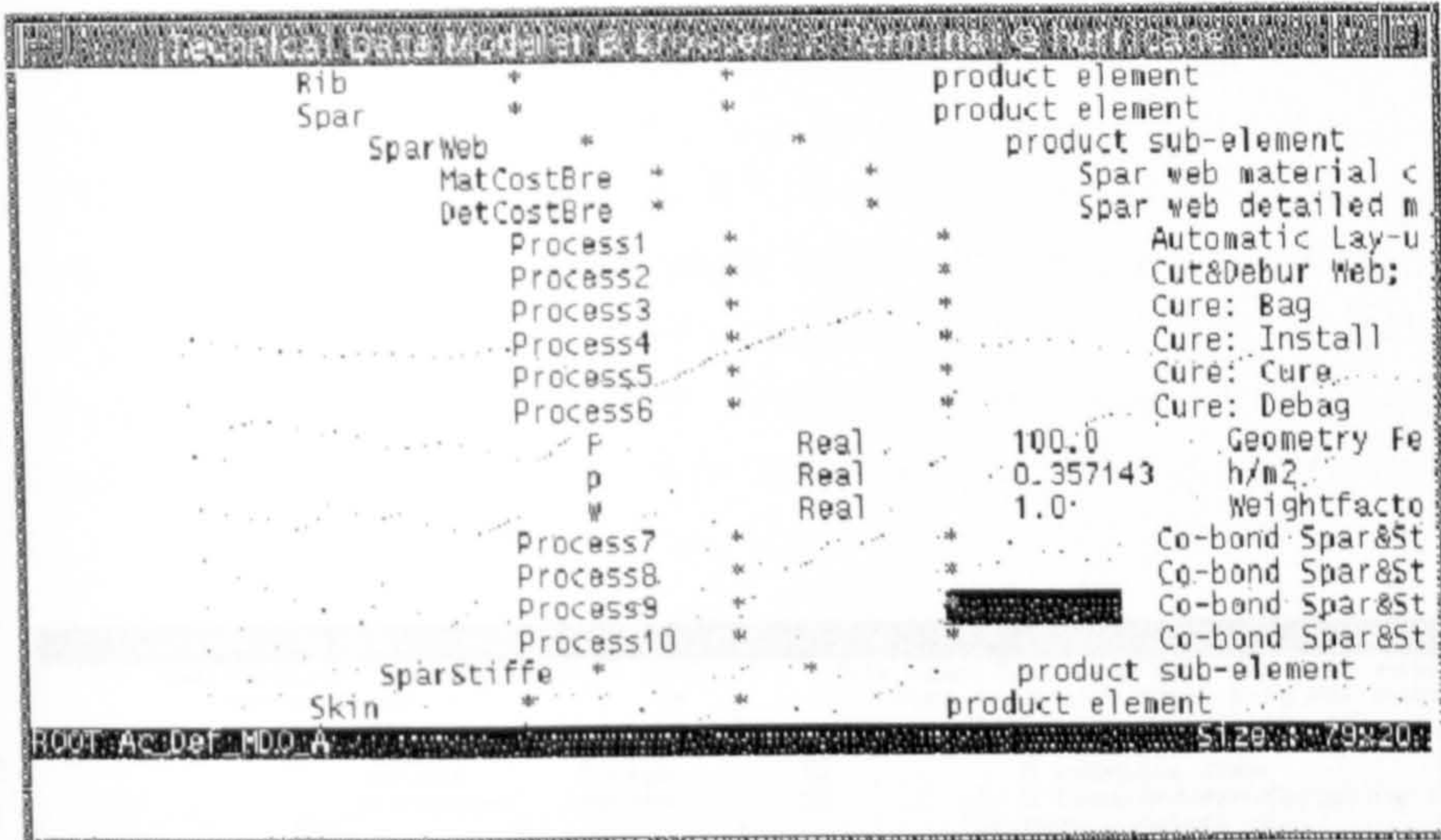


Figure B.3: Spar web, breakdown of manufacturing processes.

