UNIVERSITY OF SOUTHAMPTON

# A structural design process for a next generation aerospace design environment

by

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ABSTRACT

FACULTY OF ENGINEERING, SCIENCE AND MATHEMATICS
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The current structural sizing process used to design military aircraft was developed when the emphasis was on the design of the most advanced products possible, with the customer bearing the associated risks of its development. However the marketplace has evolved into where the customer expects 'better, cheaper, faster' products and at a lower degree of risk. It is not clear if the current structural design processes meet the needs of this type of market.

This work argues that the current proprietary process should be replaced by one that is more flexible, allowing the company to adapt its current structural sizing process to meet the needs of a particular product. It includes a study of the current and future engineering environment within a 'typical' airframe design organisation. It looked at the current use of structural optimisation technology throughout the design lifecycle and identified barriers to the potential benefits of wider use. Two existing elements of the organisation's in-house toolset were adapted to size components and the results compared against the literature. This provided an insight into the toolset and the development of proprietary tools. Finally a multilevel 'global-local' sizing approach was developed and studied as an alternative to the current, more tightly coupled, somewhat 'monolithic', sizing system. Strength, stability and stiffness design criteria were considered. Automation of the process was also considered and compared against the existing sizing process. It was found that the current labour intensive sizing process could be improved upon using some simple techniques. Based on this a future structural sizing process is suggested which could be implemented using in-house or commercially available tools.

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# Symbology

| | |
|---|---|
| $\alpha_i$ | Element scale factor |
| $\epsilon$ | In-plane strain |
| $\gamma$ | Shear strain |
| $\lambda_j$ | Lagrange multiplier for constraint $j$ |
| $\sigma$ | Compressive or tensile stress |
| $\tau_i$ | Shear stress in an element $i$ |
| $\mathbf{A}$ | Extensional stiffness matrix |
| $a$ | Panel or element width |
| $b$ | Panel or element length |
| $C_{j_{target}}$ | Target displacement for constraint $j$ |
| $C_j$ | Actual displacement of constraint $j$ |
| $D$ | Hessian matrix used in calculation to find new estimates of Lagrange multipliers |
| $\mathbf{D}$ | Bending stiffness matrix |
| $E_1$ | Modulus of elasticity in the direction of axis system 1 |
| $E_2$ | Modulus of elasticity in the direction of axis system 2 |
| $\epsilon 1_{ua}$ | Ultimate allowable strain the direction of axis 1 |
| $\epsilon 2_{ua}$ | Ultimate allowable strain the direction of axis 1 |
| $F$ | Element flexural matrix |
| $G_{12}$ | Shear modulus in axis system 1-2 |
| $\gamma 12_{ua}$ | Ultimate allowable shear strain in the plane 1-2 |
| $K$ | Element stiffness matrix |
| $L_A$ | Applied load for constraint $j$ |
| $L_D$ | Unit load in direction of displacement to be measured for constraint $j$ |
| $L(\lambda, \mathbf{X})$ | The Lagrangian used to derive the optimality criterion |
| $m_i$ | Mass of element $i$ |
| $m$ | Number of variables in the solution |
| $M$ | Total mass of the structure |
| $M_v$ | Total mass of those areas of the structure which are variable |
| $N_X$ | Panel end load in $x$ axis |
| $N_Y$ | Panel end load in $y$ axis |
| $N_{XY}$ | Panel shear load |
| $n$ | Number of inequality constraints |

| | |
|---|---|
| $p$ | Number of equality constraints |
| $\rho$ | Density |
| $S$ | Scale factor for a panel thickness |
| $t_{ply}$ | Thickness of a given laminate ply |
| $\nu_{12}$ | Poissons ratio in axis system 1-2 |
| $W1$ | Scale factor for the width of an element |
| $W2$ | Scale factor for the length of an element |
| $X_i$ | Matrix of variables which define solution $i$ |

# Abbreviations and Acronyms

| | |
|---|---|
| AFS | AirFrame Sizer - a library of structural sizing methods developed during this research work |
| ADG | Analysis Development Group within Structural Computing |
| CFC | Carbon Fibre Composite |
| CITS | Computer Integrated Technical Standards - a proprietary code which includes a composite panel stability analysis method |
| COTS | Commercial Off The Shelf |
| COM | 'Component Object Model' programming interface |
| COMBEL | A group of 'Combined Elements' within ECLIPSE |
| DMAP | Direct Matrix Abstraction Program (DMAP) - programming language which defines the solution sequences used in NASTRAN |
| DOT | 'Design Optimisation Tools' - a library of gradient based optimisation methods |
| ESO | Evolutionary Structural Optimisation |
| FBD | Free Body Diagram |
| FSS | 'Fixed Stacking Sequence' - Laminate Stacking Sequence Formulation |
| IPT | Integrated Project Team |
| LAS | Laminate Assembly Sizer - a laminate panel sizing method developed as part of this research work |
| MDO | Multidisciplinary Design and Optimisation |
| MFD | 'Method of Feasible Directions' search method |
| PAG | Production Analysis Group within Structural Computing |
| PSE | Problem Solving Environment |
| SLP | 'Sequential Linear Programming' search method |
| SOL200 | NASTRAN solution sequence which uses DOT to optimise a response for a given set of criteria |
| SQP | 'Sequential Quadratic Programming' search method |
| SRM | Stress Ratio Method |
| UAV | Uninhabited Air Vehicle |
| UCAV | Uninhabited Combat Air Vehicle |
| USAF | United States Air Force |

| | |
|---|---|
| VSS | Variable Stacking Sequence - Laminate Stacking Sequence Formulation |

## Acknowledgements

The author would like to thank his supervisors, Prof. Andy Keane, Alan Gould and Brian Oldfield. Andy helped me to understand this work both in the wider context, as well as the environment in which it was carried out. Alan provided me not only with an industrial perspective on much of this work, but also many of the contacts within the company necessary to carry it out. Brian provided me with a ready, and valuable, source of theoretical and practical experience of the particular case study in this thesis. In addition David Standingford and Paul Robinson who have provided me with support from within BAE Systems as part of the University Technology Partnership for Design.

I would also like to thank the following people: András Sóbester, Tony Scurr and Alex Forrester for making the CEDC a great place to work and providing readilly available sources of knowledge; Carren Holden and Stephen Leary for their help and advice on elements of this work; Coursemates Marc Bevan, Grant Mackinnon and Olly Bewes for the mutual support and use of their sofabed; Chris Bingham, Dave Pearson, Mark Conlin, John Ayres, Richard Howard, Nigel Warr, Rosemary Reid and Richard Tookey for their technical help and advice during my time at Warton; Sarah, Roger, Greg, Kelly, Suzy, Darran, Mark, Lisa and Helen - friends I've made whilst on placement with the company and who showed me around Preston and the North-West.

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*To my parents*

# Chapter 1

# Introduction

> The prosperity of countries depends on their ability to *create value* through their people, and not by husbanding resources and technologies.
>
> ...
>
> Under Cold War assumptions, government officials fell back on arguments that countries have to be prepared for emergencies, - that is war. Inefficient industries are subsidised in the name of national security.
>
> *Kenichi Ohmae, The Borderless World, 1991 (Ohmae (1991))*

The end of the Cold War has changed the military aircraft market. A market that was driven by the deployment of the most advanced technology, as quickly as possible, and often at a high degree of risk, has become one where the emphasis is on 'better, faster, cheaper' upgrades to existing systems, and on reducing the risks associated with new technology through more early concept and evaluation work (Walmsley (1999)). The effect on the structure of the industry has been clearly visible. There has been an increased consolidation of major aerospace firms into a number of major defence companies, with an emphasis on reducing both the direct product costs and fixed overheads of the businesses (Crute et al. (2003)). What is not clear is how, or indeed if, the technology and processes these companies use have adapted to meet these new priorities. How can manufacturers design airframes 'better, cheaper and faster'? How can they reduce the 'risk' they face with complex, often highly customised, products?

Major defence customers, such as the UK Ministry of Defence, have sought to reduce the tendency of uncertainties in projects to cause cost overruns and late delivery of projects, commonly known as the project 'risk' (Chapman and Ward (2003)). Rather than use a 'cost-plus' purchasing scheme, where the customer takes on the risk, they have moved towards 'fixed-price' contracts, where the company takes on the risk, and more recently 'smart-procurement' style initiatives, where the risk is shared, and to an extent traded, between customer and contractor (Walmsley (1999) and Crute et al.

(2003)). A core competency for competitive defence companies now needs to be their ability to manage risk. They need to understand the sources of uncertainty and how to manage their potentially positive and negative consequences on the project outcome (Chapman and Ward (2003), Crute et al. (2003), Tidd et al. (1997)). One of the main challenges they face in managing this risk is illustrated in Figure 1.1, which shows the typical engineering design paradox when developing a new product. This paradox is particularly true for military aircraft manufacturers since they develop highly complex, highly customised products (Sobieszczanski-Sobieski (1999), Crute et al. (2003)). At the start of the project the amount of knowledge about the product design is very small, but builds slowly through the concept and early development stages, until in the late development and early detailed design stages there is a large amount of product knowledge. Conversely the ability to change the design, or 'design freedom', typically decreases as the knowledge of the product increases. This is because assumptions and decisions are made which tend to 'fix' the design and effectively 'impart an inertia' to the design process (Sobieszczanski-Sobieski (1999)). The effect of fixing these decisions is to 'build-in' the costs of the product, with approximately 80% of the whole-life costs of a product built-in at the design stage (Crute et al. (2003)).

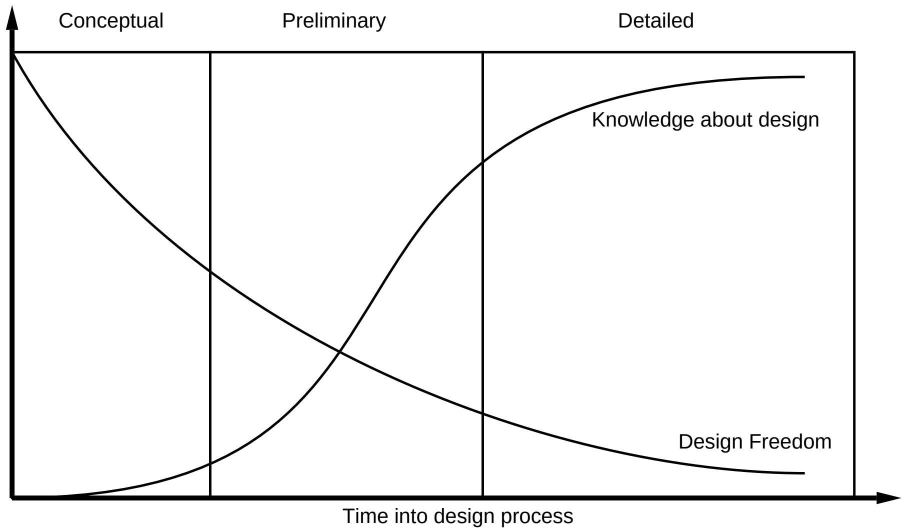



FIGURE 1.1: Knowledge of product during lifecycle of typical products (after Sobieszczanski-Sobieski (1999))

This can create dilemmas in the early stages of the project, since decisions have to be made when there are the highest degrees of uncertainty in the design (Chapman and Ward (2003) and Johnson and Scholes (2002)). Uncertainty can be present in the customer's requirements, which can change over the development time of the product, in the company's ability to understand and implement new technology, and in the ability of the company to meet the design criteria first time, minimising redesign. Examples of these issues are given in the development histories of many recent aircraft (Jackson (2005)). The Eurofighter Typhoon customer was often uncertain as to the exact role the aircraft would be used for, resulting in political decisions delaying development and production. In the case of the Bell V-22 Osprey the use of innovative propulsion technology meant engineers were not able to fully predict the behaviour of the vertical takeoff and landing system. A crash due to unpredicted handling properties in descent mode meant significant further analysis and development work. Finally, in the case of the F-22 'Raptor', substantial redesign work was required to produce a tail-fin design that reduced problems experienced with buffet, thereby adding to program delays and costs (Warwick (2003)).

Economists such as Thurow argue that companies can find competitive advantage through the tools they use to make the product (their technology), the methods they use to make the product (their processes) and the ability to understand and use them to their best advantage (through the employees and their skills) (Thurow (1996)). This is illustrated in Figure 1.2. Based on this premise he argues two points. First, that companies should not simply rely on processes that generate innovative products to sustain a competitive position. Using the example of consumer electronics such as the video camera, recorder and CD player, he argues that the inventors (the Americans and the Dutch respectively), have achieved less out of these inventions than the Japanese in terms of sales, employment and profits, despite the Japanese not having invented them. He argues their success is mainly due to their ability to produce these goods at a lower price using more efficient design and manufacturing processes, arguing "Technology has never been more important, but what matters more is being the leader in *process* technologies and what matters less is being the leader in new *product* technologies" (Thurow (1996)). He has also cited two different strategies to innovation, noting that traditionally two-thirds of research and development spending in America was spent on product innovation, whereas in Japan two-thirds was spent on process innovation (Thurow (2003)). Since it is argued that the customer's priorities have moved away from highly innovative products (Walmsley (1999)), this implies that military aerospace companies need to change their product strategy to look towards producing products more effectively, rather than just more effective products.

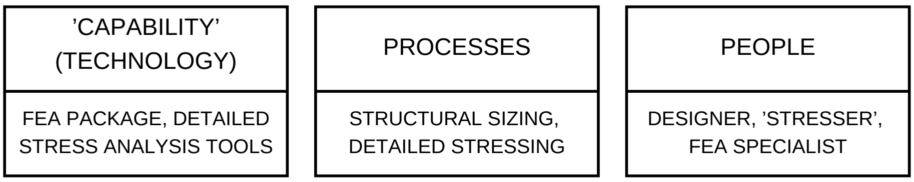



FIGURE 1.2: Three sources of competitive advantage (Thurow (1996))

Second, he argues that technology and processes are transferable and that ultimately these will be available to anyone that can afford them. Within the structural discipline this can be seen with tools such as NASTRAN. Developed as a proprietary finite element analysis (FEA) code for NASA it was subsequently licensed for external development and sale. It is now an industry standard FEA code which contains developments made as a result of its availability within the aerospace community. Thurow argues that as technology and processes proliferate and become standardised then the only real source of competitive advantage is an employee's ability to produce a better product using them than their competitors. Thus, their understanding and ability to use these processes (their 'skills') would need to be better than their rivals producing competing products.

Although it is clear that there have been significant changes to aerospace design and manufacturing organisations, it is not clear if the technology and the processes they use have changed to meet the needs of this new marketplace. One particular area of interest, and that studied in this thesis, is that of airframe structural sizing, one of a few key processes that directly affect the whole design, and hence performance, of future airframes. The structural mass of an aircraft is typically 30% of its gross weight at take-off (Niu (2001)) and determines the loads and environmental conditions which the aircraft can experience. Higher loads and harsher environments typically require a heavier structure. However heavier structures are typically more expensive than lighter structures because of the additional design, manufacturing and material costs associated with larger designs. A lighter structure could allow more fuel to be carried, thereby increasing the vehicle range; more payload to be carried, increasing its effectiveness; improved acceleration and manoeuvrability through reduced inertia (Niu (2001)).

The relationship structural mass has with airframe cost and performance means that minimising it is usually the surrogate objective in the trade-off of airframe cost and performance with the loading environment it can withstand. This trade is known as 'structural sizing'. However, it should be remembered that the mass function does not accurately represent either the airframe's ultimate cost or performance. Vehicle mass can often be reduced by developing more complex designs. However, increasing the complexity of the design will normally increase the cost of design and manufacture. Similarly a focus on minimising the mass of the design may create one which is less robust to changes in loading. The lightest mass design might work well for the loadcases studied but not perform well under similar magnitude, differently distributed, 'off-design', loads.

Current design processes involve a significant amount of user involvement to analyse a given design against different design criteria for each discipline. This means that the iterations through a number of different designs needed during product development result in a long development time (between 5-10 years for aircraft such as the A380, Eurofighter Typhoon and the Joint Strike Fighter (Jackson (2005))). The resources required over that period are an overhead which must be factored into the cost of each product. Hence, there is an implicit trade-off between the value found through the design process (how well the design that meets the expectations of the customer) and its cost.

## 1.1 Vision of the Ideal Structural Design Process

The ideal design process finds the design which the customer perceives as having the highest 'value', that can be sold at a profit, using the minimum amount of company resources in the minimum amount of time. This could be achieved through gaining more product knowledge in the early design stages, reducing uncertainty through: (i) an increased understanding of how the product would function; (ii) a better understanding of the customer needs; (iii) an increased understanding of the technology. This would be achieved through a more rapid analysis and design capability that would also allow decisions to be made later in the project, reducing the likelihood of corrective actions required to successfully complete the project.

For 'stakeholders' in the process the measures of an 'ideal' process are somewhat different. Table 1.1 shows that the company's customer wants the system to produce a low-cost, high value airframe that performs as expected. The airframe will have been designed by a project engineer who wanted to find the process easy to use, understood the result it produced and was confident that it was the best possible. For their manager the process was low cost to use, integrated into their way of working, and produced a product that met their requirements (perhaps a high value airframe that could be sold at a premium). Managing and maintaining the process are a process manager and engineer respectively. The manager is confident the process rapidly produces designs competitive with the rest of the industry sector and has a low cost of ownership. The process engineer finds the process easy to maintain and upgrade, and the skills learnt in doing this help him in his career. Suppliers to the airframe manufacturer are confident that the system has a long term future and are keen to provide value-added services to support it. The shareholders want their investment to generate value for them, which usually manifests itself in dividends paid from profits or a higher share price.

Increasingly shareholders and company directors perceive a need for the company to only perform those activities where it can most effectively add value to its products. An external organisation might be capable of more efficiently managing a company's office accommodation, an activity which adds little value to the company's products; it might more efficiently manufacture major components, an activity which adds significant value to the product, but which allows the final product to be sold at a reduced, more competitive, price in the marketplace. Various accountancy ratios are used as surrogates to gauge the efficiency of a company at utilising its assets to create value (Pizzey (1998)). In particular ratios of sales and turnover to a company's assets are used to gauge the efficiency of a company. Thus, there is often a drive to outsource activities and reduce the assets a company owns to increase its effectiveness. Common issues that arise from outsourcing in the engineering environment are discussed in Section 2.4.

| Stakeholder | Measure of success |
|---|---|
| Customer | High value product for minimum cost |
| Project Manager | High value product for low cost of design process operation |
| Project Engineer | Ease of use and performance of process |
| Process Manager | High performance and low cost of ownership (overheads) |
| Process Engineer | Easy to maintain and upgrade process |
| Component suppliers | Sustainable system that has a long term future and is possible to provide value-added services for |
| Shareholders | A competitive, profitable, company capable of generating value for shareholders |

Table 1.1: Measures of a successful process for stakeholders

## 1.2 Research Aims

The Structural Computing group within Air Systems wishes to

1. understand the issues that an aerospace manufacturer will face in sustaining a competitive airframe structural sizing process given the changing engineering environment;
2. develop and demonstrate untried elements of this technology within the engineering environment;
3. recommend a suitable process for future structural sizing based on the results of this work;

It is within this context that the research work was carried out. Specific technical aims and motivations for this research were to understand how to develop a structural design process that

1. utilises the existing toolset to reduce the fixed and variable costs of the sizing process;
2. is automated to reduce the overhead of manual intervention in the sizing process to allow more rapid analysis and design;
3. uses a free-body-diagram approach to enable interfacing with in-house and commercial-off-the-shelf (COTS) codes for a global-local sizing approach, opening up the range of suppliers and sizing methods available to the company;
4. will integrate within a future multidisciplinary design optimisation environment within the company using the same techniques to reduce the support overhead on the sizing process;

## 1.3 BAE SYSTEMS and 'ECLIPSE'

BAE Systems is a prime contractor and systems integrator in air, land, sea and space based engineering projects. In 2003 it employed nearly 100,000 employees worldwide. One of the 'Programmes' within the company is 'Air Systems' which is responsible for the design, build and test stages of new aircraft, which currently include the Eurofighter Typhoon, Nimrod MRA4, Hawk and Joint Strike Fighter products. In 2003 11,000 people were employed by this part of the business (SYSTEMS (2003)). Recently Air Systems has been undergoing significant organisational changes to reach its 'vision' of a "right sized, profitable, company".

The motivation within Air Systems for this piece of work was to understand how its internal sizing capability and process will need to change in future. 'ECLIPSE' is a proprietary sizing system that has been in use in various forms since the early 1970's (Thompson (1999)). It is an effective tool that has been used to size aircraft including the Eurofighter Typhoon, EAP demonstrator and the Saab Gripen wing amongst others. At the outset of this work in October 2003, ECLIPSE maintenance and development was managed and carried out by the 'Structural Computing' group within Air Systems. Within Structural Computing the Analysis and Development Group (ADG) were responsible for maintaining and developing the code, whilst the Production Analysis Group (PAG) supervised its use on different projects. During the period that this research work was carried out the ADG has become part of the 'Technical Computing' discipline, responsible for maintaining and developing proprietary company capability. The PAG group has become part of the 'Airframe Integration' team in the company's 'Engineering Investment' division and is still responsible for the use of ECLIPSE. For the purposes of this work the original ADG and PAG titles are used.

The processes used by a company also have a lifecycle, which covers their initial usage through to their replacement by a more competitive process. Figure 1.3 shows a generic product lifecycle which has parallels to the lifecycle of current generation of proprietary sizers. In the 1970's sizing technology was highly proprietary, with only large engineering organisations having the computational resources and expertise in structural analysis methods to develop these applications. Duysinux and Fleury's (Duysinux and Fluery (1993)) review of optimisation software in use in 1993 shows that Dassault-Breguet, RAE/Duetsche-Airbus (now QinetiQ/Airbus) had also developed systems around the same time (the 'Development' phase). The current generation of systems increased in subsequent years, with Saab, the then Dornier and MBB (now EADS) developing systems in the 1980's (effectively the 'Growth' phase). Activities such as the GARTUER programme in the 1990's (ap C. Harris (1997b)) aimed to compare these systems and share knowledge about relative performance (the 'Shakeout' phase). In addition, sizing systems have become commercially available, with GENESIS (Inc (2004)), Optistruct (Engineering (2004)) and Hypersizer (Collier Research Corporation (1997)) capable of much, but not all, of the functionality of these systems (the 'Maturity' phase). Each phase has benefits and disadvantages. The earlier in the lifecycle a new process is used the more knowledge a company has in its use compared to its competitors. However, the earlier a process is adopted the greater the uncertainty surrounding the process, and hence 'risk' that the process will have to be changed in favour of another.

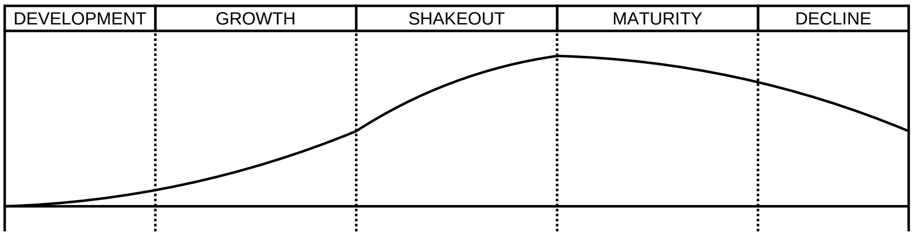



FIGURE 1.3: Lifecycle of a product in marketplace (after Johnson and Scholes (2002))

In a description of ECLIPSE as it was in 1999 Thompson (Thompson et al. (1999)) highlighted a number of problems that faced ECLIPSE and indeed similar systems. First, the system was developed using a large amount of previously written code which made assumptions no longer valid and therefore hampered further development of the system. Although written in FORTRAN there was a significant amount of platform specific code that make it a non-trivial task to port it to other platforms, although this had been realised a number of times since its initial development. The system also relied on old NASTRAN solution sequences which were increasingly becoming unsupported as well as limiting access to enhanced modules within NASTRAN. Finally the system relies on select individuals within the company for support as well as the availability of the last platform to which it was ported. This remained the situation during the period this work was carried out.

Since capabilities such as ECLIPSE are typically not seen as a core product then they often remain proprietary. Since the capability is usually maintained on an ad-hoc basis, as previously described, then it is often developed in a highly focussed manner. Many outside observers might see the result as a 'monolithic' system. An internal observer, in particular the developer, would usually see the system as a collection of methods and routines.

## 1.4 Overview of Research Work

The EngD is a four year fixed term course. In this case the first two years were spent undertaking technical and management courses together with research into the Application of Design of Experiment Techniques within BAE Systems. This work formed the basis of a previous mini-thesis. The second two years were based on placement at BAE Systems (Warton), near Preston, from October 2003 to October 2005. The project work undertaken as part of this placement was as follows:

Work over the two year placement was planned around a broad set of research as shown in Figure 1.4. The first project was an investigation into the use of ECLIPSE within a basic Evolutionary Structural Optimisation (ESO) process. This also acted as a familiarisation exercise for much of the company's toolset. The second project was an investigation into the use of the company's laminate panel analysis code to size wing skin panels for strength and stability criteria. These two projects provided an insight into the third, an investigation into the development of a structural sizing process using the company's toolset.

The emphasis on the research work has changed over the period of these two years. Initially there was a strong emphasis on the development of capability that could be developed as part of in-house software. However, with the changes within 'Air Systems' to become a more profitable 'right-sized' company this has become less so. Thus, the tendancy of the 'Structural Computing' group is turning towards considering options for more competitive sizing processes rather than in enhancing capabilities.

Results of this work have been fed back into the company through detailed reports on the laminate sizing methodology and the initial results of the structural sizing process. Investigations into the ECLIPSE stiffness sizing methodology identified and rectified some of the issues associated with updating ECLIPSE for use with later versions of NASTRAN. Together with other members of the ADG/PAG groups, and the FE code suppliers, ECLIPSE has been updated for use with more recent versions of NASTRAN - a problem which had been unresolved in recent years. A COTS code, Hypersizer, was tested and demonstrated to the Production Analysis and Analysis Development groups. Sizing was carried out on an example structure for an external customer, and on geometry provided as part of the FLAVIIR programme (Flaviir (2005)).

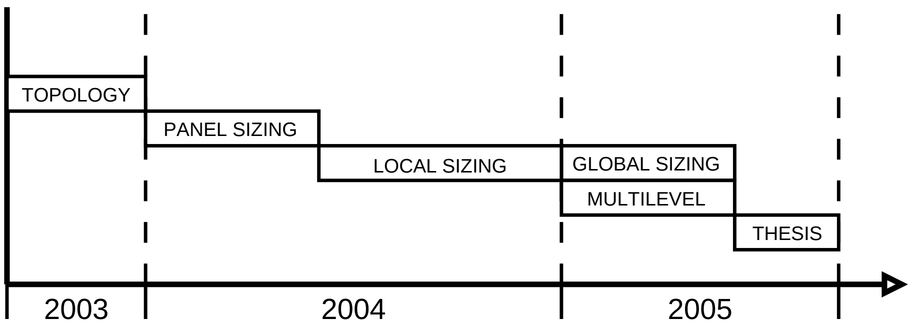



Figure 1.4: Overview of EngD placement at BAE Systems

The author would like to acknowledge the contribution to this work made by the following people given in Table 1.2.

## 1.5 Overview of Thesis

The central argument of this thesis is that companies should move away from their historical focus on a superior sizing toolset. Instead they need to focus their attention on developing processes that allow engineers to rapidly increase their knowledge in the design at the earliest stages. In particular the process used to structurally size aircraft needs to move away from the proprietary, monolithic, codes which have historically been used by all major airframe design organisations. Instead they should adopt a sizing process which first allows them to adapt the sizing toolset to the needs of the product, but more importantly allows them to rapidly size and design new airframes.

Chapter 2 looks at the current structural design process, the environment in which it operates and some of the changes that are likely to happen to it. Chapter 3 reviews current structural sizing methods used to size components and the airframe. Chapter 4 looks at how existing, proprietary, tools can be used in structural sizing processes. A multilevel approach is proposed as an alternative to the current monolithic sizing systems currently in use. Chapter 5 studies the use of free-body-diagrams of metallic panels to size a structure at a local panel level for strength and stability criteria. Chapter 6 extends the method to include a sizing method for global stiffness criteria adapted from the existing ECLIPSE methodology. An example structure is then sized for strength, stability and stiffness criteria using both the local and global sizing methods. This work is discussed in Chapter 7 and conclusions drawn in Chapter 8.

| | |
|---|---|
| **CITS code including ModelCenter Scriptwrapper and COM wrapper** | Mark Conlin, Richard Howard, John Ayres<br>Analysis Development Group<br>(Structural Computing)<br>BAE Systems |
| **NASTRAN data access routines used in ESO/ECLIPSE** | Mark Conlin<br>Analysis Development Group<br>(Structural Computing)<br>BAE Systems |
| **AFS COM wrapper** | Richard Howard<br>Analysis Development Group<br>(Structural Computing)<br>BAE Systems |
| **UAV CAD geometry and CFD data** | András Sóbester,<br>Computational Engineering &<br>Design Research Group<br>University of Southampton |

Table 1.2: External contributions to this work

# Chapter 2

# Current & Future Engineering Environment

Increasingly current structural design processes are carried out: within organisations that are changing structure; using technology and processes adopted from commercial vendors rather than developed in house; and to produce subtly different types of products. The chapter looks at the current and future structural design environment in which engineers operate in a 'typical' aerospace manufacturer. In particular it considers the factors which are likely to influence the choice of structural sizing process within an aerospace manufacturer. Section 2.1 looks at the application of structural optimisation technology throughout the product lifecycle. Section 2.2 considers changes likely to affect the military airframe design market. Section 2.3 looks at the future requirements for products in the aerospace sector, and in particular the military aircraft market. Section 2.4 looks at the toolset that is available to engineers and how it is likely to change.

## 2.1 Engineering Lifecycle

It is argued within the airframe sizing and general engineering communities that "we know far more about how to create powerful new tools than we know how to design, deploy, use and regulate them" (Rheingold (2002)). In particular Vanderplaats argues that "...the state of the art is now reasonably well refined. The challenge is to assimilate this technology into the practising design environment" (Vanderplaats (1999b)). He further argued that structural optimisation was capable of reducing development time in addition to improving the product quality. His argument concluded with the statement "It is time to move aggressively to get structural optimisation out of the research department and into the design environment". Since at the time of writing it is six years since this argument was made it is instructive to see how structural optimisation technology is being used within a 'typical' aerospace organisation and what the barriers, if any, have been to reducing product development time and improving the product quality.

This section provides a 'snapshot' of the use of structural optimisation methods across the airframe design lifecycle at BAE Systems as viewed during a placement in Summer 2002. The development process passes through a defined set of phases in a proprietary lifecycle management process. The basic phases are common to any manufacturer and are given in Figure 2.1, of which stages 1-4 ('Conceptual' to 'In-Service') are studied here. Figure 2.2 shows how the process was carried out for the Eurofighter Typhoon.

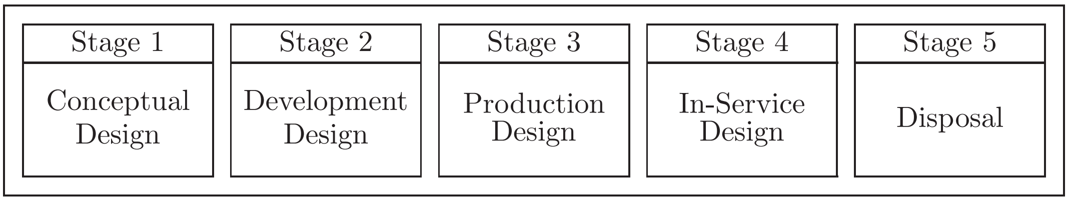



FIGURE 2.1: Simplified version of the aerospace product lifecycle

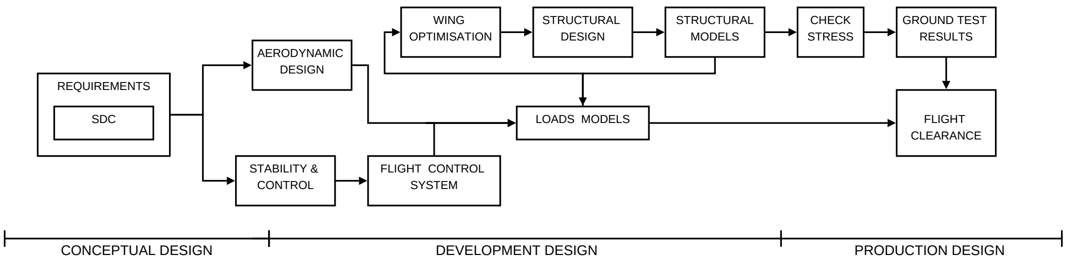



FIGURE 2.2: Air vehicle design process used for Eurofighter Typhoon (after Jimenez-Garzon (1996))

### 2.1.1 Conceptual Design

'Conceptual' design is essentially a feasibility study for the customer and manufacturer to understand the viability of the product. Engineers attempt to produce a design to meet the needs of the customer based on the results of an operational analysis and a set of customer requirements (Friemer (1996)). This process is regarded as 'organic' and can take different forms.

At this stage the engineers find an 'optimal' design through discussion with customers and engineers rather than a systematic process. In part this is because the exact criteria against which designs are judged are not clear enough to be expressed. However, two methods are used to understand likely final designs. The first is the search for a baseline design against the likely design criteria. The second is the production of design trade-offs and sensitivities that indicate potential trades that could be made.

The baseline design is based on the results of an aircraft performance optimisation using the company's proprietary Computer-Aided Project Studies (CAPS) code. This uses basic empirical performance models and statistical data from other aircraft to find design geometry that is most likely to meet performance targets such as minimum range, cruise speed, and turn rate (see Figure 2.3). In this sizing the structural design is considered only to affect the performance and is found using statistical models of the masses of similar aircraft and their performance. Where data for similar aircraft is not available the statistical models may need to be augmented with data from more detailed studies of novel concepts. Scaling of the design is performed using Newton, Fibonacci, Powell Hybrid, Muller (Press et al. (1998)) and Multivariate Optimisation (MVO) techniques (Chacksfield (1997)). Variations of the baseline design are studied to understand the design trade-offs that could be made to improve such issues as performance or likely cost. Examples of trade and detailed design studies are given for the Eurofighter Typhoon in Tables 2.1 and 2.2. Tools such as ECLIPSE are sometimes used as to determine a more detailed mass prediction of these designs, especially for novel configurations.

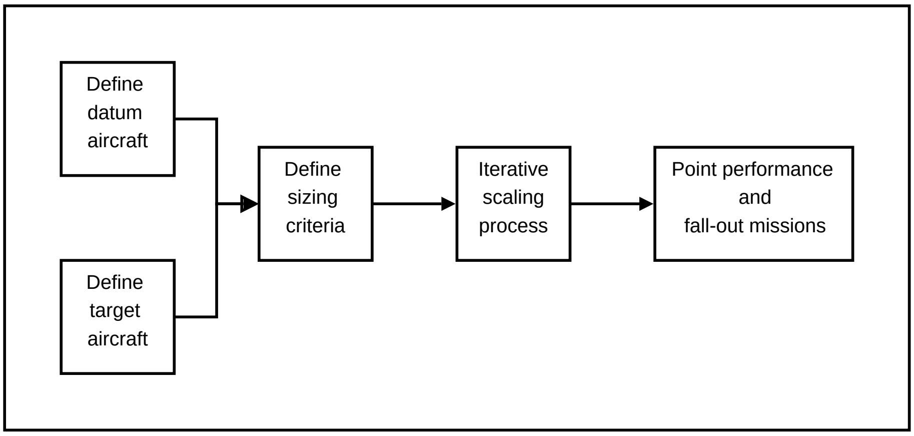



FIGURE 2.3: Overview of the conceptual design optimisation process

| **Variable** | **Possible setting** |
|---|---|
| Wing Planform | Straight Leading Edge / Cranked Leading Edge |
| Engine Intake | Curved / Rectangular |
| Engine Nozzle | Convergent-divergent / Convergent |
| Airbrake Location | Fuselage spine / Underwing / Rear Fuselage / Tip-pod mounted |
| Radome Shape | Elliptical / Circular |

TABLE 2.1: Concept trade-studies conducted for the Eurofighter Typhoon (Friemer (1996))

| **Description** | **Variables** |
|---|---|
| Wing | Wing-body setting, camber and engine/afterbody cant for optimum zero pitching moment |
| Wing flaps | Geometry, size and spanwise location |
| Engine | Engine cycle trade-off |

TABLE 2.2: Detailed optimisation studies conducted for Eurofighter Typhoon (Friemer (1996))

The trends observed in Figure 1.1 manifested themselves in two comments made by engineers at this design stage. First, the results of the conceptual design stage are generally not challenged further along the design lifecycle because the expense of re-examining previous assumptions usually exceeds the benefit of doing so. Later stages of the process generally rely on the validity of these decisions. Secondly these studies were regarded as "extremely tedious" because the highly integrated and complex design meant that changes in one feature would impact many other areas. It was noted that it required a lot of experience to distinguish the important from the unimportant and to steer efficiently within time and cost constraints to the best solution (Friemer (1996)).

### 2.1.2 Development Design

The purpose of development design is to produce a detailed design which can be built, tested and in later stages productionised. At this stage the structural design process is typically a mix of manual trade-studies and semi-automated sizings at varying levels of detail as shown in Figure 2.4. The objective function is usually mass, used as an indicator of likely cost and performance.

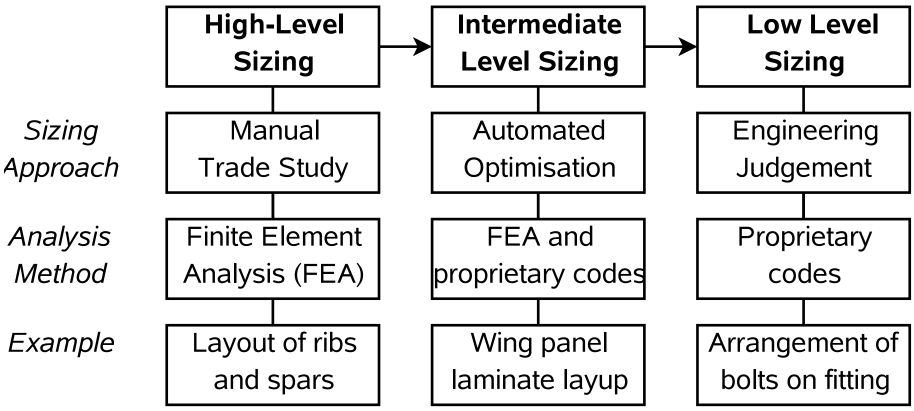



FIGURE 2.4: Multilevel sizing process

At the 'top-level' structural analysts investigate how high level variables, such as the number of ribs and spars in a wing, can be varied to meet criteria such as mass, performance and manufacturability. This would typically be carried out using manual trade studies, varying the number of ribs and spars in a FE model by hand. Research work has been conducted at a top-level looking at the variation of geometry to minimise acoustic loads in a weapons bay (Moretti et al. (1999)) and involvement in the European 'MOB' project (Morris (2001)), which included a study looking at the effect of these variables on the sizing of structure at lower levels (Engels et al. (1999)). Current projects do not use an automated approach to study these top-level variables.

As part of the top-level sizing the likely mass of the resultant structure is usually predicted by a 'medium-level' sizing tool, in this case, ECLIPSE. This involves looking in more detail at how the material is distributed throughout the structure to minimise the mass whilst meeting strength, stability and aeroelastic requirements (Jimenez-Garzon (1996), Thompson (1999)). At this stage the design complexity increases from O(100) variables to up to O(100k) variables with the use of the properties of individual finite elements as variables. This is discussed in more detail in Section 3.5. Once the design has matured sufficiently that both the customer and the company is confident in it then selected loadcases will be used together with ECLIPSE to size the final design configuration. Figure 2.2 shows this iterative process.

The 'medium-level analysis provides details about the location and amount of material, but does not normally include details such as fittings and component joints. Much of this detailed, 'low-level', analysis is left for subsequent stages of the life-cycle. However, for a prototype design and detailed design studies, the engineer will wish to look at specific areas on which assumptions have been made. BAE has a suite of commonly used stress calculations known as the Computer Integrated Technical Standards (CITS) package. These model a number of standard airframe structural problems including panel strength and stability, pin-jointed frameworks, beam sections and bolt-groups. There is no built-in optimisation capability in CITS, so these design problems are solved using an engineer's experience and judgement.

### 2.1.3 Production Design

Production design is the stage at which the detailed design is turned into a design suitable for manufacture. Assumptions made at previous stages of the design process are generally not challenged unless production work indicates a problem. Formal optimisation methods are generally not used at this stage. Instead the final design is found through a formal lifecycle consisting of repeated detailed design and review iterations. The designs are analysed by each discipline and the results used to manually create the next design iteration with the aim of minimising the aircraft mass. The number of iterations performed depends on the time available to the project.

Two types of optimisation tool are occasionally used at this stage. The first is 'built-in' optimisation methods within existing toolset, which includes ProMechanica, ViconOpt and NASTRAN. It is infrequently used because engineers find it difficult to parameterise the problem sufficiently well in the toolset. Ideally engineers would like to parameterise the problem using all the variables that might influence the responses being considered.

An example is the installation of an avionics package which needs to be isolated from significant vibration of a given frequency. Parameters can include the design of the mounting brackets; the type of vibration mountings used; the equipment location and orientation. This example might need a large number of both discrete and continuous variables to describe the problem well. However, the current range of built-in optimisers within FE packages typically offer limited control of model properties (e.g. panel thickness or shape), using relatively immature user interfaces. This can make the parametrisation and post-processing difficult or unwieldy for large problems. Moreover, it can be very difficult to represent discrete parameters such as examining different combinations of components. Much of this is due to the built-in optimisers which were not designed to solve highly complex problems.

Secondary reasons for not using existing optimisation tools include a perception that the process will take at least as long as a manual search. This is because the overhead in initially defining and parameterising the model is seen as too great. Second, there is a lack of confidence that it will find a better answer than the manual process. Finally a lack of training and familiarity with this part of the toolset means engineers do not feel confident enough to use it or have enough awareness of how it could be useful.

There are occasions where a proprietary laminate layup code is used to design laminate layups for manufacture. The code searches for a ply-layup design that meets ply-blending, damage tolerance and ply-blocking criteria for a structure defined using a number of layup 'areas'. The search is performed by randomly selecting layups and testing to see if they are feasible. It might be possible to improve this approach by adopting techniques from panel sizing techniques discussed later in Section 3.2.2.

### 2.1.4 In-Service Design

In-service design deals with needs that arise during an aircraft's operational lifetime. If a particular component requires redesigning then it will be addressed at this stage. It is unlikely that the airframe will be significantly redesigned at this stage and as such top-level design variables will remain fixed. However it can require changes to very low-level design variables (e.g. thickness of material around a lug hole). It is still not common practice to use optimisation techniques for such problems, mainly because of the limited availability of experience and design time. However, there are cases where optimisation techniques would have been good candidates for solving the problem, saving design time and potentially creating a more robust solution. One such example where optimisation may have improved the results from a design problem is in an in-service cockpit vibration problem. Equipment was removed from a series of aircraft cockpits as part of an upgrade programme. The removal of the equipment lead to pilots reporting excess vibration of a panel during flight and a solution was needed. The in-service engineers approached the problem using a 'trial and error' approach with hand calculations and experiments. This required a number of iterations, which it was felt could have been reduced had a more systematic optimisation method been used. The reason given for not using optimisation techniques was that it was felt that there was too much uncertainty around the likely success of the optimisation process.

### 2.1.5 Ownership of processes

Within aerospace manufacturers one of the main organisational changes in the last decade has been the adoption of 'Integrated Project Teams' (IPTs) that are responsible for a particular product, or family of products. Rather than divide the company along discipline boundaries such as 'engineering' and 'procurement', these teams include elements of every discipline necessary for the project. The aim is to increase the focus on the products rather than the technology within those products (Crute et al. (2003), McMasters and Cummings (2002)). Many of these IPTs are formed from a partnership between large aerospace manufacturers (Esposito (2004)) which means that there is a certain amount of diversity in the toolsets and processes used within a given company. For example, Project A might use Sizer A because there is a ready available source of expertise within the company, but Project B might use Sizer B because it needs to use the collaborative toolset agreed with partner companies.

The emphasis on the product implies reducing the product cost, leading projects to 'shop' around for capability. There is generally a core department at the host organisation advising on the process and capabilities of common tools. In the case of BAE Systems there was a 'Structural Computing' group consisting of an 'Analysis Development' group responsible for development of tools such as ECLIPSE and CITS, and a 'Production Analysis' group responsible for management of structural design processes and their support. Structural Computing offers proprietary tools such as ECLIPSE to projects but are 'paid' in 'man-hours' required to perform necessary work. This internal market differs to the external market where suppliers are paid in cash which includes a margin above the direct cost of performing the work. This margin allows the external company to re-invest in their products as they see necessary rather than as the project sees *absolutely* necessary. A further implication of the move towards IPTs has been that there is a large amount of process specific knowledge located in the projects which is difficult to share between other projects. This is partly due to restrictions on distribution of information about partners processes but also because of differences in the toolset and a reluctance to share information that was not paid for by another project.

One of the main difficulties in maintaining a capability such as ECLIPSE is justifying the funding necessary to maintain it in a competitive aerospace organisation. Although it is used in the concept, development and production design stages it would be most likely to attract maintenance funding from projects in the development phase since it will form the basis of their mass, and hence cost and performance, management of the product. Although it is used in the concept and production design stages this is often more tactical. Thus, it can be difficult to attract funding from projects that do not yet have a budget for further development, or perceive no need for that tool on their project. This means the long-term planning and maintenance of such tools can often be very difficult, as noted by Thompson (Thompson et al. (1999)).

### 2.1.6 Summary

From this study a number of clear barriers to the use of structural optimisation technology can be identified. First, there is a general lack of awareness of structural optimisation technology and potential benefits which inhibits users from even considering its use. Secondly the trend towards tactical investment in technology and processes has lead to a lack of a long-term plan for the use of this technology and a limited optimisation toolset. This in turn has lead to limited use of structural optimisation by company employees perpetuating the lack of awareness. Finally, limited sharing of knowledge within the structures discipline means that where it has been used its results have not been promulgated, again limiting awareness of the technology.

Chapter 1 argued that now, and in the future, aerospace manufacturers will need to make more advanced products more effectively. Thus structural optimisation techniques should not simply be seen as a method of finding better technical solutions to a problem. Rather they have the potential to find better technical solutions more consistently and in a reduced time. This work shows that optimisation technology is still perceived as tool to support 'strategic' design decisions making that will affect the long term performance of the product as shown in Figure 2.5. It is not widely used in more 'tactical' decisions about components, partly because engineers are not widely trained to use what existing capability there is, but mainly because the current toolset does not offer ready access to results. This imparts an 'inertia' to the lifecycle where decisions are taken on the basis of optimisation studies that take some time to report back, possibly prohibiting the examination of alternative designs later in the lifecycle. This supports the need for the development of more 'Rapid Analysis and Design' capability, not just to improve the early stages of the design process, but also to allow more responsive access at later stages of the lifecycle.

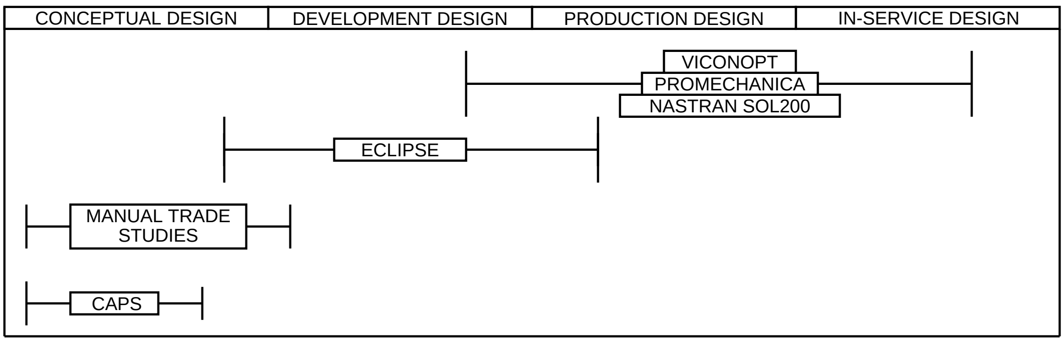



FIGURE 2.5: Summary of structural optimisation methods over the airframe lifecycle

Based this review the current, and likely future, structural design toolset is given in Table 2.3.

| **Level** | **Package** |
|---|---|
| *Local Features* | |
| Standardised structural calculations<br>Custom structural calculations | CITS<br>MathCAD |
| *Global - Local* | |
| Geometry creation<br>Finite element analysis | CATIA / PATRAN<br>NASTRAN |
| *Process Integration* | |
| Problem solving environment<br>FEA data handling | ModelCenter<br>In-house developed applications |

Table 2.3: Current toolset used by airframe structural analysts at BAE Systems

## 2.2 Military Airframe Design - Competitive Overview

Within Europe the existing competitors in the market include experienced aircraft manufacturers such as BAE Systems, EADS and Dassault, together with governmental backed organisations with airframe design activities such as QinetiQ (Kaynes (2004)). Similar competitors exist at an international level and all are competing for new airframe design work that is decreasing because of the increased lifetime of existing airframes (Esposito (2004)). However, perceived as high-technology industries, a number of governments and organisations in Asia are also looking to move into the airframe design market. Mitsubishi has airframe design and manufacturing partnerships with Lockheed-Martin to develop indigenous aircraft such as the Mitsubishi F-2 (Jackson (2005)). Studies show technology being transferred from the USA to Japan, and Mitsubishi developing a greater percentage of the products in-house (King and Nowack (2003)). Publications such as Shepard's Unmanned Vehicle Handbook (Shepards (2004)) show that there are many existing, operationally mature UAVs manufactured by experienced companies such as Israeli Aircraft Industries. These organisations could be seen as new entrants into the market, or as possible substitutes for existing manned aircraft. The Scaled Composites company is representative of a possible entrant, designing and manufacturing a number of high performance composite aircraft, including the Voyager non-stop round-the-world aircraft and the first sub-orbital civilian 'spacecraft', SpaceShipOne. Importantly they offer a full air vehicle design service, from conceptual design, aerodynamic and structural design, through to fabrication and flight testing (Scaled Composites (2005)). They use COTS FE software, including HyperSizer and NEi NASTRAN Inc. (2005).

Given that there is potential airframe design over-capacity, leading to increased competition within the airframe design marketplace, and that it is possible to structurally design aircraft, or indeed spacecraft, using COTS software, it would seem strange to develop proprietary in-house software where other organisations do without.

## 2.3 Requirements for a Structural Design Process

Ideally the structural design process should be determined by the types of product being designed, which in turn are determined by the company's strategy for its products within the marketplace. Figure 2.6 shows an abstract view of recognised 'successful' product strategies, plotting the perceived value of a product against its cost (Johnson and Scholes (2002)). These strategies trade the customer's desire for a product with their willingness to pay for it. It is possible to see examples of these strategies within the military aerospace market. An example of a focused differentiation product is the F/A-22 'Raptor' aircraft, widely regarded as the best available air superiority fighter (Jackson (2005)), and hence of high perceived added value, but also as the most expensive at \$133 million per aircraft (USAF (2005)). A trade-off between cost and capability can be seen with the F-16 fighter aircraft. This was developed as a lower cost day-only aircraft to complement the high capability all-weather F-15 aircraft, and as such exhibits a 'hybrid' strategy (Walker (1989)). The F-16A/B cost \$14.6million, whereas the F-15A/B cost \$27.9million (USAF (2005)). 'Low price' and 'No frills' strategies arguably can be seen in unmanned air vehicles such as the USAF 'Predator' UAV, which at \$4 million is around two orders of magnitude less expensive than the F-22 Raptor (Lewis (2003)).