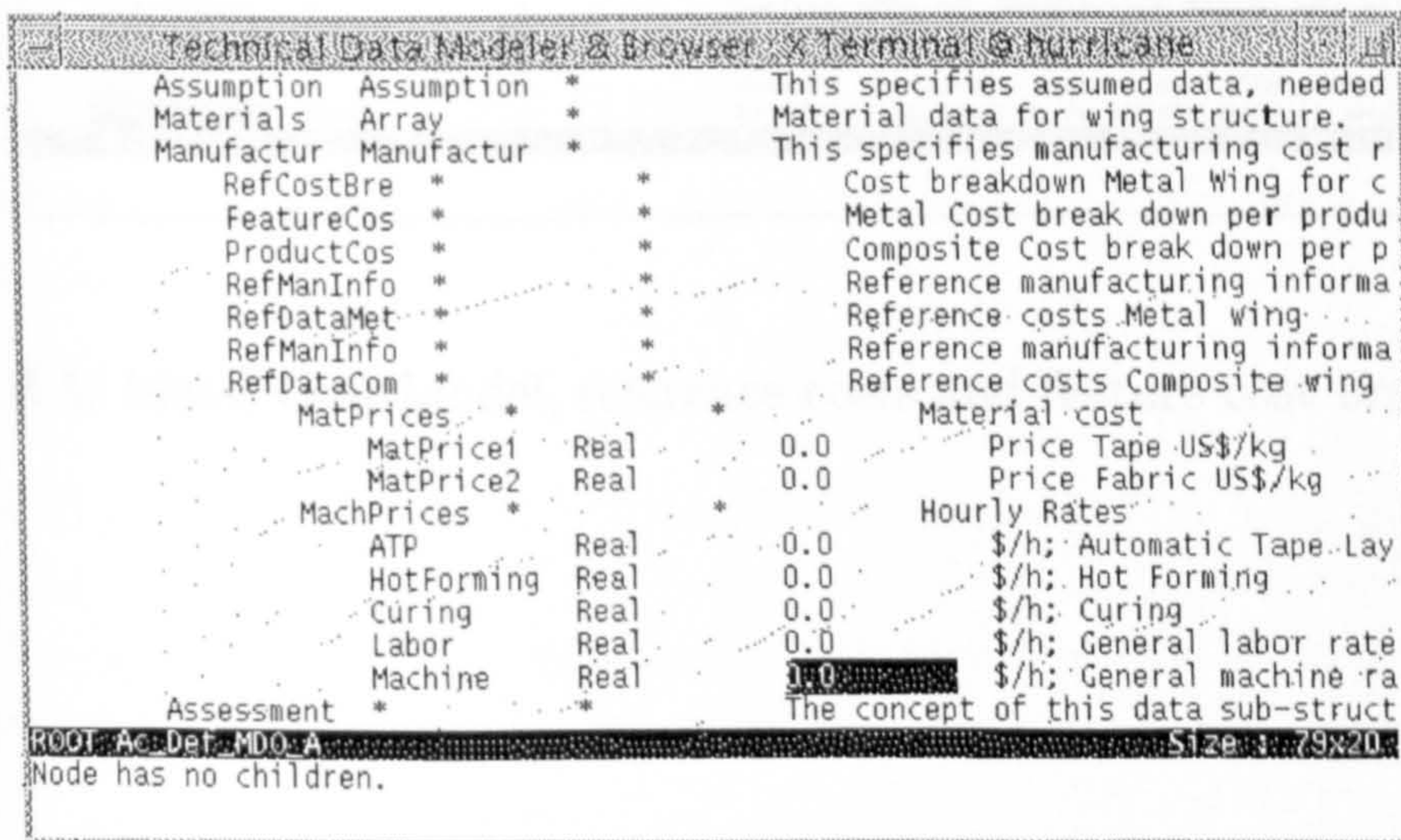


Figure B.4: Breakdown of composite components, spar web.