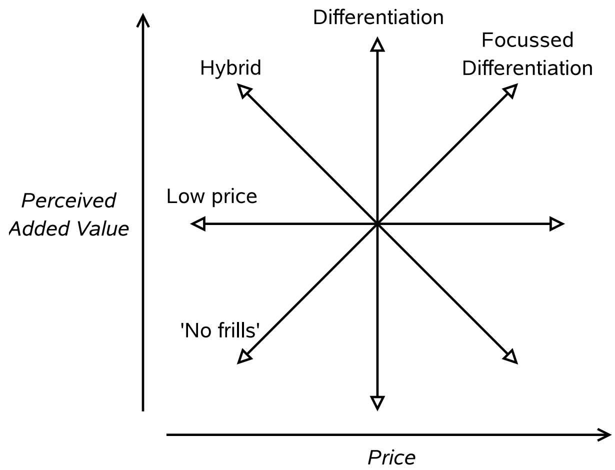



FIGURE 2.6: Competitive strategy options (after Johnson and Scholes (2002))

### 2.3.1 Generic Process Requirements

Whatever the product strategy shown in Figure 2.6 the most competitive companies will be the ones which move their products closer towards the top-left hand corner of the chart than their competitors. This implies processes that become increasingly effective, reducing the price of products whilst improving their perceived value. Crute et al review a number of terms used to describe design and manufacturing processes in the current era (Crute et al. (2003)). These include: 'mass customisation' - where products are manufactured in volume but with a recognition that the products need to reflect individual customers wishes; 'agile' - the ability of an organisation to switch frequently from one market-driven objective to another; 'lean production' - focusing on all forms of 'waste' within a system. For a company competing with a consistent product strategy 'Lean' practices would seem attractive since the processes are unlikely to change significantly between products and hence improvements in these processes would be cumulative. Within the automotive industry Lean techniques have been widely adopted by manufacturers and suppliers and are seen as necessary in order to compete in an industry that has a significant amount of over-capacity (Crute et al. (2003)). von Corswant and Fredriksson (von Corswant and Fredriksson (2002)) studied sourcing trends in the automotive industry and found that manufacturers and suppliers were placing increasing importance on improvements in key performance criteria. The most important criteria for manufacturers were 'quality' and product cost; suppliers saw, 'delivery precision' and 'product innovation' as being equally important as quality and cost which are similar to those argued for by Walmsley (1999). The difference between the automotive industry and the military / civil aircraft industry is in the volume and relative complexity of the products. Aircraft manufacturers typically produce highly complex, often highly customised products in volumes of 10's-100's of aircraft per year, whereas large-scale automotive companies produce relatively less complex, less customised, products in volumes of 100,000-1,000,000 vehicles per year. Comparisons between applying Lean practices in the automotive and aerospace sector tend to focus on manufacturing rather than the design of the product (e.g. Jina et al. (1997)). However, Boeing's implementation of 'Lean' engineering practices includes the goals of: improving the quality of the 'first pass' through the 'system'; organising teams that are fully accountable for their product; moving up the value-chain by focusing on core-competencies; reducing the company's cost-structure (Crute et al. (2003)). A similar set of successful 'lean' design rules was developed by Kelly Johnson, former Chief Engineer at Lockheed. He had fourteen operational rules within which the 'Skunk Works' advanced projects division would work (Rich and Janos (1994)). These included: delegating authority of a programme to a manager allowing them to make technical, financial and operational decisions; restricting the number of people on a project by using a small number of 'good' people; ensuring the customer timely funds projects; ensuring there is trust and very close cooperation between the customer and the contractor; rewarding good performance based on performance, not the number of people supervised.

The broad themes of these recommendations are: 1) delegation of power and accountability to those that need control; 2) close relationships with the customer to understand what they value and to ensure trust; 3) management and control of costs; 4) the effective use of good people.

### 2.3.2 Product Specific Requirements

Modern aircraft designs reflect standard structural design criteria which include static strength, fatigue life and tolerance to accidental damage (Niu (1999)). Interactions between the structure and other performance criteria mean that aerodynamic, hydraulic systems and observability issues are considered when setting criteria for the structure to meet. These then need to be considered in the sizing process, either through explicit sizing criteria such as flutter, and actuator loads, or through the implicit use of existing design criteria. The sizing process for these criteria is relatively mature and exists within proprietary company capability for slab metallic panels, laminates, and to a limited extent for stringer stiffened panels, although this is by no means a mature integrated capability in all codes. An overview of current sizing criteria is shown in Appendix A and described by Duysinux and Fluery (1993), ap C. Harris (1997b) and ap C. Harris (1997a).

Requirements for future aircraft are becoming more sophisticated and placing further implicit constraints on the structure through the interaction with other disciplines. A major influence on many future military airframes will be techniques to reduce the observability of the aircraft. The overall shape, path of engine intake ducts, storage of payload, shape of localised features and use of materials will be controlled to reduce the observability of the aircraft (Rao and Mahulikar (2002)). Moreover, since this technology needs to be designed in at the outset to be useful (Pywell (2004)), then it is necessary to be able to represent these criteria when sizing the aircraft. An example is the control of wing panel out-of-plane displacement in the optimisation in order to minimise the observable signature of the vehicle through structural deformation away from the 'ideal' shape. This may be a constraint that could be controlled using existing bending stiffness criteria methods, and is discussed later in Section 6. However it is not yet clear how this displacement is to be measured, what the tolerances are and whether existing methods are capable of sizing for these criteria. It is the interaction of these different disciplines that drives the need for a multidisciplinary sizing approach. Moreover, their varying importance between products mean that an efficient method of representing these criteria and integrating them into a sizing process needs to be found so that they do not become an overhead when not in use.

Uninhabited air vehicles (UAVs) are becoming increasingly popular for military applications because they reduce the number of personnel required to operate on the front-line of the battlefield, can reduce initial and whole-life costs and have a greater range and endurance than equivalent manned vehicles (Bushnell (2003)). It is unclear how the structural design criteria for UAVs will differ from existing manned aircraft, but there is concern that the current design criteria are not sufficient. The General Atomics' Predator, the US Air Force's most reliable UAV had 32 'Class A' incidents per projected 100,000 flight hours compared to 3 per 100,000 for the manned F-16 fighter, an order of magnitude difference. Since these aircraft are likely to carry equipment such as sensor payloads worth up to $2million then it is likely that they will need to be designed for increased reliability to ensure reasonable operating costs over the lifecycle (Hoyle (2003)). This implies that the design processes used will have to maintain similar standards to current processes.

Emerging structural technologies such as 'morphing' wings (Danieli et al. (2004)) and adaptive internal structures (Cooper and Kittipichai (2004)) will require different approaches to analysis and design. These are likely to require a greater emphasis on the bending and membrane-bending coupling properties of structural components when sizing rather than the traditional tendency to size mainly for membrane effects (Thompson (1999)). Since these technologies are still being developed then it is not clear what the sizing requirements of these techniques would be. A future system would need to be flexible enough to include them at a later date.

As the use of laminated composite materials for airframe structures increases, so the desire to improve the design of these structures will increase, both to reduce the mass and increase the manufacturability of the design to reduce subsequent work required for manufacture. One method of achieving this is to substitute the current 'pseudo-laminate' representation in the initial structural model with a model that includes additional design criteria, such as ply-blending, ply-blocking and damage tolerance criteria (Niu (1999), Middleton (1990)). Recent work in this area is discussed in Section 3. (Liu et al. (1998), Soremekun et al. (2002), Seresta et al. (2004) and Stephens and Toropov (2004))

As well as conventional materials used on manned aircraft (e.g. metallics and uni-directional / woven fabric composites), it is possible that new UAV / UCAV designs might utilise materials such as fibreglass and composites made from fibre placement techniques. Whilst it should be possible to adapt existing analysis and sizing methods, any significant differences in manufacturing constraints will need to be included in the model.

### 2.3.3 Modelling and Analysis Requirements

Reviews of the changes in structural analysis using finite element models have shown that the representation of products is becoming increasingly complex (Knight Jr. and Stone (2002), Venkataraman and Haftka (2002), Vanderplaats (2002)). Venkataraman and Haftka (2002) discussed the increasing complexity of a FE model used to model the same aerospace structure over a period of 20 years. In 1980 this structure was studied using a model with 200 degrees of freedom (DOF), increasing to 30,000 DOF in later models through to 800,000 DOF in 2002. This indicates a shift in emphasis away from the use of FEA to obtain an indication of global structural performance, towards FE models which indicate the performance of individual structural components. It should be noted that the number of DOF could be increased locally, in regions of interest, without having to increase the mesh resolution throughout the model.

Aircraft structures are usually sized using a selection of the most extreme loadcases that an airframe will experience (Sobieszczanski-Sobieski (1999), Niu (2001)). The number of load cases used for recent optimisation procedures has included 41 for the A400M rear fuselage (Schuhmacher et al. (2004)) or, in the case of the example given by Sobieszczanski-Sobieski (1999), 60-100. Table 2.4 shows an example for a supersonic transport aircraft. This was agreed to be typical of problems currently modelled by engineers at BAE Systems, although it was noted that the total number of candidate loadcases could be in the region of 4000. For each load case there can be a number of constraints that need to be satisfied, such as strength, stiffness and buckling stability.

| | |
|---|---|
| **Number of flight conditions** | |
| Subsonic | 5 |
| Supersonic | 3 |
| **Number of loading cases** | |
| Total examined | 400 |
| Selected for inclusion in optimisation | 60 |
| **Structural analyses** | |
| Number of elastic degrees of freedom | 50k |
| **Number of design variables** | |
| Configuration shape | 50 |
| Cross-sectional dimensions | 3000 |

Table 2.4: The dimensionality of a supersonic transport aircraft design problem (Sobieszczanski-Sobieski (1999))

## 2.4 Computational Engineering Toolset

### 2.4.1 Software

Structural analysis and design within the aerospace industry is a sufficiently mature discipline that there are high levels of experience with a few analysis and design packages (Knight Jr. and Stone (2002)). In the case of finite element analyses MSC/NASTRAN is the industry standard for linear static analysis, ABAQUS is widely used for non-linear analyses and LS-DYNA is widely used for dynamic analyses. The CATIA geometry definition tool has been widely adopted within the aerospace industry and is used by BAE Systems, Lockheed-Martin, Boeing (Knight Jr. and Stone (2002)) and Airbus. The makers of CATIA have recently purchased ABAQUS illustrating further consolidation in the software sector.

Standardisation towards a few common tools has let to their increased commoditisation, of which NASTRAN is an example. Developed by NASA as the Nasa Structural Analysis code, it was licenced to a number of independent software vendors of which MSC was one (Schaeffer (1979)). Versions of MSC/NASTRAN became the industry standard which MSC had control over until a 2002 US anti-competitiveness ruling that recent versions of MSC/NASTRAN should be made available to competitor companies that wished to sell it. Thus, an industry standard version of NASTRAN is now available from a number of different vendors including NX, and NEi. However, it is not clear how these companies will develop their versions, how levels of support compare or indeed the levels of in-house knowledge of the product. Many FE codes are tied into wider product ranges, or affiliations with vendors of other tools, and increasingly licenced using a token system (Clarke (2003))). This enables companies to buy a fixed amount of capability, of which the exact product mix is determined at the point of usage rather than at the point of licence negotiation.

These ranges tend to be developed through 'networks' of 'partners', suppliers of related, but not normally competing, software. These networks should help mature a collection of tools into an integrated, interoperable toolset as described by Tidd et al. (1997) for similar co-operative groups. In doing so they are establishing processes for interoperability of these tools and to a limited extent moving up the value chain. If the argument that processes will become standardised holds true (Thurow (1996)) then developing and standardising these processes will be a service provided by this network rather than in-house.

### 2.4.2 Hardware

Since ECLIPSE was first developed it has been adapted to run on a number of computer platforms. These have included an IBM mainframe, Cray, DEC VAX and SGI Origin supercomputers. However, recent IPTs have moved away from the supercomputer approach and moved towards desktop workstations with multiple processors, leading to a wider range of technical computing capability.

Current outsourcing of non-core competencies has meant large organisations typically use IT infrastructure provided and managed by companies such as CSC (BAE Systems), and EDS (Rolls-Royce). In doing so the organisations relinquish a degree of control over their infrastructure and instead use agreed processes for management of the resources. This includes tasks such as approving software for use, installing software, supporting the products and acting as an interface to the suppliers. These agreed processes allow the supplier to work around a set of known parameters when planning how to make a profit. Agreements usually exist for process engineers to have a greater degree of control, sometimes by giving them 'development' machines where they can modify the software configuration. However these engineers often still feel constrained by the processes used to manage the infrastructure and desire a greater degree of flexibility and responsiveness. This conflict in the implicit goals of the process engineer and the IT supplier could, for example, lead the process engineer to design the process to require minimum interaction with the IT supplier. Thus the nature of the relationship between the IT supplier and the process engineer can affect how the process is designed and implemented.

Within the technical computing community the use of clusters of COTS computing hardware has increased with the availability of the Linux operating system over the last decade (Wang et al. (2005)). These offer the ability to provide a computational resource using high performance, COTS PCs, thereby reducing the initial barrier to entry to high performance computing by using relatively low priced hardware compared to traditional supercomputers. However, the combination of long-term supercomputing contracts, together with the relatively specialised nature of this market and the issues surrounding open-source software have meant that companies such as BAE Systems have had issues acquiring access to this type of resource.

'Grid' computing is an emerging service which allows computational resources to be shared across networks such as the Internet (Hey and Trefethen (2002)). This will allow computational power to be commoditised as 'processor farms' are used to perform calculations for customers linked to the Grid. It will also enable software vendors to offer their products on a pay-per-usage basis rather than a yearly licence.

The advantage of this charging model is that the user no longer needs to manage or maintain the assets used to perform the calculation, potentially increasing the efficiency of the organisation since these activities are performed more effectively elsewhere. However, the potential issues with this model include a lack of control over the infrastructure and software since this has been delegated to the supplier. Second, the supplier will need to adjust their pricing model to reflect the potential uncertainty in revenue compared to a fixed licence of a year or more, and ultimately ensure a sustainable business.

# Chapter 3

# Structural Sizing - Technology Review

This chapter provides a review of the technology discussed in Section 2.1 to structurally size aircraft, and some that could solve some of the problems cited by engineers not currently using the technology. Section 3.1 introduces generic optimisation techniques that form the basis for many of the techniques introduced later in the chapter. Section 3.2 looks at the methods used to size components of generic shape which are routinely used in the aerospace industry. Section 3.3 looks at the sizing of arbitrarily shaped components. Section 3.4 brings together many of these earlier concepts and discusses their use in sizing whole vehicles. Section 3.5 gives an overview of typical structural sizing software available within a large aircraft manufacturer and Section 3.6 looks at methods used to integrate this software into the design process.

## 3.1 Generic Optimisation Techniques

The aim of an optimisation process is to find the best solution to a problem within the constraints imposed both by the problem itself and the resources available. This is achieved by selecting an efficient search method to either maximise or minimise an objective function whilst satisfying any associated constraint functions. This can be expressed as:

$$\text{Minimise } f(\mathbf{X})$$

where

$$\mathbf{X}_i^L \leq \mathbf{X}_i \leq \mathbf{X}_i^U \text{ for } i = 1..m.$$

subject to the $n$ inequality constraints

$$g_j(\mathbf{X}) \leq 0 \text{ for } j = 1..n$$

and subject to the $p$ equality constraints

$$h_k(\mathbf{X}) = 0 \text{ for } k = 1..p$$

The region of the search represented by the limit of these variables is known as the 'design space'. Within the design space a potential solution **X** is known as a 'design point'. In airframe structural optimisation the objective is usually to minimise the mass of the structure whilst satisfying various performance constraints. Such constraints vary, but for aircraft typically include strength, resistance to buckling and stiffness properties. Limits for variables are usually determined by manufacturing considerations, although these can also be represented using constraint functions.

It can often be difficult to classify a function as either an objective or a constraint. Whilst specific values of constraints may be required, unless they are equality constraints it is often desirable to also represent them in the objective function. For example, structural design problems often have minimum design criteria, such as the load factor at which a structure will buckle. However in the case of two solutions of equal mass, both of which can withstand resist buckling under the applied load, the structure which can resist buckling under the highest applied load will be selected. An example of this type of problem is demonstrated in Section 4.2.

### 3.1.1 Gradient based optimisers

Using information about the rate of change in the objective and constraint functions at a design point it is possible to infer the direction in which a 'better' design can be found, either in terms of the objective function, or in satisfying the constraints. A number of strategies exist for searching a design space using this information, and include relatively simple methods such as Newton's method (Press et al. (1998)). Commonly used methods within BAE Systems are those found in the DOT optimiser (Vanderplaats (1999a)), which is the optimiser used within NASTRAN SOL200 (Moore (1994)) and ModelCenter (Inc. (2000)). This uses the Method of Feasible Directions, Sequential Linear Programming (SLP) and Sequential Quadratic Programming methods. SLP is also used within ECLIPSE because it is relatively easy to code and uses straightforward optimisation techniques (Press et al. (1998)). It creates successive local linear approximations of the objectives and constraint functions from which optimisers such as the Simplex method are used to suggest a new point that has a better objective value or better satisfies the constraints. The process is repeated until convergence as shown in Figure 3.1.

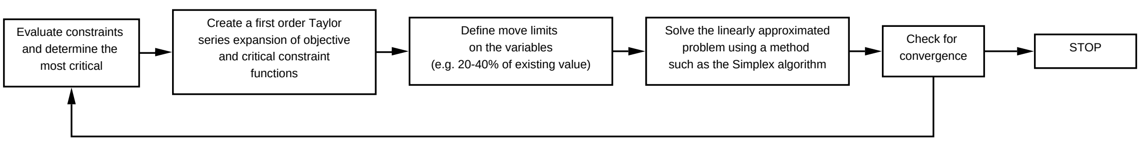



FIGURE 3.1: The Sequential Linear Programming Method

These algorithms are usually used where the variables, objective function and constraint functions are continuous since this allows gradient information to be obtained relatively straightforwardly. Moreover they are limited to problems where the functions are unimodal, since the gradient information cannot impart knowledge about optima outside of the local region. One of the major limitations of these methods is their scalability, since they become more inefficient when there are large numbers of variables and active constraints (Vanderplaats (2002)).

### 3.1.2 Optimality Criterion Methods

For a set of continuous, differentiable, objective and equality constraint functions, $f(\mathbf{X})$ and $h_j(\mathbf{X})$, that satisfy the Karush-Kuhn-Tucker (KKT) conditions (Bertsekas (1982), Khot (1982)), it is possible to determine a criterion which is satisfied at the optimum. At stationary points, $\mathbf{X}^*$, the gradients of objective and constraint functions ($\nabla f(\mathbf{X})$ and $\nabla h_j(\mathbf{X})$) are perpendicular to their respective functions. An optimum will exist if the vectors of $\nabla f(\mathbf{X})$ and $\nabla h(\mathbf{X})$ are parallel. At this point $\nabla f(\mathbf{X}^*) = \lambda_j \nabla h_j(\mathbf{X}^*)$ for all constraints $j$, where $\lambda_j$ is known as a Lagrange multiplier, creating a 'dual' problem where both the design (primal) variables $\mathbf{X}^*$, and the Lagrange multipliers (dual variables) must be found. By knowing the values of $\lambda_j$ at which the optimum occurs it is possible to calculate the solution $\mathbf{X}^*$ and vice-versa. This is achieved by constructing relationships between the primal and dual variables. It is then possible to iterate towards values of these variables that satisfy the optimality criterion. This is achieved by forming a 'Lagrangian':

$$L(\mathbf{X}, \lambda) = f(\mathbf{X}) + \sum_{j=0}^{n} \lambda_j h_j(\mathbf{X}) \tag{3.1}$$

which when differentiated with respect to $x$ will give a condition for a stationary point, $\mathbf{X}^*$:

$$\frac{\partial L}{\partial x} = \frac{\partial f}{\partial x} + \sum \lambda_j \frac{\partial h_j}{\partial x} = 0 \tag{3.2}$$

In doing so a solution $\mathbf{X}^*$ has been found which minimises the objective function and satisfies the constraints. The KKT state that for this equality problem that in addition to Eqn 3.2

$$\lambda_j \geq 0 \quad h_j(\mathbf{X}^*) = 0 \tag{3.3}$$

One iteration method used to find values of the primal and dual variables is Newton's method (Bertsekas (1982)). This technique is useful where the number of design variables is significantly more than could be solved efficiently using the methods previously discussed. It requires the number of constraints to be less than the number of design variables, and so becomes ineffective when the number of constraints becomes too high (Vanderplaats (1999b)).

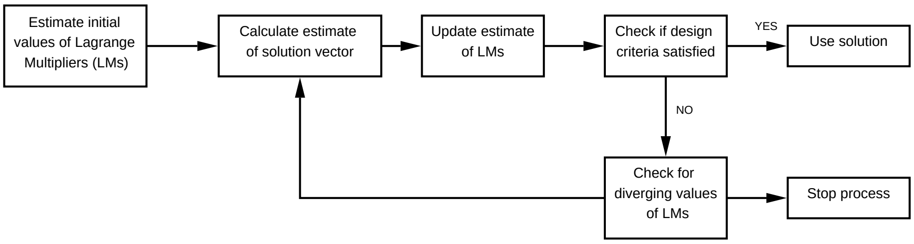



FIGURE 3.2: Operation of Lagrange multiplier method

### 3.1.3 Genetic algorithms

Genetic algorithms are a directed random search of a design space through the evolution of a population of potential solutions (Goldberg (1989)). Each of these solutions is represented by an encoded string analogous to a chromosome. The chromosome consists of genes which can take one of a finite number of values or 'alleles'. The design space is encoded by linking the values of these genes in 'genetic' space to settings of design variables in the 'real world' space. Figure 3.3 shows how the process works. The population evolves through a number of generations with new members created at each generation using genetic operators. Two common types of operator are 'crossover' where new members are created which combine features of existing members and 'mutation', which explores the design space by creating new members based on randomly perturbed encodings of existing solutions. After evaluating all the new members the initial members of the next generation are selected using techniques that bias the selection towards the 'fittest' members, defined by their objective function values.

Genetic algorithms allow parameterisation using discrete variables and the use of discontinuous functions for which it would be difficult to collect gradient information. They can be used to study objective functions that exhibit many optima, since they simultaneously search different regions of the design space. The main disadvantage of these methods are that they require objective and constraint function evaluations for each member of a generation. If these functions are computationally expensive then this method can become ineffective.

## 3.2 Structural Sizing of Airframe Panels

The primary function of a wing skin panel is to form an impermeable surface on which to support the aerodynamic pressure distribution that provides the lifting capability of the wing (Megson (1990)). They are typically thin structures, which while efficient for

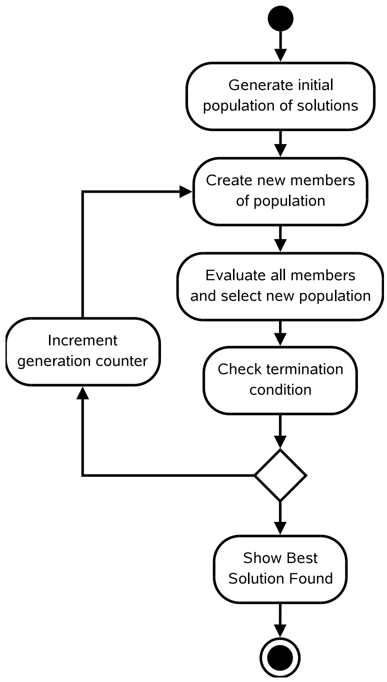



FIGURE 3.3: Operation of a genetic algorithm

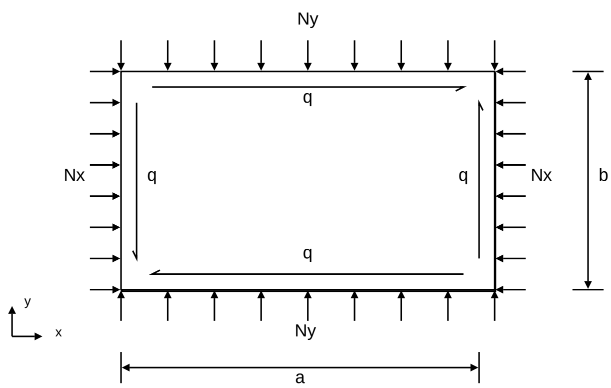



FIGURE 3.4: Example of a wing skin panel for sizing

resisting in-plane tensile loads, buckle under comparatively low magnitude compressive loads. Since a minimum mass structure is required then it is necessary to stiffen these panels to resist buckling, whilst minimising the mass of the airframe. Methods typically used in favour of metallic 'slab' panels include laminated composites, honeycomb sandwiches and the attachment of stiffeners (Niu (2001)).

In detailed stress calculations such components are traditionally sized manually using standard empirical formulae such as those found in Niu (2001), British Aircraft Corporation (1965), ESDU (2005), Young (1999) and Bruhn (1973). The design criteria used are local to the component, such as strength, buckling and manufacturing criteria. They do not usually consider sizing interactions with other components except for extraction of realistic boundary conditions. An example of the sizing of a wing skin panel is given in Figure 3.4. The design criteria state that the panel must not fail at strains below that experienced at 'ultimate' load, or buckle under loads less than the 'limit' load for metallics, or ultimate load for composites. Niu (1999) defines limit loads as the maximum loads anticipated on the structure during its service life. The definition also states that the structure should not suffer permanent, detrimental, deformation when experiencing limit loads. Niu (1999) defines ultimate loads as greater than the limit loads, usually by a factor of 1.5. Under ultimate loads a structure is simply expected to not fail. This factor is used to provide an additional margin of safety to account for the approximations used in the design process as well as uncertainty in the operational conditions in which the structure will operate.

$$\lambda_{strength} = \frac{\sigma_{ultimate}}{\sigma_{actual}} \geq \lambda_{required} \tag{3.4}$$

$$\lambda_{stability} = \frac{\sigma_{critical}}{\sigma_{actual}} \geq \lambda_{required} \tag{3.5}$$

The standard process for manually sizing a wing panel is to create a free-body-diagram (FBD) of the proposed design that is suitable representation for an analysis method. The geometry and boundary conditions of the FBD are approximations made based on the available design and loading information from the CAD model and overall, limited fidelity FE analysis results respectively. An example is the sizing of wing skin panels using the CITS panel stability method for flat panels with no stiffeners. The panel geometry is approximated to a rectangular panel as shown in Figure 3.4. Edge fixity conditions are approximated based on the known support conditions at the panel edge, somewhere between simply supported and built-in edge conditions. Loadings are extracted from the FE analysis, averaged between opposite sides of the panel and converted to a linearly varying edge load between the two edge points on either edge. Additional methods exist for panels which require better representation of the geometry, for example the representation of curved panels Niu (2001). The process of sizing these panels is represented in structural sizing codes such as ECLIPSE. Figure 3.5 shows a panel represented by four finite elements E1-E4. Here FBDs are created for each element in the model being sized. The geometry is approximated using manually pre-determined scale factors on the element size and loading is approximated from the stresses within the element. In the case of the panel shown in Figure 3.5:

$$a_{E_i} = W_1 * E_{i_{length}} \tag{3.6}$$

$$b_{E_i} = W_2 * E_{i_{width}} \tag{3.7}$$

The scale factors $W_1$ and $W_2$ must be manually defined when defining the panels representations in the ECLIPSE model (BAE SYSTEMS (1999)). The user also has to ensure that the orientation of the two values is correct with respect to the element.

Whilst this process was acceptable in models of relatively low complexity, where panels would be represented by 2-3 elements, it becomes significantly more time consuming in models with orders more elements. An alternative method used by codes such as Hypersizer(Collier Research Corporation (2003)), and later in this research (see Chapter 5) is to approximate the panel geometry using the information from the associativity of elements in the model. To approximate this model an average value of $a$ would be found from $a_1$ and $a_2$ and likewise for $b$.

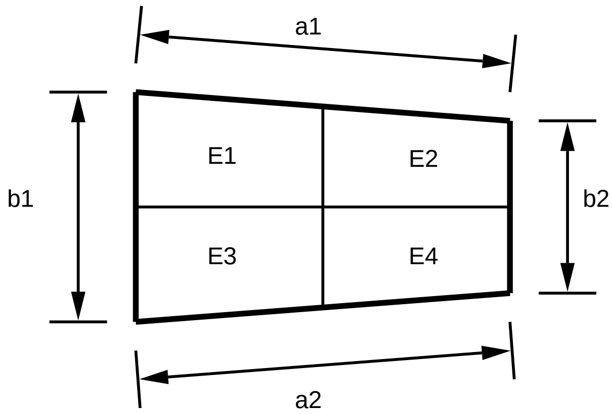



FIGURE 3.5: Geometry approximation methods

### 3.2.1 Metallic slab panels

For metallic slab panels the only variable is panel thickness, $t$, and these can be sized using a stress ratio method. For compressive and shear buckling loads the load factor $\lambda_{buckle}$ at which the panel buckles is given by CITS using finite strip theory (ESDU (2005)) or can be found using classical laminate theory as shown in Appendix B. In this work it was assumed that panels were to be sized for membrane effects only. Stress ratio methods, which invert the strength and stability equations (BAE SYSTEMS (1999)), are used here to find the optimum panel size rather than using a more generic method. The scale factor, $S$, for in-plane stress was given by:

$$S = max\left(\frac{\sigma_1}{\sigma_{AL}}, \frac{\tau_{max}}{\tau_{AL}}\right) \tag{3.8}$$

where $\sigma_{AL} = \sigma_{TA}$ or $\sigma_{CA}$ depending on the sign of $\sigma_1$

and for buckling by:

$$S = \sqrt[3]{\frac{\lambda_{req}}{\lambda}} \tag{3.9}$$

### 3.2.2 Carbon fibre composite panels

Carbon fibre composites (CFCs) are used in aircraft because of their favourable stiffness-to-mass and strength-to-mass ratios, and the design freedom to determine the membrane and bending properties of the panel layup using permutations of these laminae (Middleton (1990)). Panel membrane properties can be controlled by the relative percentages of different ply orientations, whilst the bending properties can be determined by the order, or 'stacking-sequence' in which these plies are assembled as shown in Figure 3.6 (Zenkert (1995)).

Whereas slab metallic panels have only one design variable, the extra design freedom in laminated composites increases the dimensionality of the problem. Potentially it is possible to vary the thickness, in-plane orientation and material type for each ply in a stacking-sequence. However, in practice the two formulations in use are 'fixed stacking sequences' (FSS), where ply thicknesses are varied in a fixed stacking sequence; 'variable stacking sequences' (VSS), where ply orientations are varied whilst ply thicknesses remain constant. Section 2.3.1 noted that the laminate layup is often designed in two stages, the first a sizing for the proportion of the different laminate types (to create a 'pseudo-laminate') and the second a determination of the actual layup for the detailed design which includes localised thickening of the layup around features such as bolt groups.

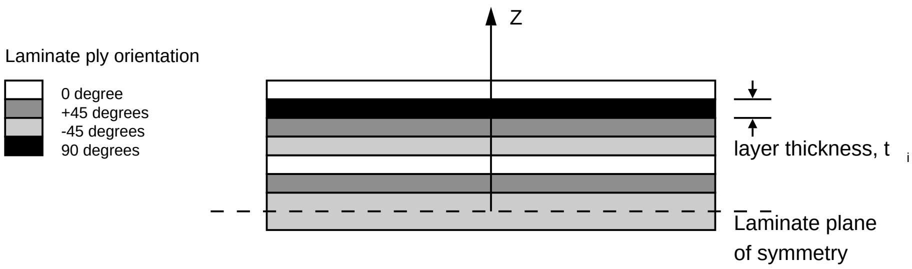



FIGURE 3.6: Example of a composite stacking sequence

Calculations for strength and stability criteria are based on classical laminate theory, as described in Zenkert (1995), an overview of which is given in Appendix B. The formulation of the stiffness matrices, A, B and D, make the calculation more complex and computationally expensive than that for a simple metallic panel. This process is simplified within BAE by the use of a CITS method for panel stability codes which calculates these matrices and performs the structural analysis using a finite strip method (ESDU (2005),BAE Systems (1999)). CFCs have additional constraints that need to be met by the design before the design can be manufactured. These are shown in Table 3.1 and can be included in the design sizing using a variable stacking sequence formulation.

For many applications, especially in the early stages of design, it is sufficient to size a CFC panel for its relative composition of ply orientations (Starnes Jr and Haftka (1979)

| Design Criteria | Description |
|---|---|
| Ply Blocking | To prevent interlaminar shear effects such as delamination under static and / or cyclic loading (see Niu (1999)) |
| Damage Tolerance | May be augmented by placing the least strength critical plies near the outer mold line of the surface (see Niu (1999)) |
| Ply Blending | To prevent stress discontinuities and hence concentrations at panel boundaries |

TABLE 3.1: Additional design criteria for laminated composites

and BAE SYSTEMS (1999)) to produce 'pseudo-laminate' designs. This formulation will indicate the likely size and hence mass of the component with the exact layup being determined later in the process. In such cases the problem is formulated as a fixed stacking sequence, usually a symmetrical sequence of the possible ply orientations repeated 2-3 times as shown in Figure 3.6. The number of variables required to describe this problem is therefore the number of different ply orientations in the fixed sequence multiplied by the number of times that sequence occurs in one half of the total stacking sequence. Usually the sizes of adjacent $\pm 45$ deg plies are linked to create a balance layup, reducing the number of variables required. Thus the FSS layup shown in Table 3.2 would be described by 9 variables. Membrane properties are controlled by simultaneously changing the thicknesses of plies of a given orientation, whereas the bending stiffness can be varied by changing individual ply thicknesses. This formulation is suitable for use with gradient based optimisers since the variables are continuous and the objective and constraint functions differentiable using numerical methods. For this type of formulation Starnes Jr and Haftka (1979) use the SLP algorithm as does ECLIPSE.

| Formulation | Example layup |
|---|---|
| Fixed sequence | $[(0/\pm 45/90)_3]_{symmetric}$ |
| Variable sequence | $[(90)_2/\pm 45/90/0/(\pm 45)_3/(0)_2]_{symmetric}$ |

TABLE 3.2: Composite layup notation - a subscript denotes multiple copies of that ply or plies

The variable stacking sequence formulation offers the ability to vary the through thickness properties of the panel to a greater extent, allowing improved out-of-plane stiffness properties for criteria such as panel buckling. In addition, composite design criteria can be considered because the exact layup is defined. To reduce the effect of interlaminar shear, and hence matrix cracking, a ply-contiguity constraint can be defined to limit the number of adjacent plies of the same orientation (Le Riche and Haftka (1993)). To increase the manufacturability of a design, and reduce stress concentrations at panel edges, plies are blended between adjacent panels. Merit functions, which measure the amount of blending between adjacent panels in a design can also be constructed (Soremekun et al. (2002)). Finally, damage tolerance criteria, such as the suggestion that the least strength critical plies (typically $\pm 45°$) should be located towards the outside of the layup (Niu (1999)) could be included. The number of variables required to describe this problem is therefore the maximum number of layers in one half of the stack.

Genetic algorithms are typically used to solve this combinatorial problem, since they allow the use of discrete variables to represent the layup, whilst also being capable of searching a design space which has many optima. In general the GA is directly linked to a local panel analysis code, such as CITS. An early example of this method was to maximise the buckling properties of a layup of fixed number of plies, whilst satisfying strength and ply contiguity constraints (Le Riche and Haftka (1993)). Subsequent work has increased the effectiveness of these methods by considering the use of binary trees to store previous solutions (Kogiso et al. (1993)), the use of local improvement techniques to improve the solution by 'repairing' it at the chromosome level ('Lamarckian' repair) or at the physical laminate level ('Baldwinian' repair) (e.g. Kogiso et al. (1993), Liu et al. (1998)). More recently the focus has moved towards inclusion of criteria to enable the sizing and manufacture of whole systems from composites. Work such as that by Liu et al. (2000) has considered a decomposition approach to sizing a wing subject to strength and buckling constraints. Here a number of panel sizings were carried out for these criteria and the results used to train a response surface. The response surface was then used in place of this lower level panel optimisation. Soremekun et al. (2002) and Seresta et al. (2004) have considered sizing structures for local constraints, whilst satisfying global ply-blending constraints. Recent work by by Airbus and Altair (Stephens and Toropov (2004)) has sized a composite wing rib modelled using GA linked to a FE model consisting of local layup zones. Design criteria included strength, stability, ply-blocking and ply-blending. Evaluations of members of the GA population were performed in parallel to reduce the time taken to evolve the solution. The results of designs were also stored to reduce the computational expense of repeat evaluations.

### 3.2.3 Stringer stiffened panels

Metallic stringer stiffened panels have traditionally been sized using manual iterations of empirical data sheet methods, such as those given in Niu (2001), British Aircraft Corporation (1965), Bruhn (1973), or through the use of accompanying codes such as that provided by ESDU (ESDU (2005)). The increased complexity of composite stiffened panels means they need to be sized using an automated method such as the panel sizing codes 'VICONOPT' (Butler et al. (1999)) and 'HyperSizer' (Collier Research Corporation (2003)).

Stiffened panels are parameterised using a mixture of discrete and continuous variables. For a metallic integrally stiffened panel the variables are simply the number of stiffeners on the panel (a discrete variable), the stiffener thickness and height and panel thickness (continuous variables). In multilevel optimisation problems the parameterisation is usually simplified by setting a fixed stiffener pitch and using a gradient based optimiser to size the continuous variables. This approach does not find the true optimum, but gives an indication of the potential weight savings available by using stiffened panels in place of slab panels. Multilevel approaches are discussed in Section 3.4.

## 3.3 Structural Sizing of Components of Arbitrary Shape

In many cases a greater degree of control is needed over the design to further reduce the mass or increase the structural performance of a component. The three commonly used optimisation methods shown in Figure 3.7 are: 'topological' optimisation - which varies the layout of material within a defined boundary; 'shape' optimisation - which varies the boundaries of the material used in the structure by parameterising the boundary geometry; 'sizing' - which sizes areas of material selected on the basis of likely requirements for local areas of similar thickness material.

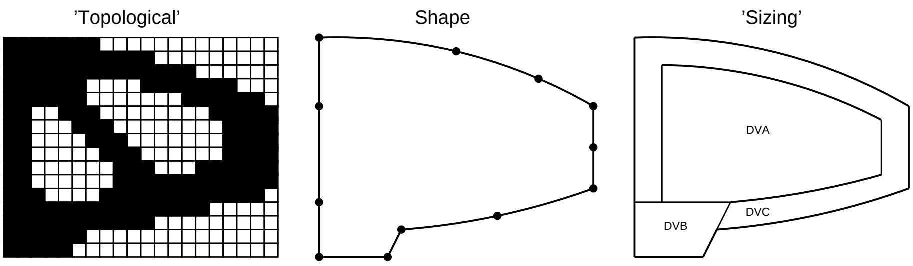



FIGURE 3.7: Methods of sizing a wing rib tip (after Kicinger et al. (2005)

FE packages such as NASTRAN and ANSYS have some limited optimisation capabilities, usually enabling material thickness or basic shape parameters to be used as variables, sizing the structure for strength, stiffness and stability criteria (Vanderplaats (1999b)), although this is changing with the inclusion of BIGDOT in NASTRAN2005 for topological optimisation. Such 'Top-level' optimisation typically can control the distribution of material in one dimension giving a limited sizing capability, or two dimensions giving a shape optimisation capability. NASTRAN SOL200 is based on the DOT suite of optimisers, containing the method of feasible directions (MFD), sequential linear programming (SLP), and sequential quadratic programming (SQP) gradient based optimisers (Moore (1994)). SOL200 also contains a Fully Stressed Design (FSD) optimiser for strength cases. However, the gradient based methods are limited in the number of parameters they can handle efficiently and the FSD method is only applicable to strength criteria.

Within the aerospace and automotive industries two commonly used topological optimisation methods are 'Homogenisation' and 'Evolutionary Structural Optimisation' (ESO) methods. Both methods use an initial design space modelled using a FE mesh. Homogenisation methods parameterise the space by varying the 'porosity' of the material in an element which manifests itself as a change in the bulk properties of the material in that element (Rozvany et al. (1995)). The structure is then typically optimised using an optimality criterion method which aims to minimise the mean compliance subject to a constraint on the volume fraction of material used (Bulman and Hinton (1999)). Since creating areas of porosity in a structure would require many expensive machining operations a penalty function is often introduced to measure, and hence used to reduce, the cost of manufacture. The 'Solid Isotropic Microstructure with Penalty' (SIMP) method (Rozvany et al. (1995)) introduces a fabrication cost to offset the reduction in material cost of intermediate porosity regions. The fabrication cost increases as porosity decreases, although for very low porosity regions this tends to zero.

The ESO method uses the assumption that material within the design space that has low relative stress or strain energy density is being used inefficiently and hence can be removed from the structure (Steven (1997)). This is typically achieved by varying the modulus of elasticity of the element or simply deleting it from the FE mesh (Bulman and Hinton (1999)).

## 3.4 Structural Sizing of Airframes

A conceptual FE model of an airframe typically models major components in sufficient detail that the airframe can be sized and likely properties of the component determined. For instance, a wing skin panel might be represented by as little as one element, sufficient to indicate the likely mass of the panel and the magnitude of loads it will experience. Often the model is increased in resolution around areas of specific interest such as engine and control surface attachment points. Detailed features, such as the stringers on wing skin panels, are usually modelled using approximations where rod elements will be sized to represent a number of parallel stringers. This reduces the complexity of the model which in turn reduces the computational cost of the model, the development time required to define the features, and the complexity of the optimisation problem.

It is worth noting the complexity of potential sizing problems which could be constructed for an airframe. For a 'typical' airframe such as the UAV structure described later in Chapter 5 the optimisation problem was restricted to 103 panels for a metallic model of half of the structure. These panels potentially could be manufactured from metallic slabs, laminated composites, or stiffened panels. As noted previously laminated composites and stiffened panels typically increase the dimensionality of the problem by a factor of 9 and 4 respectively. Thus, the dimensionality of current problems is potentially O(900) variables.

The selection of optimiser is a trade-off of optimality of the solution against the time and computational expense of finding a solution. Normally the best solution found in a 'reasonable' time is taken as the optimum (Thompson (1999)). For the purposes of this discussion potential sizing methods are categorised as direct optimisers, which treat the analysis method as a 'black-box' evaluation function; optimality criterion methods, which require an understanding of the structural response and how the variables affect its response; and decomposition methods which break the whole airframe sizing problem down into sub-problems.

'Direct optimisation' in this context means the coupling of a generic optimiser with objective and constraint evaluation functions. In this case it is not necessary to understand the specific relationship between the variables and the evaluation functions, although this helps when selecting the optimiser to use for a particular problem. Since there is no requirement to utilise information other than the objective and constraint functions then it is usually a relatively straightforward process to link a direct optimiser with an evaluation function. However, the main difficulty with this approach is in formulating the problem and selecting an optimiser that will enable a reasonable answer to be found without excessive computational resource. Gradient based methods typically require function evaluations to calculate the sensitivity of the objective and constraint functions to each variable. Although only constraints which have violated the constraint boundary, or are close to doing so, are evaluated, the traditional gradient based method can be excessively computationally expensive to use for a large number of variables and active constraints. Moreover, they can often become trapped in local minima. Whilst global search techniques such as genetic algorithms are more likely to find a global minima, they can experience difficulties in finding the local minima, and can be prohibitively computationally expensive, especially when compared against the methods considered later in this section. Moreover, the fidelity of the current conceptual models is not sufficient to model all of the parameters using one evaluation function alone. However, the increases in readilly available computational power will reduce the real-world cost of performing these calculations. In addition improvements in hybrid optimisation techniques, which combine local search methods with global search methods, offer a potential future search method (e.g. the GLOSSY algorithm described in Keane and Nair (2005)).

Within the current toolset gradient based optimisers are limited to the MFD, SLP and SQP algorithms found in the DOT package used by both the NASTRAN SOL200 optimiser and the ModelCenter PSE. There is no readilly available global search method available within the current toolset. Thus direct optimisation approaches within the current environment are limited to simple structural models as the wing-box model used later in Section 5.4. Here it was possible to formulate the problem as an optimisation of O(10) variables, where variables were the thickness of wing skin panels.

The first of two methods used to solve this problem is the optimality criterion method discussed previously in Section 3.1. Here the relationship between the variables and the structural response must be known, or assumed, to construct a relationship between the variables and the Lagrange multipliers in the form of an optimality criterion. Khot (1982) describes implementations of the optimality criterion method for strength, stability and stiffness criteria. In this case the most critical of these criteria determines the particular method that is used to size the structure. The method described for stiffness criteria, and that used within ECLIPSE, uses virtual strain energies which represent deformation of the structure in a particular direction for different loadcases. A derivation of the method used in ECLIPSE is included in Appendix C. This method estimates initial values for the Lagrange multipliers. The assumed relationship between the values of the Lagrange multipliers and the virtual strain energies of the elements is then used to calculate the element sizes necessary to reduce the virtual strain energies to the target values. A new estimate of the Lagrange multipliers can then be calculated and the process repeated until convergence of the constraint values. A further FE analysis of the resized model takes place and the process repeated for a fixed number of loops.

The virtual strain energy represents a deflection of the structure in the direction of the displacement being measured. Typically formulations of this method, such as that used by ECLIPSE (Thompson (1999)), require a breakdown of the contribution to the virtual strain energy in the structure from its membrane and bending stiffnesses. This is found by modifying the FE strain energy calculation to 'cross' the displacement from a unit load in the direction to be measured with the displacement from the actual load (Equation C.6 in Appendix C). In NASTRAN this is achieved by modifying the solution sequences (e.g. the SOL101 linear static analysis method) which is defined using the NASTRAN Direct Matrix Abstraction Program (DMAP) language (Schaeffer (1979)). However, this returns total strain energies and not the components of the membrane and bending stiffnesses. ECLIPSE overcomes this problem by creating a model with duplicate, elements of identical size. One element is given material properties for membrane stiffness, and the other properties for bending stiffness. When this structure is analysed the strain energies produced represent the equivalent membrane and bending energies in the original structure. The disadvantage of this approach is first that the sizing process is heavily dependent on a particular version of the FE code since the solution sequences can change significantly between versions. Second, the routines used to size the structure are complicated by the need to recreate a model with duplicate elements.

The second method decomposes the global model of the structure into smaller sub-problems. Since these sub-problems are treated as independent then the effect on the global model must be considered. This is achieved by updating the global model with the new sizes and a further FE analysis to allow any redistribution of loads within the structure. This process is shown in Figure 3.8. Decomposition methods become especially useful if it is assumed that the global model can never fully represent the level of detail, and hence structural response, which of the structure being sized. However, they are limited to problems where the sub-problem is a sufficient approximation of the response of the sub-structure within the global model. Thus, they typically consider strength and local stability design criteria rather than global stiffness. An example of the benefit of this approach over the direct sizing of the whole model would be the sizing of stiffened panels. To accurately model, and then size, a stiffened panel with a buckling constraint the global FE model would need to model each panel in detail, varying parameters such as stiffener pitch. In order to vary the stiffener pitch the panel may need to be remeshed to allow stiffeners to be attached to panel nodes. The mesh would also need to be sufficiently dense to meaningfully represent a buckling response. The decomposition approach allows buckling responses to be modelled using a dedicated panel sizing code. The global model, with 'pseudo-stiffened' panels, models the displacement response. The cross-sectional area of a fixed number of rod elements are varied to modify the stiffness properties of panels. Panels are sized for stability criteria using a stiffened panel model and the two models linked together by creating stiffness targets.

For indeterminate structures this process can often lead to solutions which are not fully converged. Areas of structure often oscillate between two solutions as load continues to be resized and redistributed. Typically this is solved by the user reformulating the problem to link the resizing of structure in these areas. The decomposition method is used within ECLIPSE for metallic and composite slab panels. It is often used to enable standard structural calculations to be used as part of an automated sizing process. Ragon et al. (2003) use decomposition in a method for linking a global sizing process for strength and stiffness with a local sizing process for stiffened panel stability. The method linked a global sizing code, ADOP, with a local panel sizing code, PASCO, to size an untapered wing model. In this work the two codes were 'linked' through the use of an approximate, 'surrogate' model of the local sizing code, discussed later. However, in essence the information passed from the global model to the local model were the in-plane loads acting on the panel ($N_X$, $N_Y$, $N_{XY}$) and 'target' in-plane stiffness values ($A_{11}$, $A_{66}$). PASCO then attempted to find the minimum mass stiffened panel that met the buckling load factor ($\lambda_{stability} \geq 1.0$) and overall had the same in-plane stiffness (measured using 'smeared' stiffness approximations, $\bar{A}_{11}$, $\bar{A}_{66}$). The mass of this solution was then fed back to the global optimiser and fed into an additional top-level constraint aimed at minimising the difference between minimum mass solution found at the lower level and the solution found at the top-level. The purpose of the approximation of the lower-level sizing was to improve the response time between the two levels. Seresta et al. (2004) uses decomposition within a method for the design of laminates blended using a 'guide based' approach. Here a structure is sized by varying a template, or 'guide', layup and by varying the number of plies of that template in each panel of the structure. The template is varied at the global level in order to meet a stiffness constriant. At a local level each panel in the structure is assigned the guide stacking sequence. The proportion of the guide stacking sequence used is then varied to attempt to reduce the mass of individual panels whilst meeting local panel strength and buckling properties. Since all panels share a common layup template then there is implicit blending of plies between panels. This process is repeated until convergence or stopping criteria are met.

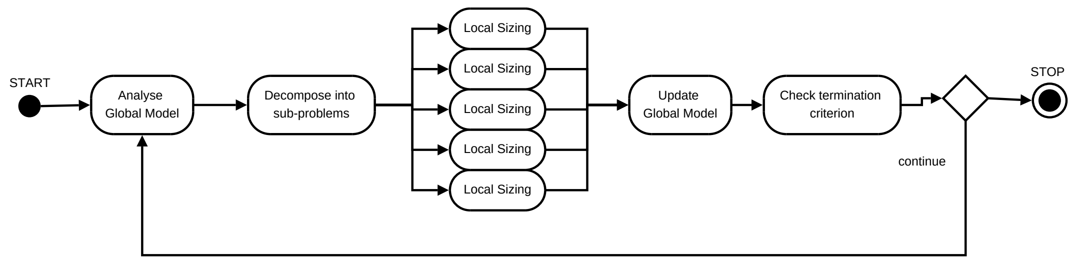



FIGURE 3.8: Example of a fixed composite stacking sequence

## 3.5 Structural Sizing Software

The current generation of proprietary software codes have been in existance in organisations such as BAE Systems since the early 1970's, maturing to meet the specific sizing needs of the companies and their design processes (Duysinux and Fluery (1993), ap C. Harris (1997b)). Within BAE Systems the ECLIPSE code was developed to provide engineers with a material distribution within an aircraft configuration that minimised the mass of the structure whilst meeting structural design criteria. These criteria included strength, stability, generalised deflection and aeroelasticity phenomena such as flutter.

The principle behind ECLIPSE was to use the results of a FE analysis of the structure to calculate the cross-sectional properties of elements in the model necessary to just meet the design criteria. The implication is that in doing so the mass of the structure will be minimised. The FE model would then be updated with these properties and reanalysed to allow for internal redistribution of loads. The process would then be repeated until the structural mass converged to a similar value between iterations, typically less than a 5% change, or until other stopping criteria were met. Strength and panel stability criteria were met using stress ratio methods (SRM) and sequential linear programming (SLP) for metallics and laminates respectively (as described in Section 3.2). The panel stability method was a decomposition approach, with panel geometry based on a manually defined scale factor of finite element width for a given set of elements representing a panel. This assumed that the element sizes within a panel were roughly uniform. Optimality criterion methods using an element strain-energy formulation, discussed previously in Section 3.4, were used to size metallics and laminates for stiffness related criteria (Thompson (1999), Thompson et al. (1999)).

Figure 3.9 shows a top-level overview of the ECLIPSE sizing process. A FE model would be analysed using NASTRAN and the resulting element stresses used to size the model for strength and stability criteria. The initial model would then be sized to meet stiffness criteria using the strain energy approach. The FE models obtained from these two sizings were then combined using a proprietary set of criteria to produce a structure that aimed to meet both sets of constraints. The sized model would then be reanalysed and the process repeated until convergence or other specified stopping criteria were met.

ECLIPSE has been used to size the EAP aircraft and Eurofighter Typhoon and Gripen wings. Other similar, proprietary, codes include: STARS (Bartholomew and Wellen (1990)); ASTROS (Neill et al. (1990), Canfield and Venkayya (1990)) and LAGRANGE (Schuhmacher et al. (2004)). These capabilities are summarised in Appendix A.

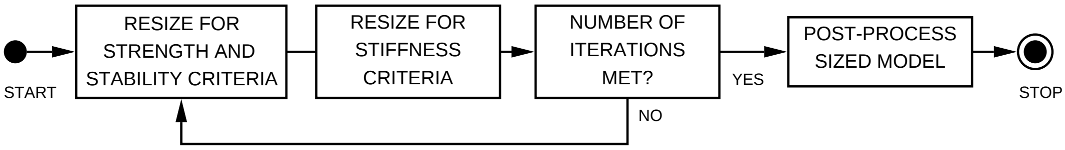



FIGURE 3.9: Air vehicle design process used for Eurofighter Typhoon

Commercial alternatives to proprietary capability are increasingly available both as stand-alone applications and as part of wider suites of packages. DOT (Vanderplaats (1999a)) and BIGDOT (Vanderplaats (2002)) are two such optimisers used within FE analysis codes such as NASTRAN and GENESIS. These optimisers are used for structural problems with low numbers of variables ($O(100)$) and very high numbers respectively ($O(100k)$). Within DOT it is possible to use the Method of Feasible Directions (MFD), Sequential Linear Programming (SLP) and Sequential Quadratic Programming (SQP) methods. BIGDOT uses an exterior penalty function method to create a representation of the objective and constraint functions for nonlinear problems. The minimum of this function is then found using a conjugate direction method (Vanderplaats (2002)). DOT is available as part of the GENESIS (Inc (2004)) structural optimisation toolkit, the NASTRAN SOL200 solution sequence (Moore (1994)) and Phoenix Integration's 'ModelCenter' problem solving environment. BIGDOT is used by GENESIS and NASTRAN v2005.

Altair Engineering's 'Optistruct' code is used within the automotive and aerospace communities, with a number of examples of its application on structural optimisation of vehicle components. It implements the SIMP layout optimisation method for strength, stability, and stiffness related criteria discussed previously in Section 3.3. It has been used by Airbus to size the wing leading-edge ribs on the A380 aircraft (Krog et al. (2004)).