## B.2 TDMA Manufacturing Section

The purpose of this section is to define all data elements which have to be defined in the special data model. The manufacturing information which normally stays constant irrespective of the design model. The manufacturing section contains all the data which is normally generated for a specific aircraft model. The data is generated for a design parameter is changed and the flight performance is then calculated. This section can be broken down into two main sections: 'Manufacturing' and 'CostModel'. The 'Manufacturing' section contains the physical manufacturing data for each structural element. The 'CostModel' section contains the data which is used to calculate the cost of the aircraft.

| Technical Data Model & Browser - X Terminal @ Toronto |               |          |  | PRG 0                                      |
|---|---------------|----------|--|--|
| Manufacturing   | Manufacturing | *        |  | This specifies manufacturing cost refer    |
| RefCostBreak  | *             | *        |  | Cost breakdown Metal wing for calc         |
| Stringers   | *             | *        |  | Stringer cost consists of:                 |
| MatCost   | Integer       | 30       |  | * Material cost                            |
| AssCost   | Integer       | 30       |  | * Assembly cost                            |
| DetManCost  | Integer       | 30       |  | * Detailed Manufacturing c                 |
| Ribs  | *             | *        |  | Rib cost consists of:                      |
| Spars   | *             | *        |  | Spar cost consists of:                     |
| Skins   | *             | *        |  | Skin cost consists of:                     |
| FeatureCostP  | *             | *        |  | Metal Cost break down per product          |
| Stringer  | *             | *        |  | *  |
| MatCostParam  | *             | *        |  | Stringer material cost per                 |
| AssCostParam  | *             | *        |  | Stringer assembly cost per                 |
| DetManCostPa  | *             | *        |  | Stringer detailed manufact                 |
| Parameter   | *             | *        |  | length                                     |
| c   | Real          | 160.1374 |  | I/n. costfactor                            |
| w   | Real          | 1.0      |  | Weightfactor: give                         |
| Rib   | *             | *        |  | *  |
| Spar  | *             | *        |  | *  |
| Skin  | *             | *        |  | *  |
| ProductCostP  | *             | *        |  | Composite cost break down per prod         |
| RefManInfoMe  | *             | *        |  | Reference manufacturing informatio         |
| RefDataMetal  | *             | *        |  | Reference costs Metal wing                 |
| RefManInfoCo  | *             | *        |  | Reference manufacturing informatio         |
| RefDataCompo  | *             | *        |  | Reference costs Composite wing             |
| assessment  | *             | *        |  | The concept of this data sub-structure     |
| Definition  | *             | *        |  | This details the full geometric and struct |
| Manufacturing   | Manufacturing | *        |  | This details the manufacturing driving per |
| ControlData   | *             | *        |  | Control Data for aeronautics design and an |
| Results   | *             | *        |  | Analysis results representing the aircraft |
| Plots   | *             | *        |  | Plots of the reference data                |
| assessment  | *             | *        |  | Overall assessment against specified assas |
| application   | Text          | *        |  | External application to invoke product ana |
| 2001  | to Def_M001   |          |  | Size 1.88v1                                |

Figure B.5: Metal Cost Model, reference costs and feature cost breakdown.

The 'CostModel' section contains all the results of the cost model. It contains the manufacturing information, the definition of how the cost results have been calculated and the results of the cost model. The results of this section are being used to calculate the overall cost of the aircraft.