An example of a COTS decomposition type optimiser is found in Hypersizer/Pro. Hypersizer is derived from the NASA ST-SIZE code and is used to analyse stiffened panels for failure criteria including strength and stiffness (Collier Research Corporation (2003)). It contains a number of empirical analyses for stringer stiffened panels with different stringer types, honeycomb stiffened panels and laminated composite panels. Hypersizer/Pro includes the ability to generate free-body diagrams from a FE model using the element stresses. These free-body diagrams can include consideration of the effect of panel curvature on failure criteria and pressure bending effects. Sizing then takes place for each free body panel, and it is possible to subsequently re-combine these in the FE model for later re-analysis in the FE package. The sizing process uses a design space discretised by assigning variables a fixed number of levels. The effect is that it is not possible to find a true optimum in a given sizing, but rather the value of the closest design point. The suppliers argue that one of the main benefits of this method is that it can search a design space described by discrete and continuous variables (Collier Research Corporation (1997)). However, since the analysis functions are accessible through a COM interface it should is possible to use different optimisation methods when necessary. Hypersizer has been integrated into a structural sizing process within the ModelCenter PSE (Cerro et al. (2002)).

## 3.6 Process Integration Techniques

There are three levels of information exchanged between components in a typical sizing system. The first is 'top-level' information specifying the problem being solved, its location; 'medium'-level information defines the structural sizing problem, both in terms of the model and the constraints being used to size it; the 'low-level' process information contains specific needed to size the model for a given loop. In systems such as ECLIPSE a number of different techniques are used to handle this information, although ideally the number of different techniques would be reduced, or standardised to reduce the dependence on specific operating systems or components within the process.

In ECLIPSE top-level information is handled using a machine specific script that controls the overall operation of the process, currently a Unix c-shell. Within the script medium and low-level information is passed between components of the process using NASTRAN input and output files and FORTRAN 'COMMON BLOCK' file formats. A requirement of the sizing process is that cross-strain energy calculations be performed for the stiffness sizing method. These calculations are performed internally to NASTRAN since the information required does not then need to be re-processed by routines that would have to be specially written. To achieve this the NASTRAN solution sequence is modified using the NASTRAN DMAP language as described in Section 3.4. One of the problems with this method is that solution sequences often change between versions of NASTRAN meaning that the DMAP modifications also need to be updated to allow ECLIPSE to operate with more recent versions of NASTRAN. This process is hampered by a lack of recent information on the DMAP language offered by the supplier. Instead the user often has to ask the supplier's technical support detailed queries.

An emerging alternative to the operating system specific shell script is the Problem Solving Environment (PSE). Keane and Nair (2005) describe PSEs as 'design integration systems', by which it is meant that the design team's codes are wrapped with a common interface which allows them to be linked together, at run time, without significant further effort, to meet the current needs of the team. Wrapping languages vary, but include PSE specific 'filewrappers' (ModelCenter), operating system specific scripts (VBScript

(Modelcenter) and shell script specific (Frontier)) through to platform independent Java (Modelcenter). All of these scripts are tied to the PSE that they have been written for. One major advantages of a PSE is that the process is integrated using a visual representation making it easier to understand and assemble novel processes, provided they are not too complex. However there are questions about long term management of processes within these environments, partly due to their relatively opaque management of component wrapper 'metacode', but also due to their relative immaturity. In addition the integration of computationally inexpensive calculations in a highly iterative process is undermined by the overhead of communication between components. Some PSEs still do not offer features necessary for true multidisciplinary process integration, such as simultaneous execution of components (e.g. execution of a CFD analysis at the same time as FE analysis). At some point a 'shake-out' of these methods is likely as the industry moves towards a standard methodology. It is likely that there will some standardisation in the PSE used in order for different organisations to collaborate. As such companies should consider how to abstract the wrapper code to minimise any overhead in moving between PSEs. Alonso et al. (2004) suggest the use of an emerging platform independent language, 'Python', as the basis of component integration. They argue that many of these issues can be solved using this approach. Other languages used in this role include Matlab and Tcl/Tk (Keane and Nair (2005)).

# Chapter 4

# Structural Sizing of Components

The purpose of this chapter is to understand the issues associated with using existing BAE analysis methods coupled to optimisation methods, both as a capability in their own right and in the case of the panel sizing method as a methodology for sizing panels in a decomposition type sizing system. Section 4.1 looks at the use of ECLIPSE as part of an Evolutionary Structural Optimization (ESO) style process. Section 4.2 examines the use of the CITS panel stability calculation to size a laminated panel.

## 4.1 An ESO Sizing Method Using ECLIPSE

One method for extending the use of ECLIPSE through the lifecycle, and thereby getting better utilisation out of it as a development overhead, would be to include a topological optimisation capability built around its existing sizing methodology. EADS have already achieved this with their sizing code LAGRANGE (Schuhmacher et al. (2004)). Two candidate methods were the homogenisation and ESO methods discussed in Section 3.3. The pre-existing optimality criterion method lends itself to the adoption of a homogenisation approach. However, this would require a redefinition of the optimality criterion and subsequent redefined relationships between element size and the Lagrange multipliers which it was not possible to achieve without a detailed understanding of the ECLIPSE source code. Instead, this study looks at feasibility of an Evolutionary Structural Optimisation (ESO) approach to sizing both the shape and internal layout of structural components for strength criteria using the existing ECLIPSE system. In this instance the benefits of this method over the homogenisation approach were that it could be wrapped around the existing sizing methodology without the need to further develop ECLIPSE.

Areas of structure sized thinly by ECLIPSE were considered to be originally lightly stressed and hence inefficient. Using the ESO principle that lightly stressed areas of structure are inefficient and should be removed, elements were deleted on the basis of thickness. Therefore the results of a single strength sizing loop of ECLIPSE were interrogated and the thinnest $X\%$ of the structure defined as 'removable' was deleted. This process is shown in Figure 4.1. A feasible region of structure was defined, loaded and meshed in PATRAN. The model was then strength sized using one iteration of ECLIPSE. The results of the ECLIPSE output file were then interrogated and elements sorted based upon thickness. The thinnest $X\%$ of elements were deleted, where $X$ was selected to allow smooth redistribution of load amongst remaining elements between iterations whilst removing sufficient structure for a solution to be found in a reasonable time. In addition to removing the thinnest elements it was also necessary to remove elements that, as a result of the deletion of other elements, were unconnected to the loaded structure. The resulting structure was then reanalysed in ECLIPSE and the process repeated for a fixed number of iterations.

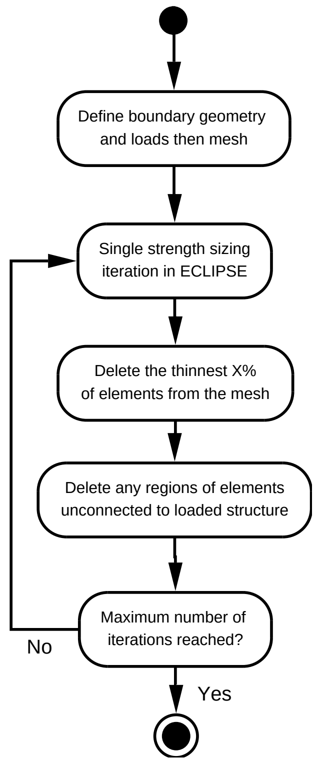



FIGURE 4.1: An ESO strength sizing process wrapped around ECLIPSE

### 4.1.1 Test Cases

Two standard test cases from the literature were used to test the method as well as a more realistic example based on a wing leading-edge rib. Figure 4.2 shows the sizing of the 'Clamped Deep Beam' problem (Bulman and Hinton (1999)) using this method. The initial structural domain is shown in Loop 0 and evolves through to the structure seen in Loop 112. Figure 4.3 shows the MBB beam problem sized after 60 iterations. Figure 4.4 shows an example problem for a wing rib leading edge described in detail in

Appendix D. This problem was created within the company and as such there was no generally accepted solution. The sizing process was stopped at loop 60 because it was not felt the solution was converging to a manufacturable solution.

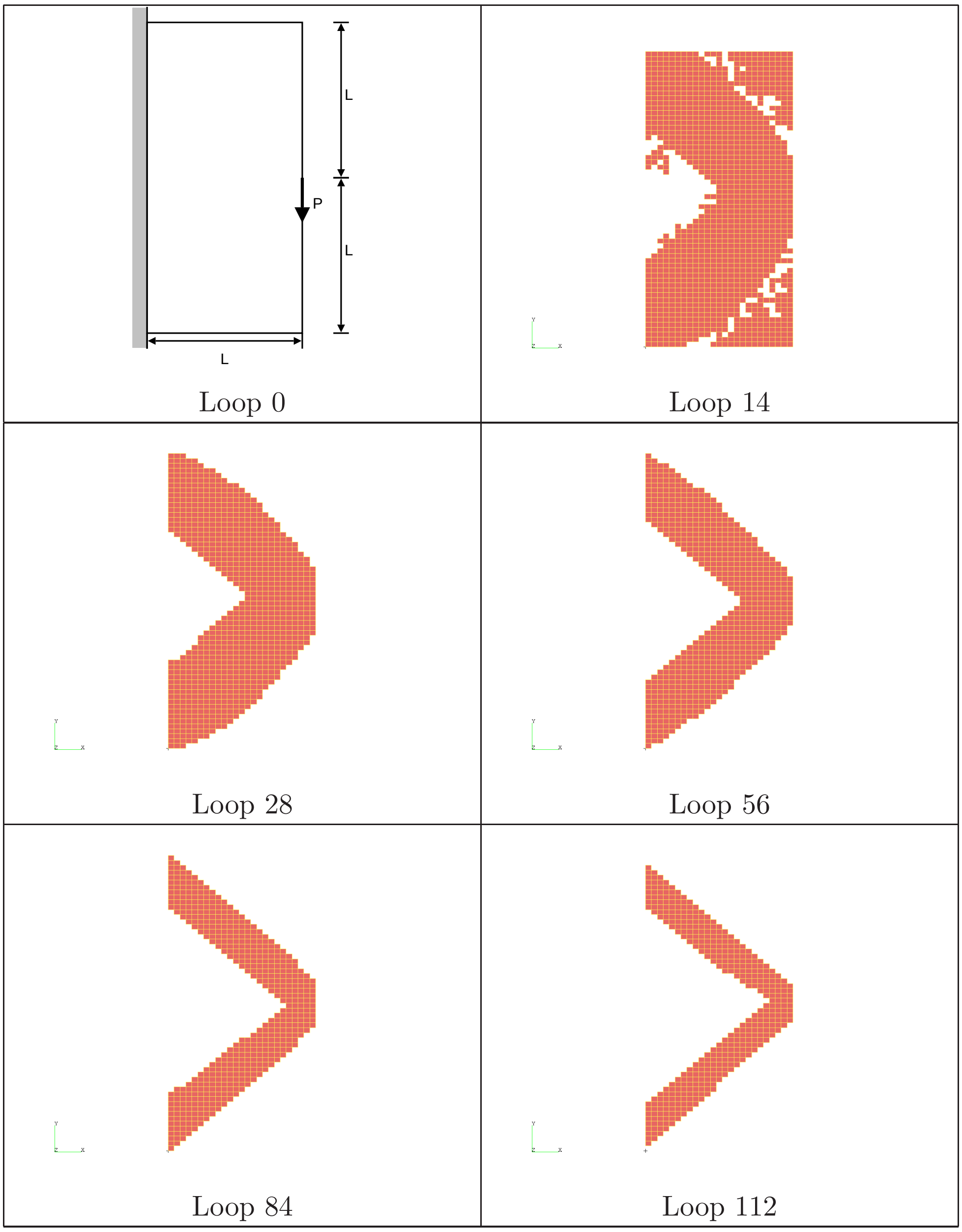



FIGURE 4.2: 'Clamped deep beam' problem sized using a 1% removal rate

### 4.1.2 Discussion

Both Figures 4.3 and 4.4 show the symptoms of 'stringiness' and 'checkerboarding' commonly associated with layout optimisation methods (Rozvany et al. (1995)). The solution found at loop 60 for the MBB beam consists of a number of fine struts between the upper and lower surfaces of the beam. This was a valid design since the stresses in the mesh did not exceed the maximum allowable. However, it would be more difficult

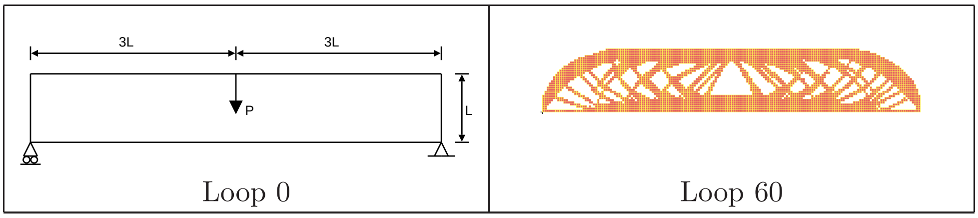



FIGURE 4.3: ‘MBB beam’ problem sized using a 1% removal rate

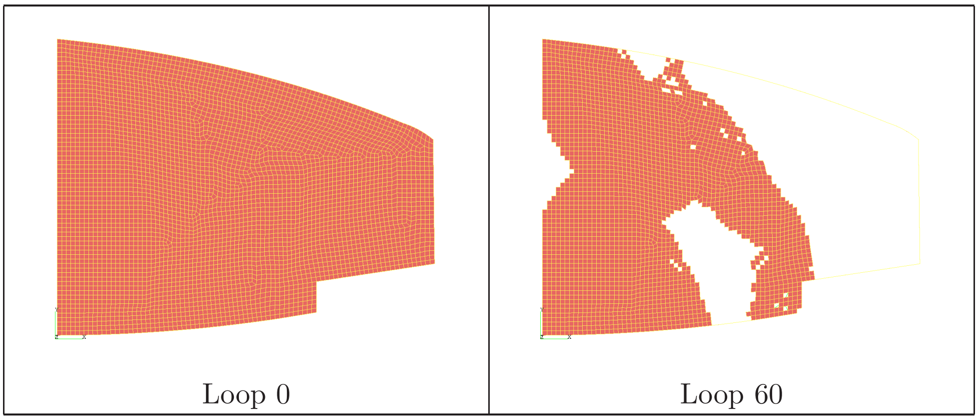



FIGURE 4.4: ‘Wing leading-edge rib’ problem sized using a 1% removal rate

to manufacture compared to a structure with fewer, wider struts. The wing rib example shows a number of areas where single elements have been removed from the structure, especially in the region towards the top of the rib. This leads to difficult to define geometry and questions about the validity of the solution since it is not clear what the structure represents.

The particular implementation of this method also exhibits a tendency to temporarily retain areas which should have been removed. Figure 4.2 shows that at loop 14 the unconstrained corners of the initial domain remain whilst the regions connecting these corners to the emerging struts are deleted. This is possibly to be due to the redistribution of load in the structure as material is sized. As material is thickened in areas neighbouring elements of low stress then this will tend to further reduce the stress in these low-stress areas. It will also decrease the stress relative to neighbouring low-stress elements. Hence when the mesh is subsequently reanalysed and sized this will create the thinnest structure at the border between the high and low-stresses. Thus the corner regions of structure survive for longer than they would otherwise do. These regions are eventually removed because they are the lowest thickness compared to the surrounding material, or become detached from the main structure and hence infeasible.

The application of this method to standard test problems such as the ‘Clamped Deep Beam’ and ‘MBB beam’ showed that it was possible to produce similar solutions to those found in the literature. However, the application of this method to a wing rib test problem showed common practical problems of topological optimisation approaches.

Whilst this sizing process is suitable for simple, well defined, problems it is clear that more complex problems need further investigation.

However, the practicalities of wrapping ECLIPSE in a suitable process becomes one of diminishing returns, since the development resources needed to interrogate and process the ECLIPSE output could be better used interfacing directly with the FE results and meshes to create a simpler solution.

In summary test cases show that this form of ESO produced similar answers to examples from the literature but that for example problems such as the wing leading-edge rib it did not converge to a manufacturable solution. Further work to improve the problem definition may have helped resolve the issues seen here. However, if a 'topological' sizing capability were to be added to ECLIPSE it would be more efficient to integrate it into the existing code to reduce the repeated post-processing of the FE model. It would also allow easier access to parameters required for analysis of constraints such as stiffness. Moreover, the optimality criterion method already within ECLIPSE already uses methods common to the homogenisation method.

## 4.2 Sizing of Laminated Composite Panels

This section describes an investigation into the reuse of the panel analysis capability within 'CITS' within two different panel sizing processes. The code was first linked to the gradient based optimisation package 'DOT' (Vanderplaats (1999a)) within the ModelCenter PSE to perform a fixed stacking sequence (FSS) sizing as discussed in Section 3.2.2. Secondly the code was tightly coupled to a genetic algorithm to examine the variable stacking sequence (VSS) sizing formulation discussed in Section 3.2.2. The results are compared against the company's existing fixed stacking sequence sizing method within ECLIPSE and an example from the literature. To put this work in context, at the time it was carried out the ADG group were interested in understanding the merits of sizing using both methods. The solutions from each process were to be compared in terms of mass, computational expense and reliability.

Figure 4.5 shows problem definition for the panel. End loads $N_X$, $N_Y$ and a shear force $q$ are applied to edges as shown. For all sizing methods the problem was enforced to be balanced (for every $+45°$ ply a $-45°$ was placed together with it) and symmetrical as shown in Figure 4.6. The material properties used were the same as those used by Le Riche and Haftka (1993): $E1 = 127.59 * 10^{10} N/m^2$, $E2 = 13.03 * 10^9 N/m^2$, $G12 = 6.41 * 10^9 N/m^2$, $\nu_{12} = 0.3$, $t_{ply} = 0.000127m$, $\rho = 1630\frac{kg}{m^3}$, $\epsilon_{1_{ua}} = 0.008$, $\epsilon_{2_{ua}} = 0.029$, and $\gamma_{12_{ua}} = 0.015$.

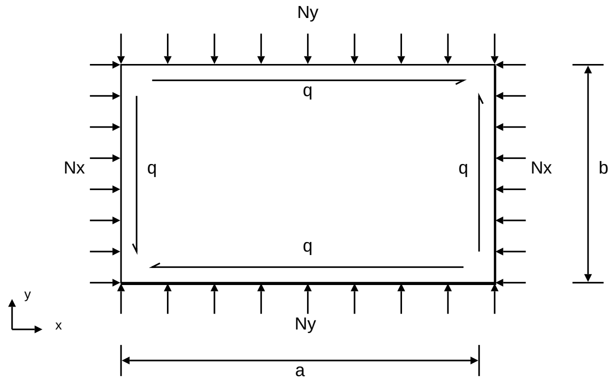



FIGURE 4.5: Wing skin panel geometry and loading conditions

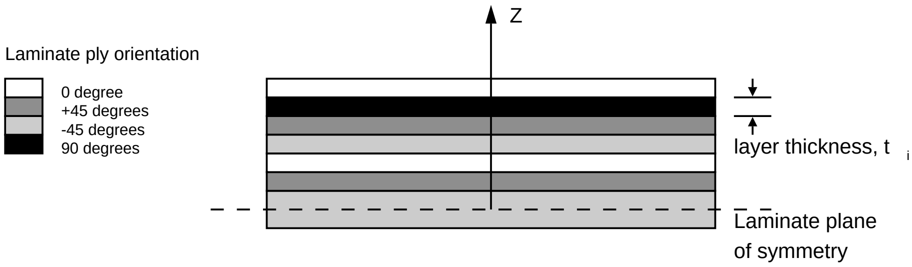



FIGURE 4.6: Composite stacking sequence

### 4.2.1 Fixed stacking sequence (FSS) sizing methodology

FSS layups were represented using a sequence of twelve plies which were then reflected about the line of symmetry as shown in Figure 4.6. This sequence was $(0/+45/-45/90)$ repeated three times, the thickness of each layer being determined by a variable $t_i$. The thicknesses of sets of $+45$ deg and $-45$ deg plies were linked by using one variable for each set of these plies. Thus the 12 ply layup was represented by 9 variables.

Figure 4.7 shows the arrangement within ModelCenter for the case where CITS was linked to DOT. The AD group developed a COM interface around CITS to allow communication with ModelCenter through Visual Basic Script. Resources were only available to develop the COM interface for the panel stability method so tests were restricted to compression loads where panel stability would be the critical constraint rather than strength. The CITS wrapper for this method ('CITS_Panel') was generated as a ModelCenter Visual Basic 'ScriptWrapper' component with the layup, panel geometry, loads and materials as inputs. The wrapper code constructed the necessary laminate within the CITS object which then returned the panel mass and load factor at which buckling would occur. Of the three DOT optimisers the 'Method of Feasible Directions' method found the minimum mass answers for this problem based on default settings.

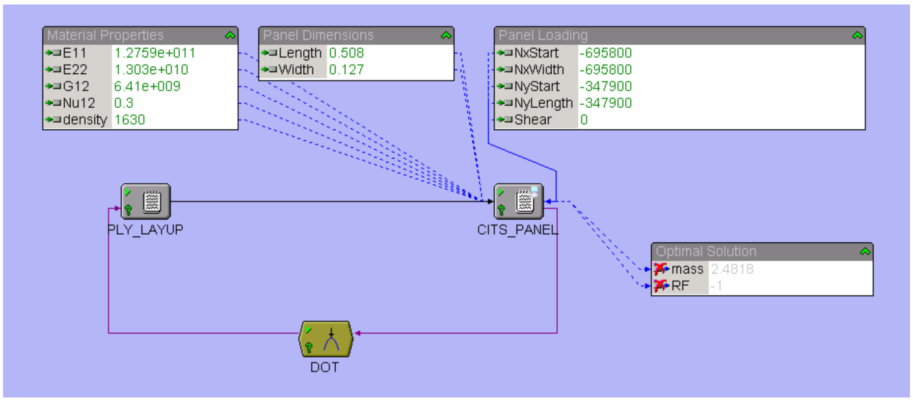



Figure 4.7: CITS linked to DOT within ModelCenter to size laminates using a fixed stacking sequence

### 4.2.2 Variable stacking sequence (VSS) sizing methodology

The variable stacking sequence problem was addressed using a methodology based on those of Le Riche and Haftka (1993), Kogiso et al. (1993), Soremekun et al. (2002). A genetic algorithm was used to evolve a stacking sequence to minimise the mass of the laminate whilst meeting the design criteria (panel strength, stability and ply blocking). Sourcing of a ready-made GA proved difficult. Despite there being two other GAs developed within the company, neither was available for this application. The first was developed as a test-bed for optimisers but did not yet work for discrete variable problems. The second was developed as part of a package for scheduling of aircraft maintenance. It was developed for, but subsequently not delivered to, an internal customer. Attempts to use a copy of this code were blocked because it was 'owned' by another department who were not prepared to share it. In addition it was not possible to use 'open-source' code because of internal company prohibitions surrounding its use in possible future production code (see Section 2.4.1). There was a trade-off here between non-technical issues and development of a solution to solving the problem and a basic GA was developed based on the methods given in Goldberg (1989) and Papalambros and Wilde (2000) and described later in this section. The results of this were fed back to the ADG team in terms of a generic piece of source code and a detailed report for implementation at a later date. It should also be noted that this work was carried out when the CITS panel stability method had not yet been wrapped with a COM interface and so the CITS C++ library was integrated with the GA code, the resulting application being called 'LAS'. These issues illustrate some of the difficulties of process development and installation within a large organisation, more of which are discussed later.

Figure 4.8 shows the arrangement of modules within LAS. Using the method of Soremekun, but restricting its use to a single panel, the GA (`CGA_LAS`) was linked to a laminate 'assembly' (`CGA_Assembly`) which was a collection of laminated panels (`CLPanel`). Each panel was linked to the laminate panel strength and stability analysis classes `CCNS_FTPanel_Strength()` and `CCNS_FTPanel_Stability()` respectively. The CITS

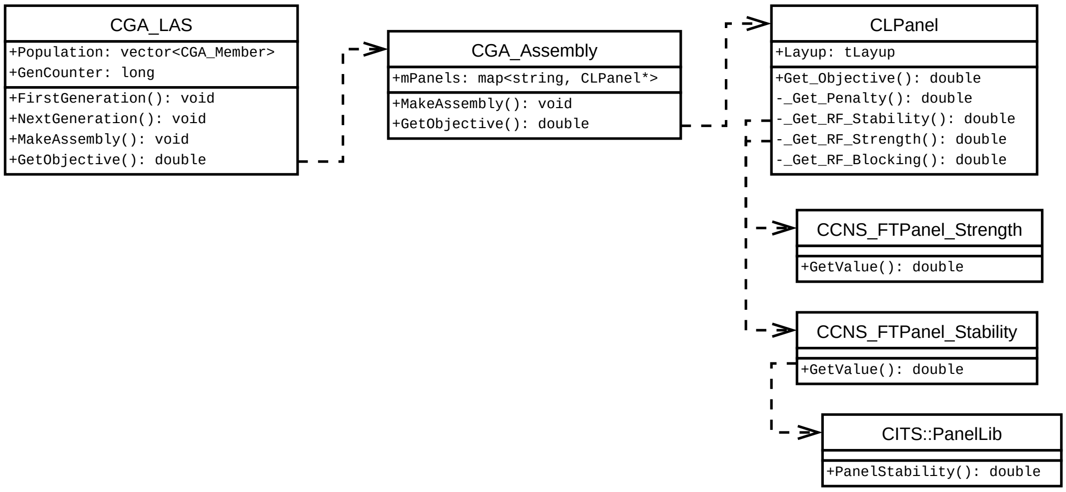



FIGURE 4.8: Arrangement of classes within LAS

panel stability calculation was reused by linking functions from the CITS panel stability library to the code.

The layup in each panel was defined using design variables which refer to sections of the genotype, allowing panels in the collection to share common sections of layup. This multiple panel approach was taken to allow possible future ply blending. Each allele was encoded as a list of materials, in this case ranging from an empty list through to a list with two plies: 0 - no plies; 1 - 0° ply; 2-±45°; 3-90°. The phenotype was constructed by adding together the lists in the sequence given by the genotype. Thus '0123' would give a total layup of $0°/+45°/-45°/90°/90°/+45°/-45°/0°$.

### Operation of the GA

Generic GAs rank their population based on the fitness measured using the objective function. To represent constraints it is necessary to include objectives which guide the search in the direction of satisfied constraints, whilst searching the underlying objective function. These objectives are described here as 'penalty' and 'bonus' functions and essentially turn the problem into a multiobjective optimisation problem. Penalty and bonus functions are weighted against the underlying objective function to ensure the solution found is both valid and optimal.

Penalty functions were added to the basic objective function in order to distinguish between designs that violated constraints. This is shown in Figure 4.9(L), where it is clear that there is no change in the value of the function when solutions lie in the infeasible design space. The penalty function used to address strength and stability criteria contained two components, the first being a step penalty to clearly distinguish between valid and invalid designs. Shown in Figure 4.9, solutions which violated one or more design criteria were penalised by the addition of a $10^{10}$kg penalty. This value was chosen to create a new objective function within which solutions could be easily identified as feasible or infeasible by the GA, since a real-world solution would not have this mass. Whilst it would be possible to find a minimum mass solution using this function it is inefficient for two reasons. First, it does not differentiate between solutions of equal mass but different strengths and stiffnesses. This would be useful when selecting members of the next generation because it allows the GA to converge to solutions which have the better properties, increasing the possibility that this will allow further layers to be removed and the laminate lightened. Secondly, if the initial population existed solely within the infeasible design space, then it was possible that the GA will converge to a population of very low mass, but infeasible designs, effectively becoming trapped in the infeasible space. The second component penalised designs in proportion to the amount with which they violated the design criteria. This created an objective function that encouraged an initial population of infeasible designs to converge towards more feasible designs:

$$P_i = 10^9 + \frac{1}{(\lambda_{achieved})} \tag{4.1}$$

A similar approach was used to differentiate between designs of the same mass but different constraint values. A bonus value was subtracted from the mass of the solution such that a solution of equal mass, but high buckling load factor would have a slightly lower mass than one with the same mass but a lower buckling load factor. It was found in initial trials that typical improvements in laminate strength were larger than the typical improvements in the stability load factor. For example one solution might have a $\lambda_{strength} = 7.0$ and $\lambda_{stability} = 1.0$, whereas another might have $\lambda_{strength} = 2.0$ and $\lambda_{stability} = 1.4$. A linear bonus function would give preference to the solution with higher strength, whereas a lighter mass solution is more likely to be found by removing a ply from the second solution with constraints that both exceed the target value. Hence a non-linear bonus function was used to encourage equal growth in both stability and strength properties. To achieve this

$$B_i = -0.001 * \mathrm{m} * \left(1 - \frac{1}{(\lambda_{achieved} - \lambda_{target}) + 1}\right) \tag{4.2}$$

This also limited the bonus parameter that could be applied, since $B_i \rightarrow 1.0$ as $\lambda_{achieved} \rightarrow \infty$. The ply blocking constraint was set to add an additional fixed contribution of $10^9$ to the penalty function if it was violated.

Thus the modified objective function was

$$f(x) = M + B + P \tag{4.3}$$

where $M$ was the mass of the panel, $B$ was the total bonus value for the panel solution and $P$ was the total penalty value.

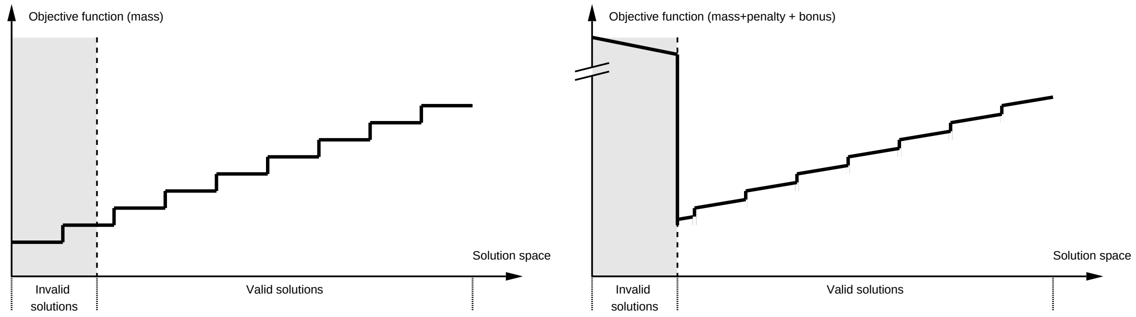



FIGURE 4.9: Unconstrained objective function (mass of laminate)(L), and modified objective function with constraint penalty and bonus functions (R)

Three operators were used: a mutation operator was used to maintain diversity within the population by changing a random gene to a random allele; crossover was applied to create offspring based on a random splice of one parent's genes with another, to create two new members; a permutation operator was used to reverse the genetic sequence between two random points in the chromosome (Le Riche and Haftka (1993)). Whilst similar to the often used 'inversion' operation, 'inversion' also transfers the meaning of the gene at that location, whilst permutation exchanges just the value of that gene. The practical effect of this operator is to vary the stacking sequence, whilst maintaining the same amounts of each ply orientation.

Once evaluated members of the parent and offspring populations were selected for the next generation using a roulette wheel approach. Both populations were combined, ranked, and a portion of the space [0,1] allocated to each member based on the rank given to it. The proportion of the wheel was allocated using the linear normalisation method (Le Riche and Haftka (1993)) which varies linearly with the rank of that member in the group (Equations 4.4 and 4.5). A limited amount of elitist selection was also used, with the best member from each generation was automatically carried forward into the next. The probability that solution $i$ was selected, $P(i)$, was given by:

$$P(i) = A + (m - i)B \tag{4.4}$$

where if $A = B$ then

$$P(i) = (m - i + 1)B \tag{4.5}$$

where $\sum_{i=m} P(i) = 1$, $A$ and $B$ are constants, $i$ is the rank of the member and $m$ is the total number of members in a population. $A$ is the probability of selection of the least fit member and $B$ is the difference between two members ranked next to each other. In this case $A$ was set equal to $B$.

### GA parameter settings

Some limited testing was performed to determine effective parameter settings for this GA. Similar GA settings to Le Riche and Haftka (1993) for the probability of different operations (P(Mutation) set to 0.03, P(Permutation) set to 0.8 and P(Crossover) set to 1.0), the number of generations without improvement, and the size of the population, were varied about values found through experience from initial testing. During initial testing it became apparent that lower mass answers were found more routinely for small GA populations, of around 6 members, run for a large number of generations. The chromosome length was fixed at 40 variables since this provided an adequate range of plies for the problems being studied. Ideally a full study of the GA parameters would have been conducted, however this was not possible within the time available for this work.

Pre-empting work discussed later in Chapter 5, the free body diagrams from panels being sized in the rectangular wingbox were used as test problems for this work. These are given in Appendix E. This allowed a range of different loading conditions and panels sizes to be sized by LAS. By normalising the masses of the solutions produced, by dividing the solution mass by the best found overall, it was possible to examine which settings were most likely to find the best solution for a given panel. However, it should be noted that these settings would not necessarily be suitable for sizing a whole structure since there may be settings which more effectively size the heaviest panels to create a lower overall mass for the structure. In the case of lightly loaded structures the GA will search within an over-dimensionalised design space, since it will require a low number of plies (most of the solution will be empty plies), whereas for the heavily loaded case the GA will need to search the wider design space to find suitable solutions.

For each test point the set of 33 panels given in Appendix E was sized 20 times and the results recorded. Figure 4.10 shows the performance of the GA as the number of generations without improvement is increased for a population size of 6. Figure 4.11 shows the effect of varying the GA population size on the normalised mass of the panels for a total number of generations without improvement of 2000.

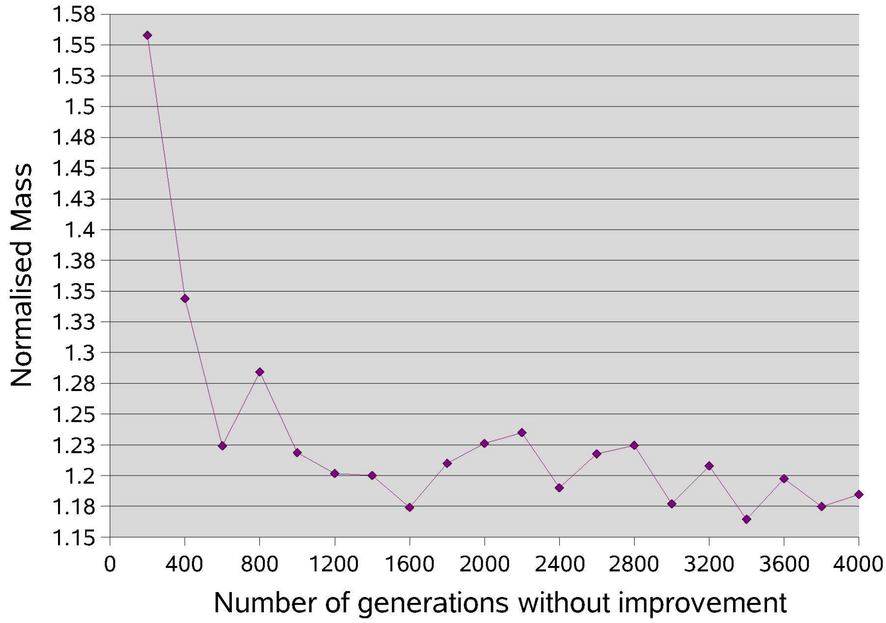



FIGURE 4.10: Effect of varying the GA stopping criteria on the average normalised mass for a test set of problems (GA population size of 6)

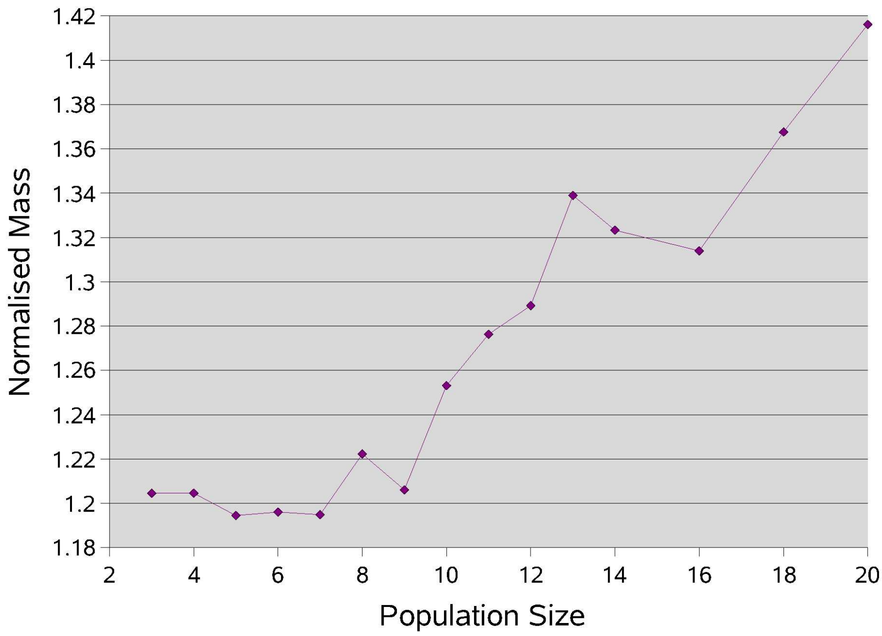



FIGURE 4.11: Effect of varying the GA population size (stopping criteria of a maximum of 2000 generations without improvement)

### 4.2.3 Testing

This section describes a comparison of the DOT/CITS and LAS/CITS panel sizing methodologies with the ECLIPSE laminate panel sizer for the set of test problems studied by Le Riche and Haftka (1993) shown in Table 4.1. Since LAS/CITS was a stochastic process each test case was run 200 times to study its reliability. A population size of 6 was used and the maximum number of generations was limited to 2000. The run time per panel sizing with these settings is given in Table 4.2. This is based on a CITS panel finite strip mesh of 4 x 4 strips, using a compiled version of LAS with default Microsoft Visual C++ 'Release' compilation settings on a 500Mhz processor. The ECLIPSE panel sizing was tested using a simple FE model of the panel constrained to prevent out-of-plane displacements at the edges. Rigid body motion in-plane was prevented by supporting the mid-nodes of the panel edges. One iteration of the ECLIPSE sequential linear programming (SLP) search method was performed for each test case. Default parameters were used for both the DOT/CITS and ECLIPSE optimisers.

| **Description** | **Length a(m)** | **Width b(m)** | **Nx (N/m)** | **Ny (N/m)** | **q (N/m)** | Plies |
|---|---|---|---|---|---|---|
| 1 | 0.508 | 0.127 | $1.751 * 10^6$ | $8.755 * 10^5$ | 0.0 | 48 |
| 2 | 0.508 | 0.254 | $6.958 * 10^5$ | $6.958 * 10^5$ | 0.0 | 64 |
| 3 | 0.508 | 0.127 | $2.367 * 10^6$ | $2.959 * 10^5$ | 0.0 | 48 |

Table 4.1: Geometry and loads used in test cases

| **Coupling Approach** | **Code** | **Number of Evaluations** | **Total time (secs)** | **Evaluation and communication (secs)** |
|---|---|---|---|---|
| *Tight* | LAS | 30896 | 815 | 0.03 |
| *Loose* | DOT/CITS | 550 | 550 | 1.00 |

Table 4.2: Time per evaluation using tightly and loosely coupled integration based on time to size test case 2

#### Performance of the different panel sizing methods

Table 4.3 shows the results for the deterministic sizers ECLIPSE and DOT/CITS, together with the target results for the problem found by Le Riche and Haftka (1993). Table 4.4 shows the results obtained by LAS/CITS, together with the frequency with which they occurred during testing. The maximum difference between the target solution masses and those produced by ECLIPSE was 7.8% (test case 1) and 9.0% for DOT/CITS and test case 3. The maximum difference between the target and solution masses for LAS/CITS was 6.2%, found in 1% of the attempts at test case 2. Figures 4.12, 4.13 and 4.14 show the solutions obtained from the ECLIPSE, DOT/CITS and a solution picked at random from LAS/CITS. Each solution is distinctly different, indicating the range of local optima within the design space, although there is some broad agreement in the magnitude of $\pm 45$ deg fibres used in each solution.

| Testcase | Method | Mass (kg) |
|---|---|---|
| 1 | *TARGET* | 0.641 |
| | DOT/CITS | 0.668 |
| | ECLIPSE | 0.691 |
| 2 | *TARGET* | 1.709 |
| | DOT/CITS | 1.739 |
| | ECLIPSE | 1.709 |
| 3 | *TARGET* | 0.641 |
| | DOT/CITS | 0.699 |
| | ECLIPSE | 0.679 |

TABLE 4.3: Deterministic methods of panel sizing optimisation processes

| Testcase | Method | Mass (kg) | Reliability (%) |
|---|---|---|---|
| 1 | LAS/CITS | 0.641 | 19.0 |
| | | 0.677 | 81.0 |
| 2 | LAS/CITS | 1.769 | 99.0 |
| | | 1.816 | 1.0 |
| 3 | LAS/CITS | 0.641 | 100.0 |

TABLE 4.4: Reliability of different methods of panel sizing optimisation processes (Reliability calculated using results of 200 searches using LAS)

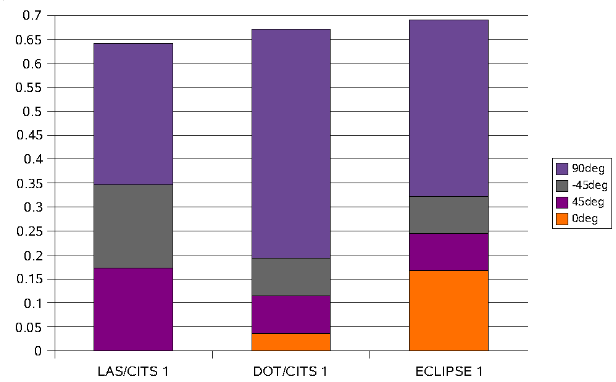



FIGURE 4.12: Breakdown of the solutions for test case 1 by ply orientation mass

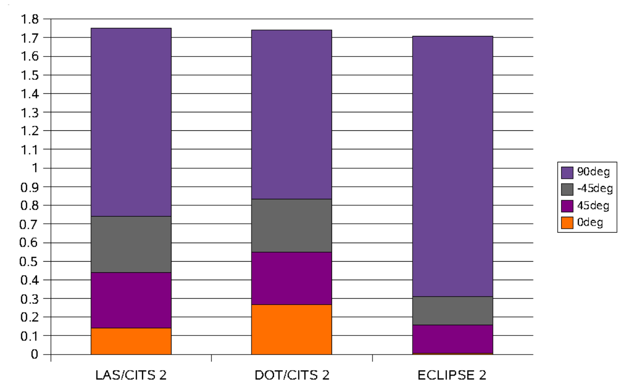



FIGURE 4.13: Breakdown of the solutions for test case 2 by ply orientation mass

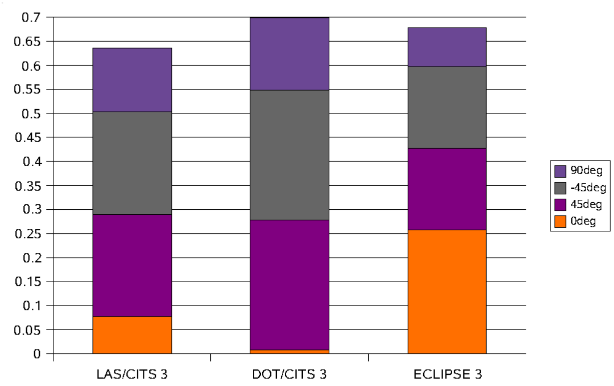



FIGURE 4.14: Breakdown of the solutions for test case 3 by ply orientation mass

### 4.2.4 Discussion

**Performance of the different integration methods**

Both closely coupled and loosely coupled integration approaches were used to link the optimiser with the analysis function. Table 4.2 shows that the time to communicate with, and evaluate, one request to the loosely coupled function was a factor of 33 greater than that for the tightly coupled function. This was because the optimisation method and analysis code were linked together within the same executable code, allowing information to be exchanged between processes on the same machine. The loosely coupled component exchanged information through interfaces to components on different machines. This indicates that it is currently inefficient to conduct searches of a design space described by a computationally inexpensive function, since the majority of the elapsed time for a search is spent transferring information rather than performing the search itself. However, the loosely coupled approach is more convenient for rapidly reconfiguring, and replacing, the search and analysis methods.

**Integration of existing company capability into sizing processes**

Although limited in complexity compared to later systems studied, using ModelCenter to integrate CITS into a sizing process highlighted a number of issues that will need to be resolved if PSEs are to be used in a product development environment. They relate to maintenance and development of a process which is significantly more complex than a typical script because the flow of information is 'hidden' behind an interface that must be interrogated to extract information about the process. The process shown in Figure 4.7 was developed by manually connecting to the remote machine where the `CITS_PANEL` script was stored, editing the script and placing 'pop-up' error messages in the script which would then appear on the machine running ModelCenter. At the same time the `PLY_LAYUP` script was stored and accessed through ModelCenter. This complicated the development by separating the declaration of variables from the main script. This caused additional complexity, and itself introduced errors, into the development process, a problem reported as a result of this work and corrected in later versions of the PSE. A lesson learnt from this issue was to wrap a code in only one, multifunctional, script. This simplifies the development process and allows reuse of mature wrappers. An example is discussed in more detail in Chapter 5.

Second it was not possible to fully debug the panel sizing process within the PSE. The various layers of abstraction restricted the users ability to trace the flow of information through the process. Information which could be extracted through the CITS COM interface could be collected through the wrapper. However, information within the CITS executable was inaccessible within ModelCenter. This meant the process had to be debugged up to the limit of information in the wrapper, and the calculation replicated in a local version of CITS. For future integration it would be useful to be able to save a calculation within the component to enable it to be analysed in an interactive version of the code later.

Thirdly, the lack of a central repository for component wrappers was a common problem during process development. This meant that wrappers were stored on the local AnalysisServers and managed locally. Often AnalysisServers were not available, which lead to difficulties since different versions of wrappers would be located on different machines. It also meant it was difficult to routinely back-up the wrappers since they were located on many different machines which could not be directly accessed from the main terminal. In part this was overcome by creating a ModelCenter model specifically to copy files from remote machines based on a list of files stored in an Excel spreadsheet. However the real solution would be to map a drive from each AnalysisServer to a centrally maintained repository. Overall the environment in which the process is run needs to be mature and use well known, systematic, procedures for maintaining and storing component wrappers.

# Chapter 5

# Strength Based Sizing of Airframes

The discussion in Section 3.4 noted the limitations of individual optimisation methods available for sizing whole airframes. No one method currently offers the ability to size structures for the desired range of design criteria and manufacturing methods within the limits of available resources. This is why existing sizing codes use more than one optimisation method when sizing structures for strength, stability and stiffness criteria. The mixture of optimisation methods in current generation of sizers such as ECLIPSE result in hardcoded, monolithic, applications which require knowledge of the system and its source code in order to adapt the sizing process.

Figure 5.1 shows a proposal for a multilevel sizing system, whereby the FE model is sized for global stiffness, and local features, such as wing panels, are sized for criteria, such as strength and stability, using local sizing methods. The difference between this system and existing proprietary codes is that the architecture is more open and the multilevel approach more explicit. The aim is to allow a company to have a sizing process which can be adapted to meet the needs of the current project. The different sizing components are integrated within a problem solving environment to allow process engineers the ability to pick the best mix of methods based on technical, and process management, criteria. This, for example, would allow the company to maintain a standard set of interfaces between its CAD system, FE package and sizer, but rapidly reconfigure the sizer for the particular needs of a specific project (e.g. it replaced the standard stiffness sizing routines with a routine for specific aeroelastic criteria).

The aim of this work was to understand the issues in developing the proposed process within the current toolset. Specific issues to be addressed were the performance of the process compared to existing processes; the potential for reuse of existing and COTS capability; and process integration within the current and likely future toolset. This

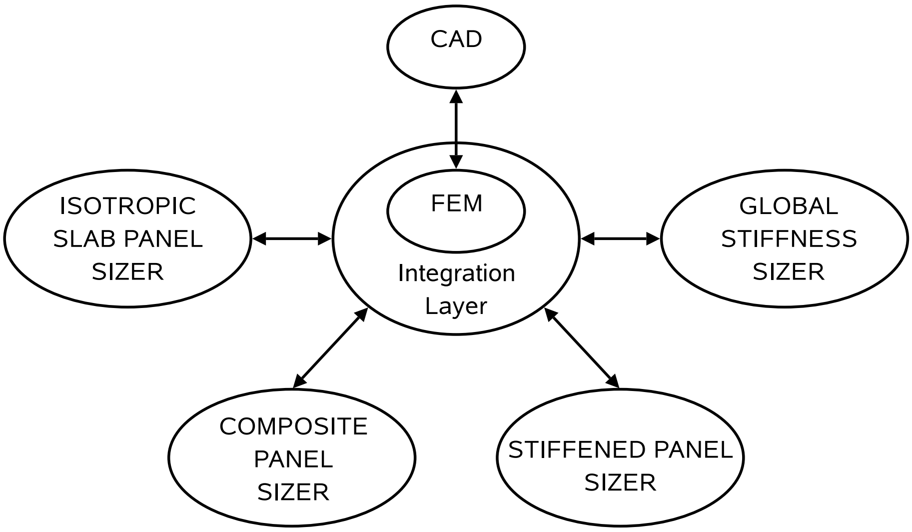



FIGURE 5.1: Proposed Future Sizing System

chapter describes the design (Section 5.1), sizing methodology (Section 5.2), implementation (Section 5.3) and testing (Section 5.4) of a system for structurally sizing aircraft using a free body diagram approach as described in Section 3.2.

## 5.1 System Design

Preliminary system design was based on a set of process 'use cases' for a project engineer (an employee developing a new vehicle) and a process engineer (the person responsible for maintaining the process). Figure 5.2.a shows the stages in the process that a project engineer would follow to size a vehicle based on a concept vehicle layout and a set of loading conditions. It is assumed that the sizer is part of a larger vehicle design process which determines the layout and loading of the structure. The purpose of the sizer is to determine the optimal material distribution within that structural configuration. It is assumed, as is usual practice in industry, that the most extreme loadcases which the aircraft will see in service have been captured and will be used in the sizing process (as described in Section 2.3.3). The stages briefly are:

1. **Geometry definition** - This would be performed in a geometry definition package such as CATIA V5 and would include internal structure such as wing ribs and spars;

2. **Identification of features** - For more automated process features, such as skin panels, need to be defined so that the sizer can idealise models of these from the FE analysis. In the current sizing process this is conducted manually after the FE model has been defined (as discussed in Section 3.2) If it were possible to link the

geometry and FE mesh then potentially the links between features and the mesh could be automated here;

3. **Structural model idealisation** - If not performed in step (2) then the engineer needs to create a FE model from the geometrical definition and identify areas of this model as features to size;

4. **Structural sizing** - This is effectively a black box function used by the engineer to size the structure. Although the engineer can specify materials and constraints they do not necessarily need to know how the sizing process operates;

5. **Post-processing** The sized structure needs to be reviewed to ensure all constraints, both explicit and implicit, have been satisfied. For the test cases here this included a linear-static eigenvalue buckling analysis of the sized structure to determine the overall buckling load factor of the structure.