## B.2 TDMB Manufacturing Section

The previous section described all the information which has to be defined in the specification section. It contains information which normally stays constant irrespective of changes in the wing model. The manufacturing section contains all the data and results which have been generated for a specific aircraft model. This data is updated each time a design parameter is changed and the Multi Model Generator has been executed. This section can be broken down in two main sections: 'Components' and 'CostModel'. The 'Components' section contains all the required manufacturing data for each structural component such as the all the cost drivers (weight, length, area, number of stringer cut outs, number of fasteners, etc.). This is shown in figure B.6 for the stringer components.

| Control Data for Aeronautics design and Analysis |              |         |   |
|--|--------------|---------|---|
| Definition *                                     |              |         | This details the full geometric and str |
| Manufactur                                       | Manufactur * |         | This details the manufacturing driving  |
| Components *                                     |              | *       | Breakdown of wing components            |
| RibCompone *                                     |              | *       | *                                       |
| Skins *  |              | *       | *                                       |
| StringerCo *                                     |              | *       | *                                       |
| NumberStri                                       | Integer      | 38.0    | Total Number of Stringers               |
| MassString                                       | Real         | 5828.03 | Total Mass Stringers Metal              |
| MassString                                       | Real         | 0       | Total Mass Stringers Compos             |
| LengthStri                                       | Real         | 831.831 | Total Assembly Length Strin             |
| LengthStri                                       | Real         | 0       | Total Assembly Length Strin             |
| NumberFast                                       | Integer      | 28657   | Total Number of Fasteners n             |
| NumberFast                                       | Integer      | 0       | Total Number of Fasteners n             |
| Topskin *  | *            |         | Stringers in top skin                   |
| Botskin *  | *            |         | Stringers in bottom skin                |
| Spars *  | *            | *       | *                                       |
| Fasteners *                                      | *            | *       | Summary of Number of Fasteners          |
| CostModel *                                      | *            | *       | *                                       |
| ControlDat *                                     |              |         | Control Data for Aeronautics design and |
| Results *  |              |         | Analysis results representing the aircr |
| ROOT: Ac-Dat-MDO-A Size: 7/8/20                  |              |         |   |
| Node is already condensed.                       |              |         |   |

Figure B.6: Manufacturing section, component breakdown.

The 'CostModel' section contains all the results of the cost models, for both metal and composites. An overview of how the cost results have been broken down can be seen in figure B.7. The results of this section are being updated, each time the cost model is run.



| Aeronautics Data Model - Cost Model - Results |      |                   |   |
|---|------|-------------------|---|
| CostModel                                     | *    | *                 | *                                       |
| CostRefere                                    | *    | *                 | Results Recurrent Cost Estimate         |
| CostMetal                                     | *    | *                 | Results Recurrent Cost Estimate         |
| MaterialCo                                    | *    | *                 | Material Cost Metal wing                |
| AssemblyCo                                    | *    | *                 | Assembly Cost Metal wing                |
| DetailedMa                                    | *    | *                 | Detailed Manufacturing Cost             |
| Skin  | Real | 812575            | Detailed Manufacturing                  |
| Rib   | Real | 651562            | Detailed Manufacturing                  |
| Spar  | Real | 158393            | Detailed Manufacturing                  |
| Stringer                                      | Real | 281387            | Detailed Manufacturing                  |
| TOTAL   | Real | 1.90392e+0        | Total Detailed Manufact                 |
| TOTALCOSTM                                    | Real | <b>5.34684e+0</b> | Total Metal Recurrent Cost              |
| CostCompos                                    | *    | *                 | Results Recurrent Cost Estimate         |
| CostVarian                                    | *    | *                 | Results Recurrent Cost Estimate         |
| ControlDat                                    | *    | *                 | Control Data for Aeronautics design and |
| Results                                       | *    | *                 | Analysis results representing the aircr |
| Plots   | *    | *                 | Plots of the reference data.            |
| Assessment                                    | *    | *                 | Overall assessment against specified as |
| Applicatio                                    | Text | *                 | External application to invoke product  |
| <b>ROOT: Ac_Dat_MDO_A Size: 7920</b>          |      |                   |   |
| Node is already condensed.                    |      |                   |   |

Figure B.7: Display of the cost model results in TDMB.