A process engineer is responsible for maintaining and developing the sizing capability. Figure 5.2.b shows the operations that a process engineer would use to maintain and update the process. Tasks might include:

1. Ability to 'wrap' and install components on component servers;
2. Ability to modify the process by changing the sequence or components;
3. Ability to integrate the process easily into an MDO process;
4. Ability to test the process

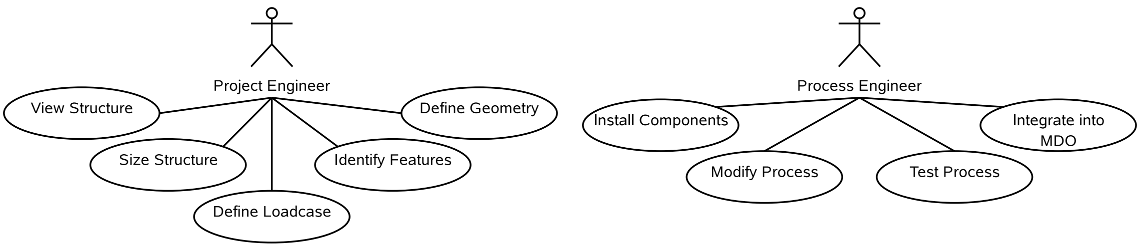



FIGURE 5.2: Sizing system use cases for (a) a project engineer(L) and (b) a process engineer(R)

From these use cases a design was developed for a sizing system using the free body diagram approach discussed in Section 3.2. The system was designed with the goal of utilising components that already existed (e.g. NASTRAN, CITS panel calculations) and for which there are potential commercial alternatives (e.g. Hypersizer), although the demonstration system design is a compromise based on the availability of the current toolset for testing. Figure 5.3 shows the overall sizing process, grouped into pre-sizing, sizing, and post-sizing stages. Pre-sizing is the stage at which the geometry is defined,

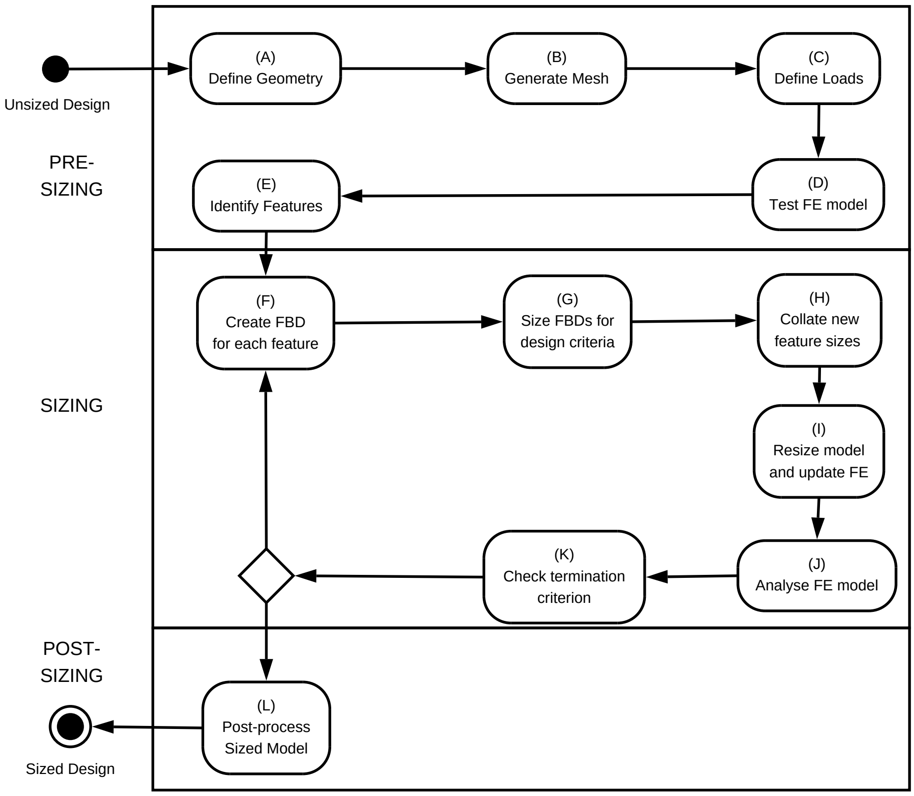



FIGURE 5.3: Structural sizing process based on the project engineer use cases

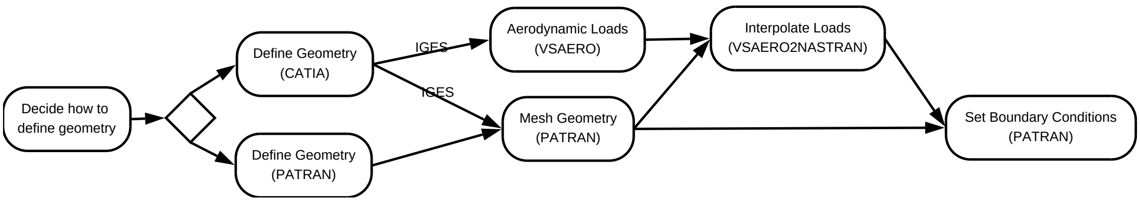



FIGURE 5.4: Pre-sizing stage: Structural model generation

a structural model created and loaded. It is also the stage at which 'features' of the structure are identified for sizing. Figure 5.4 shows the process used to generate the test models. Specific details are given in Section 5.4, but for this demonstration system it was necessary to allow geometry and loads to be created in PATRAN as well as through CATIAV5 and VSAERO[1].

It should be noted that a distinction was made between the different types of information being processed within the system. Three levels of data flow correspond both to the volume of data and degree of control of the process. The 'top-layer' of information specifies the model being sized and process parameters such as the number of iterations to be conducted. The 'middle-layer' of information defines structural features and the sizing criteria. The 'lowest-layer' is the data from the global FE model and loadcases.

[1]CFD package used to generate aerodynamic pressure distribution over the surfaces of the geometry

The sizing stage starts with the decomposition of the global model into models of local features such as wing skin panels. Free body diagrams (FBDs) are created for every loadcase being sized. To create the FBDs it was assumed that the elements in the FE model would be related to 'features' using a unique property card for each feature. The property cards would be manually assigned at the geometry creation and meshing stage, either through CATIAV5, or PATRAN. Figure 5.5 shows the relationship between these entities. Each feature will have a number of constraints which it must satisfy for a given loadcase. These include strength, buckling and stiffness criteria. In this system a constraint is not unique to a given panel and can be referenced by many features simultaneously. An example constraint is that the panels must not buckle at less than 1.5 times the load the panel experiences in a given load case. Appendix F shows an example constraint definition file linking a feature with a set of constraints.

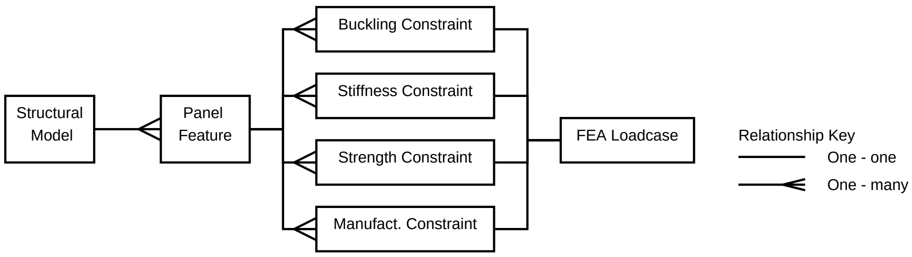



FIGURE 5.5: Entity relationship diagram for AFS

Each free body diagram is then sized such that it satisfies the constraint it was derived from. This is a process that could be performed in parallel since no interaction takes place between features at this stage. The results of these analyses are collated and every feature sized using a set of rules to determine the critical size from the constraints imposed upon it. For local sizing of metallic panels the largest thickness was found for both the strength and stability criteria, and the largest value used. Maximum and minimum thickness limits were then applied such that the final panel thickness met manufacturing constraints. The FE model was then updated using these new sizes and reanalysed. The sizing process was repeated for a fixed number of iterations.

## 5.2 Feature Identification Methodology

It was assumed that the main type of features being sized in this system were structural panels, a four sided object approximated by a rectangular panel free body diagram constructed from specific nodes and elements in the FE mesh. It was also assumed that the overall sizing process should be automated wherever possible and a method is described here for identifying panels, detecting edges and approximating the panels as free body diagrams.

To automatically construct a FBD it was necessary to relate nodes and elements in the model to a feature and determine how to extract loads from the nodes and combine them to create a panel loading. Figure 5.6 shows the relationship between property cards, elements and nodes. Assuming a unique property card is assigned to each feature, and all the elements representing that feature in the FE model use that property card, then it is possible to identify all elements and nodes associated with a feature through their relationship with the property card.

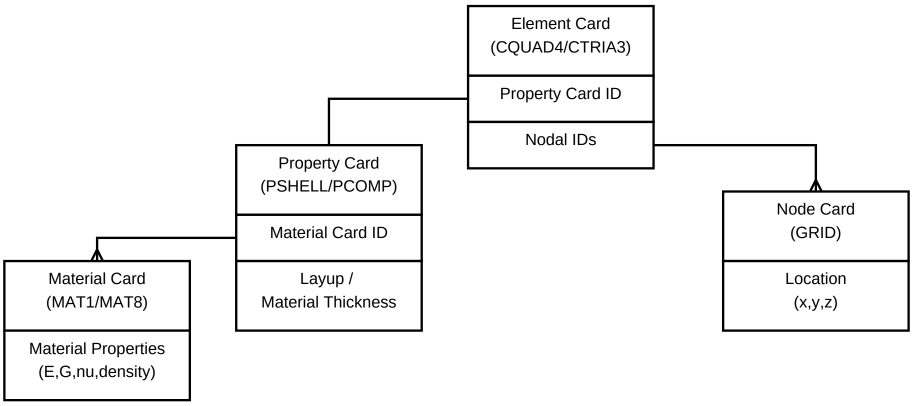



FIGURE 5.6: Relationship between FE model entities

Loads and geometry were calculated by assuming edges on the panels could be identified automatically. To achieve this it was necessary to identify nodes on the panel boundary and then identify edges. Boundary nodes were identified by looking at the connectivity of nodes in a feature. Each node was connected to another node in the model by association with at least one element. If any nodal pair was related to only one element within the feature it was considered to be on the panel edge. Angles were then calculated between the vectors of each nodal pair on the boundary using the relationship given in Equation 5.1 (Kreyszig (1993)). Nodes with the four largest values of $\gamma$ were then assumed to be corners of the panel:

$$\cos \gamma_{AB} = \frac{\vec{A} \cdot \vec{B}}{|\vec{A}||\vec{B}|} \tag{5.1}$$

where $A$ is the vector between nodes 1 and 2 and $B$ is the vector between nodes 2 and 3 as shown in Figure 5.7. Panel edge loads were calculated using the 'Grid Point Force Balance' output from NASTRAN rather than the element stress output method as used by Hypersizer (Collier Research Corporation (2003)). This method was chosen to potentially allow other types of free-body diagram to be created at a later time. For standard rectangular panels, loads were calculated acting into and along the edges of the panel as shown in Figure 5.7.b. FE grid point force balance loads were converted from the NASTRAN 'basic' (global) coordinate system into the local panel edge coordinate system as defined by the vector between the first and last points on the edge. Loads ($N_{alpha}$ and $N_{beta}$) were calculated by summing the nodal forces along an edge and creating an average of the loads on opposite edges to get average panel end loads and shear. The total panel shear force was found by averaging shear forces along all panel edges. The free body diagram geometry was approximated by taking the average length of opposing sides of the panel. The length of a side was calculated as the distance between corner points. e.g. the distance between nodes 1 and 5 for edge A, $|\vec{N_1 N_5}|$.

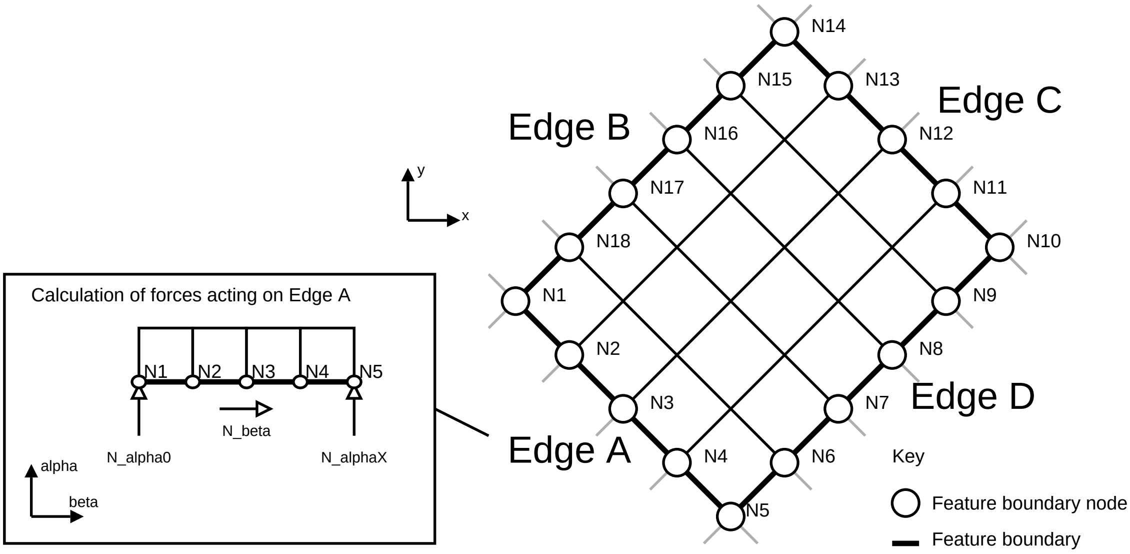



FIGURE 5.7: Free body diagram generation process

## 5.3 Implementation

### 5.3.1 AFS Code

A library of C++ objects collectively known as 'AFS' (AirFrame Sizer) was developed as part of this work. AFS contained routines for: interfacing with NASTRAN FEM and results files; generation of panel free-body-diagrams from FEM nodal force data; laminate panel strength and stability calculations; genetic-algorithm, stress-ratio and optimality criterion search methods for structural optimisation. In addition a 'COM' interface wrapper was developed for AFS by the Structural Computing Analysis Development Group (ADG) within BAE Systems. This allowed individual routines to be accessed by ModelCenter when AFS was run on a machine together with an AnalysisServer. The AFS FE model manipulation routines were designed to use techniques similar to methods developed, but currently not completed, by the ADG. The ADG methods were not used because the developer who had written them was seconded to another project. Figure 5.8 shows the arrangement of classes implemented in AFS. The `CStruct_Model` class brought together overall control of the AFS routines and was used as the main interface to the routines. It contained the FE model being sized, stored in an instance of the `CFEM_Model` class. The results of the FE analyses were stored in an instance of the `CFEM_Results` class, which contained an instance of `CFE_Results_Subcase` for each load case being studied. Structural features and associated FBDs were stored in a C++

'map' container, which provided a named index of pointers to structural features, with the generic `CFT_Generic` base class. `CStruct_Model` also contained a similar map of all user specified constraints that features could be sized against (`CCNS_Generic`).

NASTRAN bulk data (.BDF) and standard output (.F06) FEM files were used to pass FE model information between NASTRAN and the classes within AFS. Both are ASCII files and their use is considerably slower than interfacing with alternative formats such as the NASTRAN XDB binary file formats. However these methods were easier to implement and test. The `CFE_Results_Subcase` class included a feature to extract only grid point force data for nodes associated with a feature boundary to reduce the overhead of data being parsed by the system.

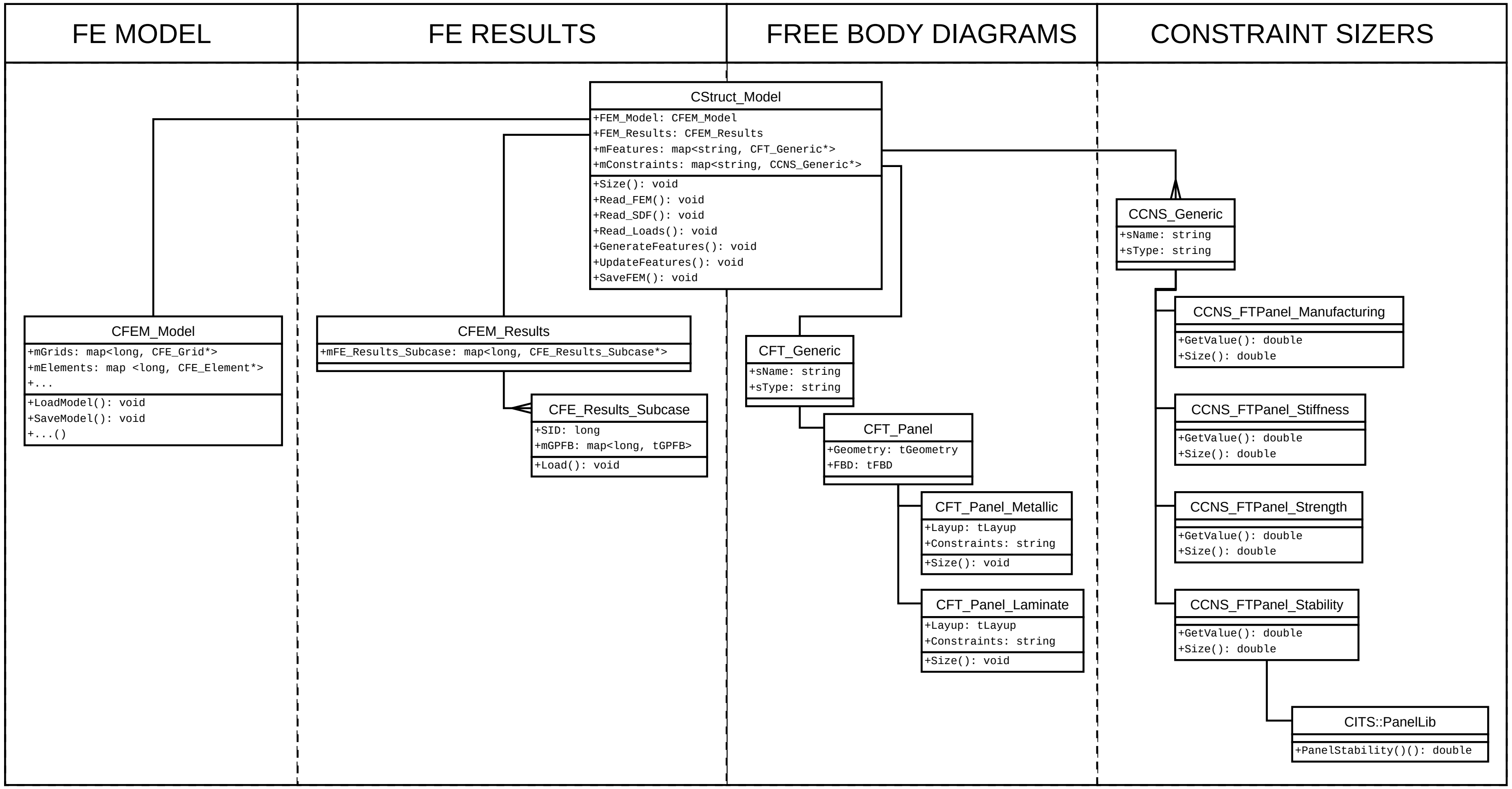



FIGURE 5.8: Arrangement of classes within AFS

### 5.3.2 Process Integration

The sizing process was integrated within the ModelCenter Problem Solving Environment (PSE) previously described in Section 3.6, since this was the likely future MDO and airframe process integration environment within the company. The system consisted of three main groups of components: the AFS sizing modules; the NASTRAN FE analysis code and data transfer routines necessary to handle the different levels of data between components. Two 'Analysis Servers' were used, shown in Figure 5.9, one on a dual processor 500MHz MS Windows NT PC running the AFS COM object, and the other on an Origin O128 machine to interface with NASTRAN. ModelCenter itself was run on a dual processor 500MHz Windows NT machine.

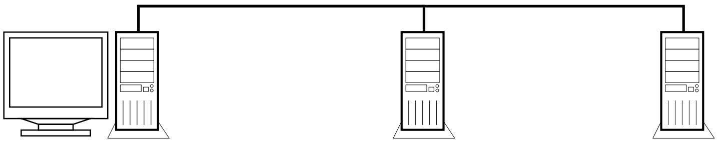



FIGURE 5.9: ModelCenter and AnalysisServer arrangement

Figure 5.10 shows the single level sizing process assembled within ModelCenter. The leftmost components with a notepad symbol are the overall process control scripts. 'INIT' accepts process parameters from the dialog boxes on screen and validates them before allowing the main 'DRIVER' script to execute the sizing process. Subsequent modules below the 'Global Model' banner create the free body diagrams to be sized. The aeroplane logo denotes these actions are performed within the AFS module. Once the features have been generated they are sized using the AFS 'SizeFeatures' module shown below the 'Size Local Model' banner. The global model is the reconstructed and its mass calculated. The result is extracted by the 'MASS' component and displayed. The 'PC_TO_ORIGIN' component is used to export the file from the AFS AnalysisServer to the NASTRAN AnalysisServer, where it is analysed. The results are retrieved using the 'ORIGIN_TO_PC' component and fed back into the 'DRIVER' process controller.

## 5.4 Testing

The AFS sizing process for strength and stability criteria was compared against sizing processes that represented the current proprietary standard (ECLIPSE) and the most readily available COTS alternative (NASTRAN SOL200). A standard rectangular wingbox model from the literature (Liu et al. (1998), Seresta et al. (2004)) was used to test the free-body-diagram approximation method for thin-plate, metallic, rectangular panels. An uninhabited air vehicle (UAV) structure was used to test the process of

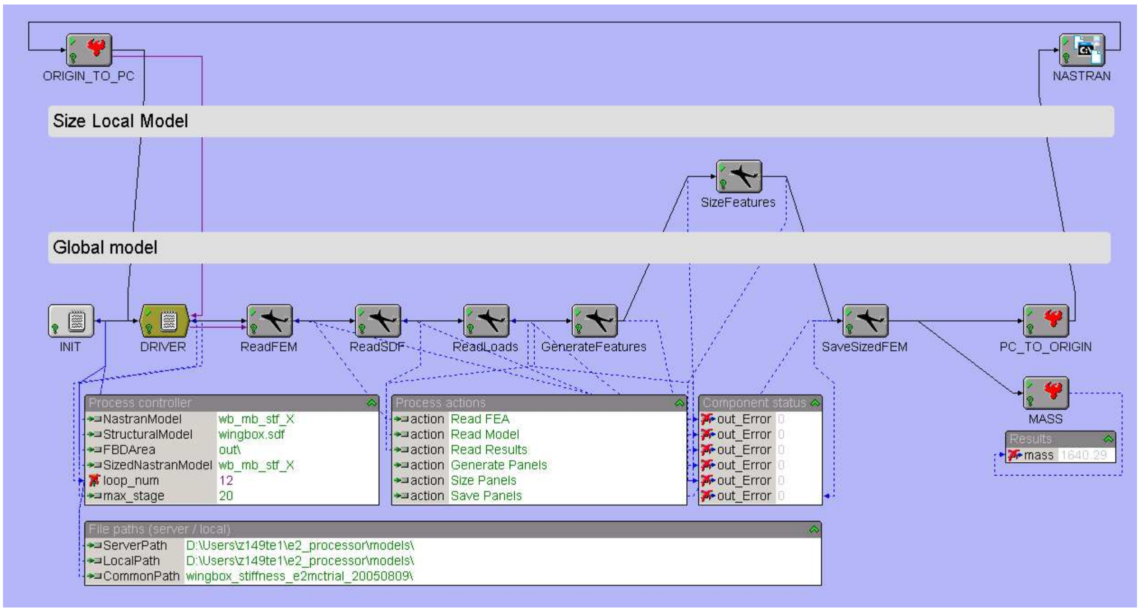



FIGURE 5.10: AFS process represented in ModelCenter

sizing airframes from the geometry definition and aerodynamic modelling stages through to structural sizing, and to understand the performance of such a system when approximating these panels as rectangular.

For these tests ECLIPSE was run for 20 iterations with elements grouped by panel association using 'OPTEL' cards. These specified the width factors necessary to approximate the panel geometry as described in Section 3.2. Sizing of panels was conducted with and without combined elements ('COMBEL') cards defined. These determined whether elements within a panel were sized based on failure criteria calculated for that particular element, or failure criteria for the worst element in the group. The NASTRAN SOL200 optimiser was run for 20 iterations using the MFD optimiser. Problems with the convergence of the eigenvalue solution meant the SOL200 optimiser would fail during the sizing process for the UAV model. The material properties used for both the rectangular wingbox and UAV models were based on aluminium as were as follows: $E = 7.2 * 10^{10} N/m^2$, $G = 2.76 * 10^{10} N/m^2$, $\nu = 0.33$, $\sigma_{ua_{normal}} = 5.03 * 10^8 N/m^2$, $\sigma_{ua_{shear}} = 2.9 * 10^8 N/m^2$ and $\rho = 2770 \text{kg}/m^3$.

### 5.4.1 Example 1 - Rectangular Wingbox

The rectangular wingbox problem described by Liu et al. (1998) was chosen as the first problem to size because of its relative simplicity. The upper skin comprises rectangular panels which should be well represented by the rectangular panel analyses used by both ECLIPSE and AFS. When loaded at the tip in a positive 'z' direction the upper skin panels will be in compression. Since the panels are rectangular and of equal size the structure should buckle globally at a load factor similar to the buckling of the individual upper skin panels.

Figure 5.11 shows the geometry for the test problem. The model and loads were defined in PATRAN. The structure was divided into groups of rectangular panels representing the ribs, spars, upper and lower skins. Each panel was assigned a unique property card, named to indicate the area of structure being sized. This naming convention is shown in Appendix E. Reserve factors of 1.0 were enforced for strength and stability constraints. Panel thicknesses were constrained to between 1.0mm and 40.0 mm. Initial panel thicknesses were set to 1.0mm. Tip forces were applied of $9.00 * 10^4 N$, $1.88 * 10^5 N$, $1.88 * 10^5 N$ and $3.80 * 10^5 N$ in the z-axis at points A, B, C and D respectively as shown in Figure 5.11. The model was supported in all freedoms at the wingbox root.

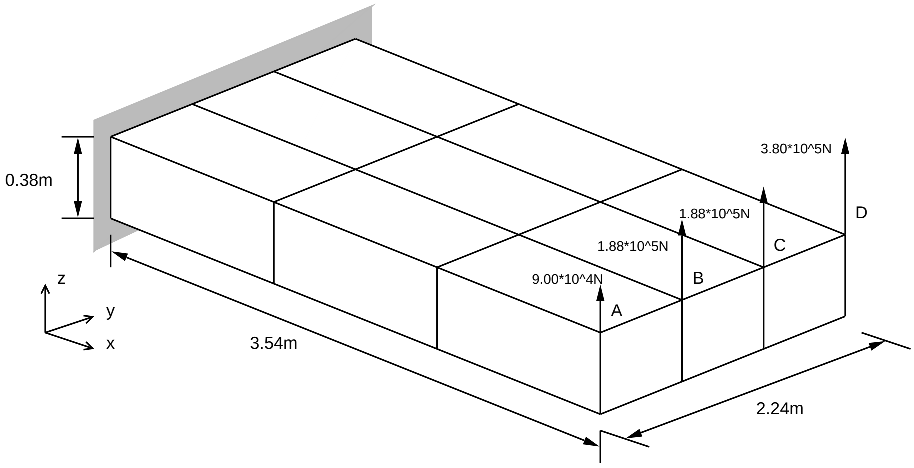



FIGURE 5.11: Rectangular wingbox geometry and load application points (internal ribs and spars are present where indicated by lines)

The solutions produced by the various systems converged to solutions between 531.6kg and 687.3kg, a difference of 155.7kg, or almost 29.2% of the lightest solution. Table 5.1 shows these upper and lower bound solutions were produced by ECLIPSE, with the AFS solution 8% (42.6kg) and NASTRAN SOL200 solution 13% (68kg) heavier than the lightest solution. Figure 5.12 shows the convergence history for the ECLIPSE and AFS sizers for the first 20 iterations. This shows that the ECLIPSE and AFS decomposition approaches converge to a given mass within the first 2 iterations, resulting in the solutions shown in Figure 5.13.

| | Masses (kg) | | | | |
|---|---|---|---|---|---|
| **Sizer** | **Spars** | **Ribs** | **Lower Skin** | **Upper Skin** | **Total** |
| ECLIPSE (COMBEL) | 96.2 | 58.5 | 159.9 | 372.8 | 687.3 |
| ECLIPSE (NOCOMBEL) | 70.2 | 49.0 | 96.7 | 315.7 | 531.6 |
| AFS | 81.9 | 47.1 | 125.8 | 319.4 | 574.2 |
| NASTRAN SOL200 | 91.0 | 63.9 | 126.9 | 317.8 | 599.6 |

TABLE 5.1: Masses of sized wingbox structural components

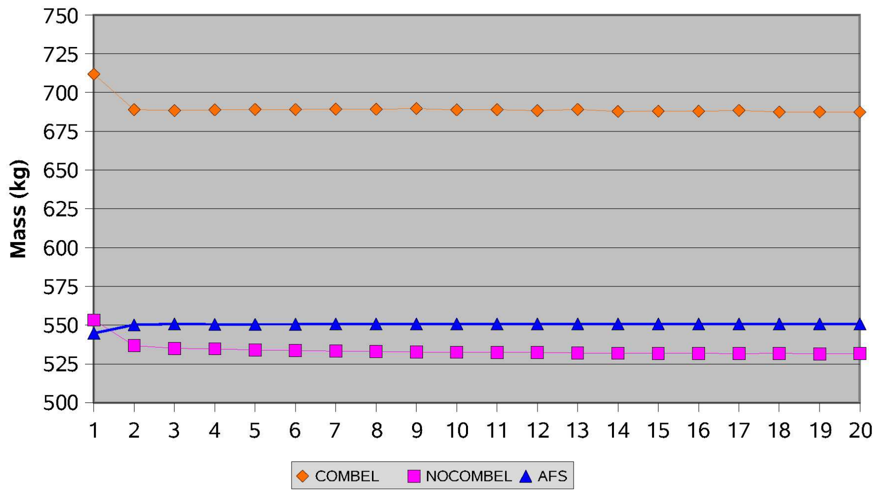



FIGURE 5.12: Convergence histories for wingbox problem

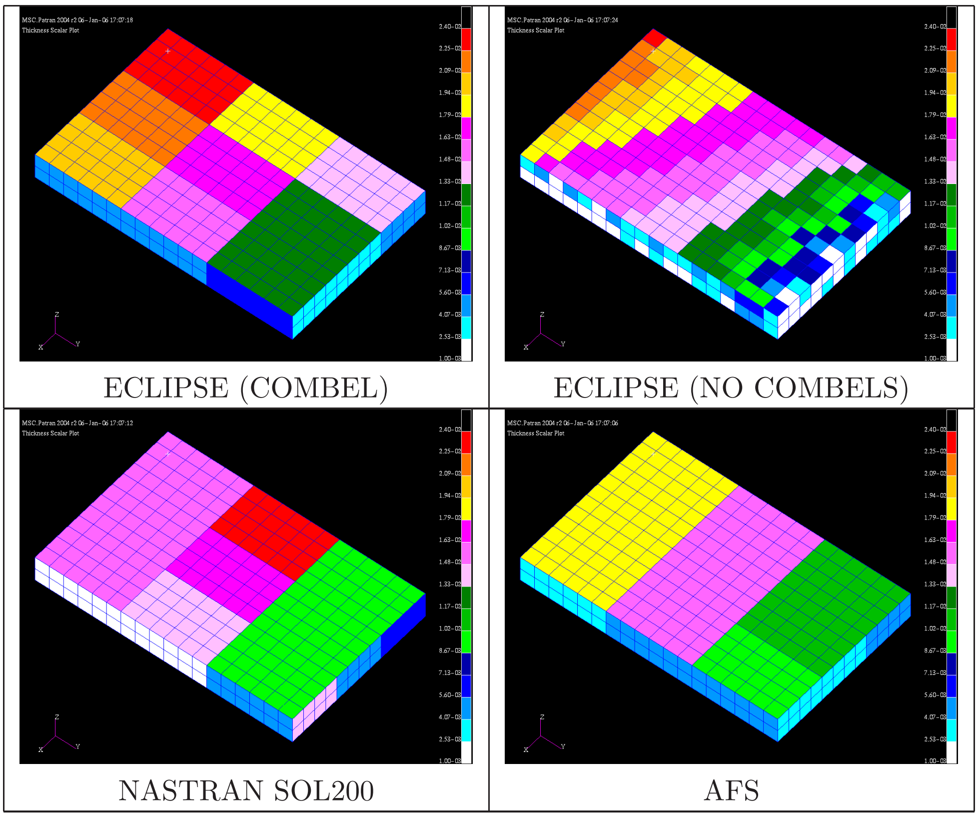



FIGURE 5.13: Solution thicknesses after 20 loops

Linear static FE analyses were performed on the sized structures to measure the strength and stability properties of the solutions. Figure 5.14 shows that the ECLIPSE (COMBEL) solution buckled under the largest load at a factor of 1.62. The NASTRAN SOL200 solution buckled closest to the constraint boundary of 1.00. It should be noted that SOL200 uses the same evaluation function. Of the three decomposition approaches AFS produced the solution closest to the constraint boundary, buckling at a load factor of 1.08. Figure 5.15 shows the stresses within the final sized solutions. Red regions indicate areas of the structure where the maximum allowable stress has been exceeded. This occurred at the interface between the spars and the lower skins of the ECLIPSE (NOCOMBEL) solution and at the edges of the root lower skin panel of the AFS solution.

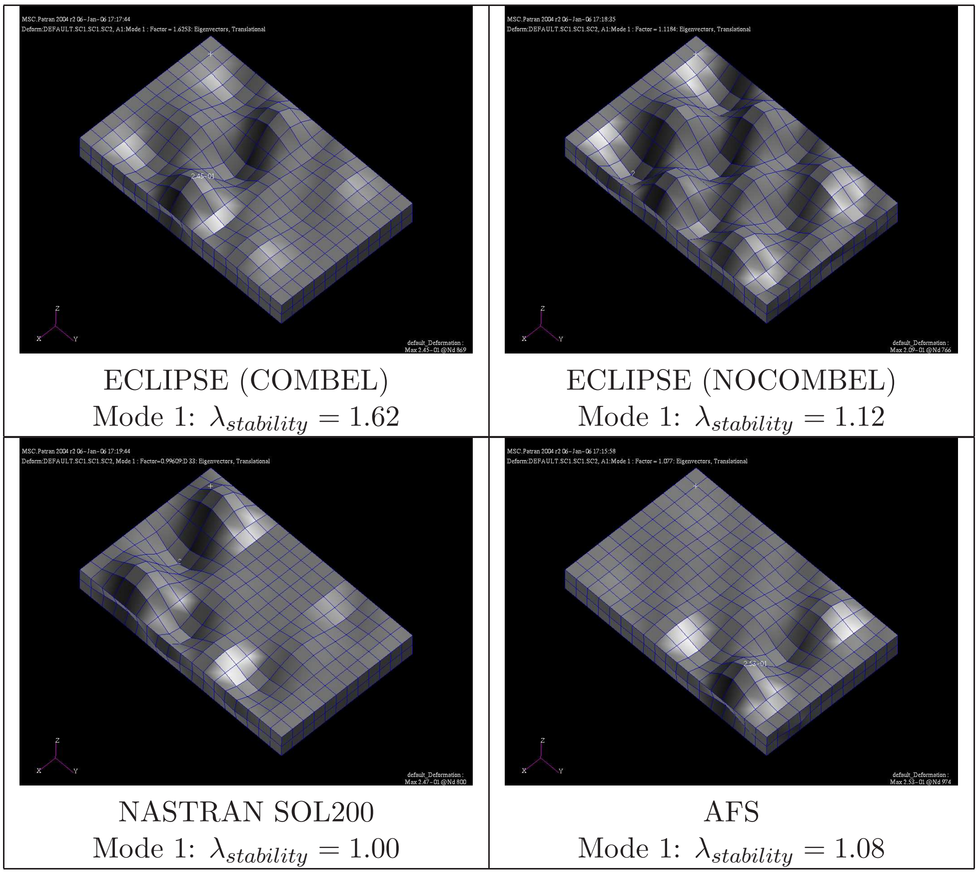



FIGURE 5.14: Verification of solution structural stability for a target value of $\lambda_{stability} = 1.00$ (after 20 loops)

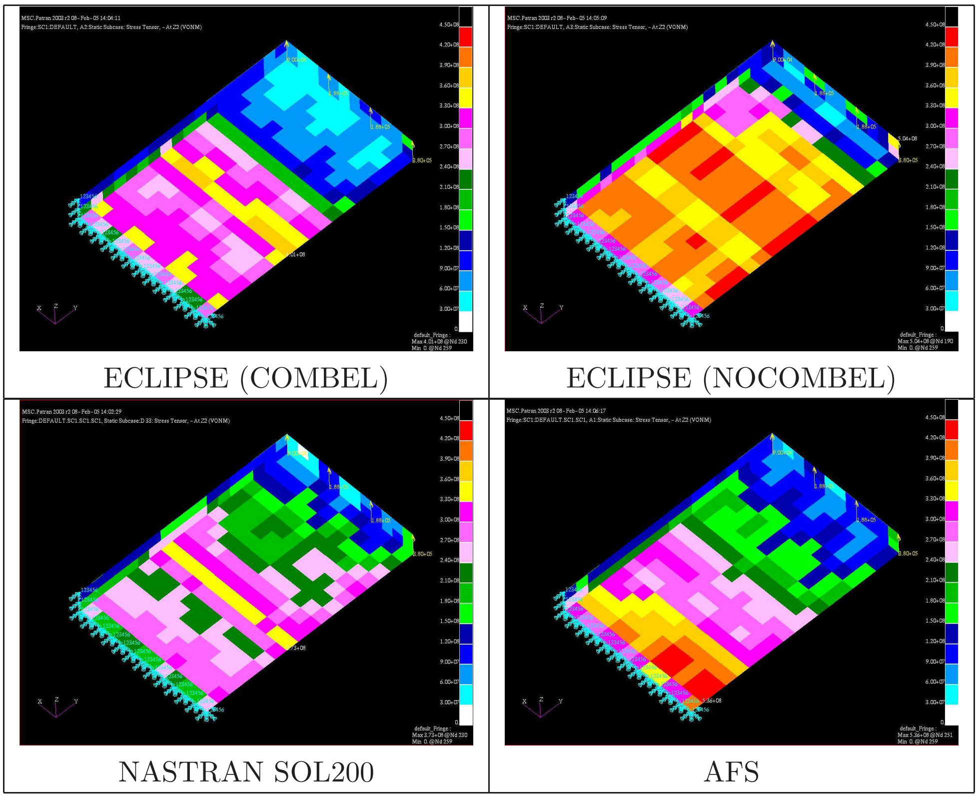



FIGURE 5.15: Verification of solution strength criteria (after 20 loops))

### 5.4.2 Example 2 - UAV Geometry

The UAV airframe shown in Figure 5.16 was generated in CATIA and analysed using VSAERO CFD for a cruise flight condition. These results were passed to the author in the form of an IGES geometry file and aerodynamic pressure coefficients over the upper and lower surfaces of the airframe. The IGES data was read into PATRAN and modified to ensure a suitable structural mesh could be produced. Modifications necessary included removing degenerate edges and ensuring nodes were located in suitable places for location of a panel edge. It is worth noting that this work was needed since the original CATIA model had not been developed for the purposes of structural sizing. The mesh was constructed using NASTRAN CQUAD4 shell elements. Net pressure differences were calculated at all points on the CFD mesh using the VSAERO pressure coefficient data as shown in Figure 5.17, the total net lift force being 26.8kN. To transfer pressure loads between the aerodynamic and structural models it was assumed that the structural mesh density was at least as high as the CFD mesh density and that they lay on the same surface. Pressure differences were calculated at each point on the structural mesh by taking an average of the equivalent values at the nearest two points on CFD mesh. This data was then exported as NASTRAN PLOAD4 cards, which could then be imported and added to the model. Suitable boundary conditions were added and the model tested in NASTRAN prior to being sized. The nosecone and centre frames of the UAV were not sized and instead fixed at 20mm thickness before sizing took place to prevent buckling in the post-processing stability analysis.

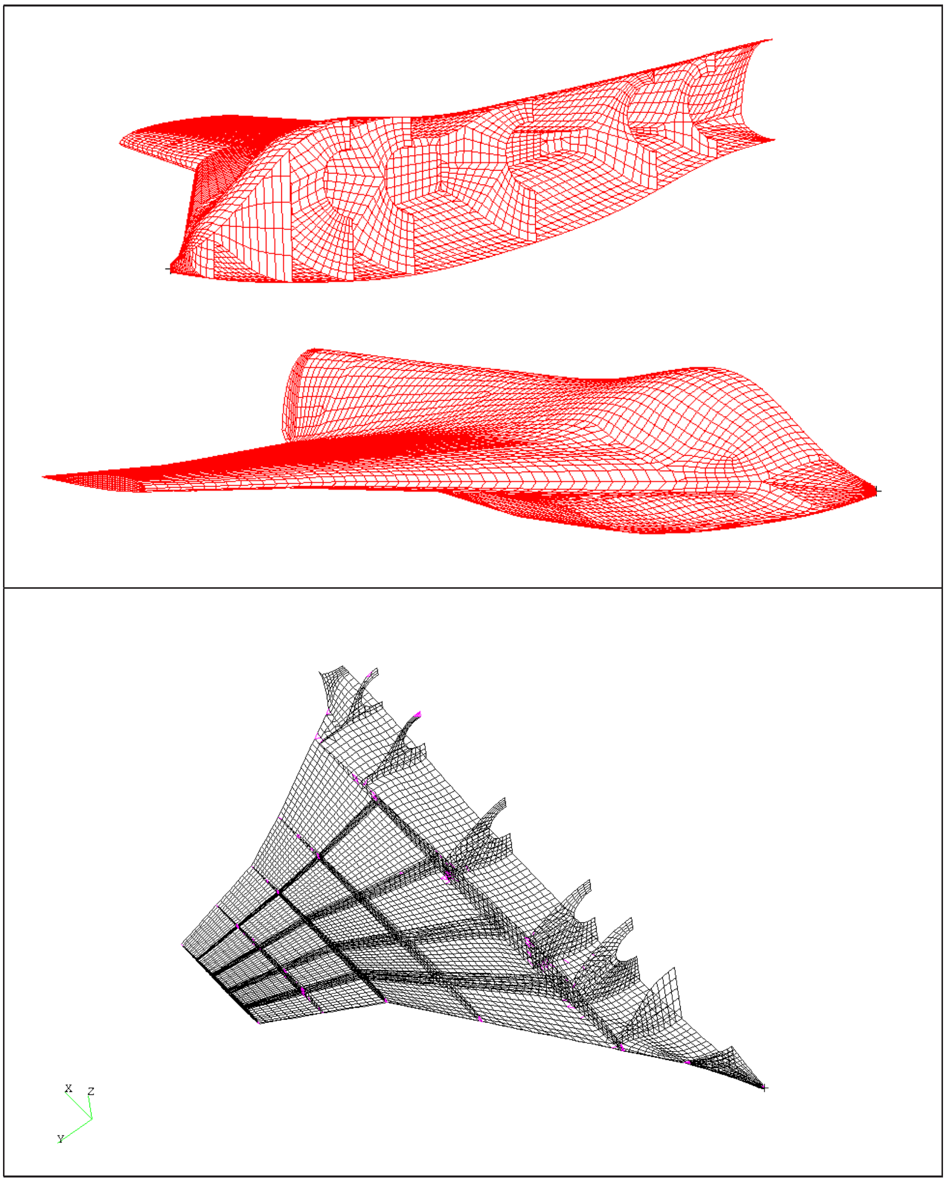


Figure 5.16: UAV external geometry (top) and internal configuration (bottom))

Table 5.2 shows the masses of the final solutions produced by ECLIPSE and AFS after 20 loops. The solution found by AFS was 91kg (16%) heavier, but 200kg (30%) lighter than the NOCOMBEL and COMBEL ECLIPSE solutions respectively. The significant differences can be clearly seen by the thicknesses plotted in Figure 5.19. The masses of the solutions proposed during sizing are plotted in Figure 5.18. It is clear that some redistribution of load occurs during the sizing process, particularly at iteration 7 and 16 of the AFS process. At these points the mass of the solution fluctuates by around 20kg. The redistribution of load, and consequent resizing of panels, is shown in the thickness plots for these solutions shown in Appendix G for the AFS solutions. Whilst

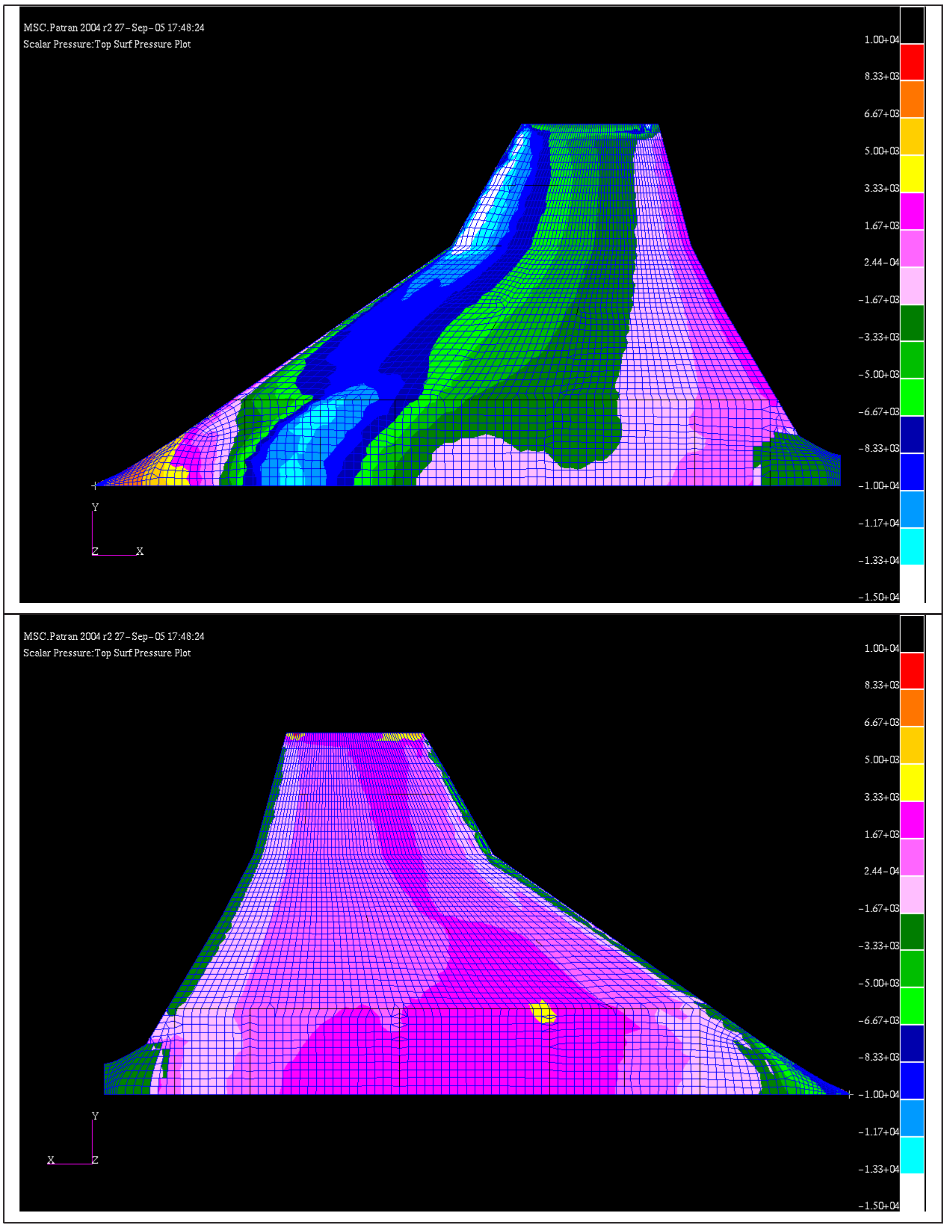



FIGURE 5.17: Pressure distribution over upper and lower surfaces giving a total resultant lift force of 26.8kN

the magnitude of the panel sizes remains constant there is noticeable oscillation between solution thicknesses.

Results of the strength analysis of the AFS and ECLIPSE solutions in Figure 5.22 show that it was not a critical constraint. The maximum von-Mises stresses observed were less than $4.14 * 10^7 N/m^2$ in the upper and lower skin panel regions. Figure 5.20 shows the results of eigenvalue buckling analyses of the sized structures. The first two buckling modes for each solution are plotted. Only the ECLIPSE COMBEL solution buckles above the target buckling value of 1.00. However, Figure 5.21 shows that although the first mode of the AFS and ECLIPSE (NOCOMBEL) solutions is less than the

| | Masses (kg) | | | | | |
|---|---|---|---|---|---|---|
| **Sizer** | **Spars** | **Ribs** | **Lower Skin** | **Upper Skin** | **Fixed Size** | **Total** |
| *ECLIPSE (COMBEL)* | 24.5 | 35.0 | 212.4 | 314.7 | 263.5 | 850.1 |
| *ECLIPSE (NOCOMBEL)* | 11.5 | 25.3 | 109.0 | 150.1 | 263.5 | 559.3 |
| *AFS* | 11.3 | 20.3 | 180.4 | 175.2 | 263.5 | 650.6 |

TABLE 5.2: Masses of sized UAV structural components

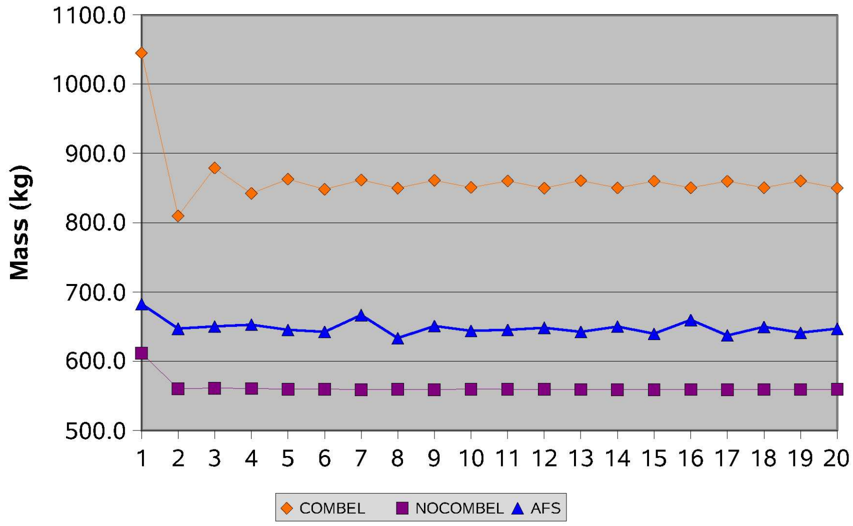


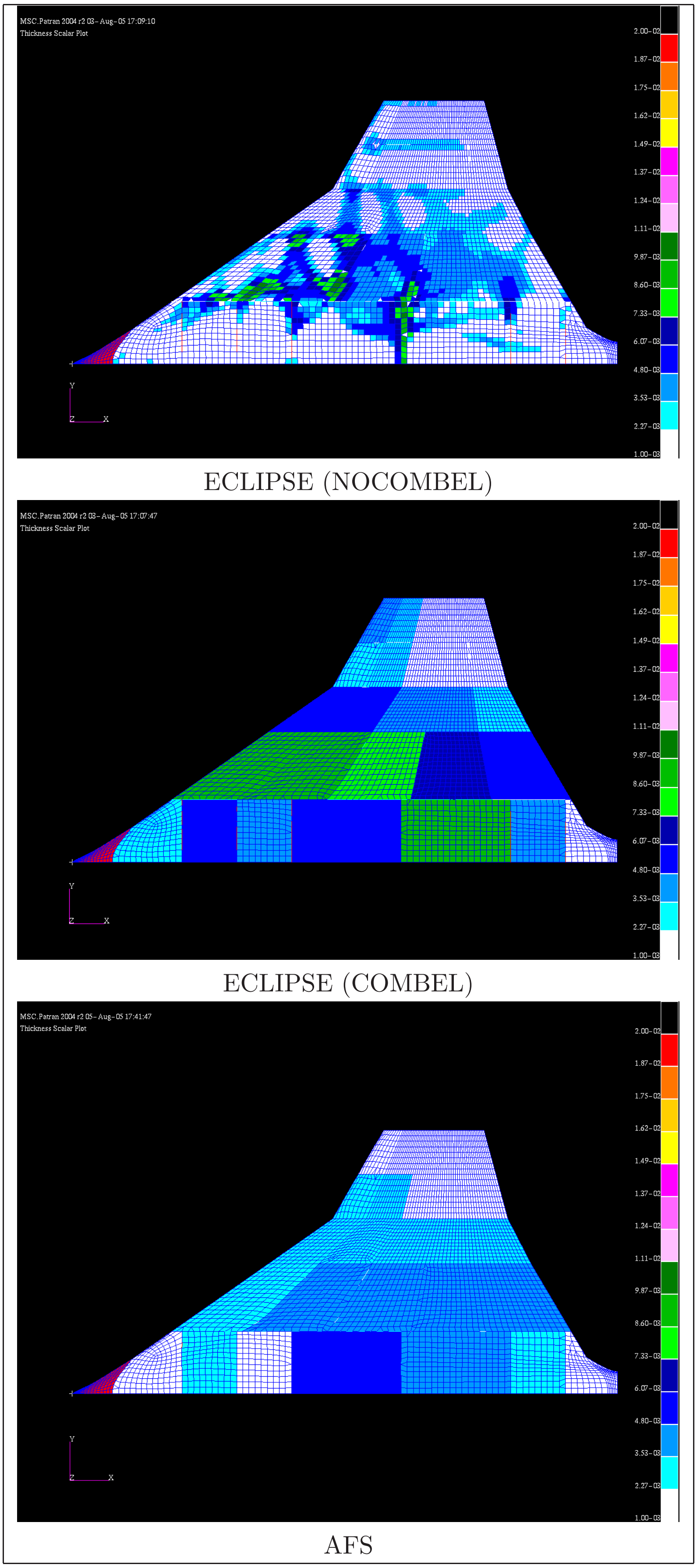

FIGURE 5.18: Convergence history for sizers used to size UAV problem

target value, the first ten buckling modes lie closer to this constraint boundary than the ECLIPSE (COMBEL) solution.

## 5.5 Discussion

### Performance of the local strength and stability sizing methods

The range of local strength and stability sizing methods tested here produced significantly different results. However, in all cases the upper and lower bounds of these answers were provided by the ECLIPSE COMBEL and NOCOMBEL solutions respectively. It is clear therefore that engineers need to understand how to interpret such results to produce a sized structure. The two UAV designs suggested by ECLIPSE would either have produced structure which was difficult to manufacture because of the highly complicated thickness distribution, or in the case of the COMBEL solution, one which was significantly heavier than the alternatives. The FBD sizing method represents a compromise between the two ECLIPSE methods. However, for strength criteria it should be noted that local stress concentrations might mean the solution exceeds the allowable stress, such as those seen in the metallic wingbox example.



ECLIPSE (NOCOMBEL)

ECLIPSE (COMBEL)

AFS

FIGURE 5.19: Thickness distribution for sized structures

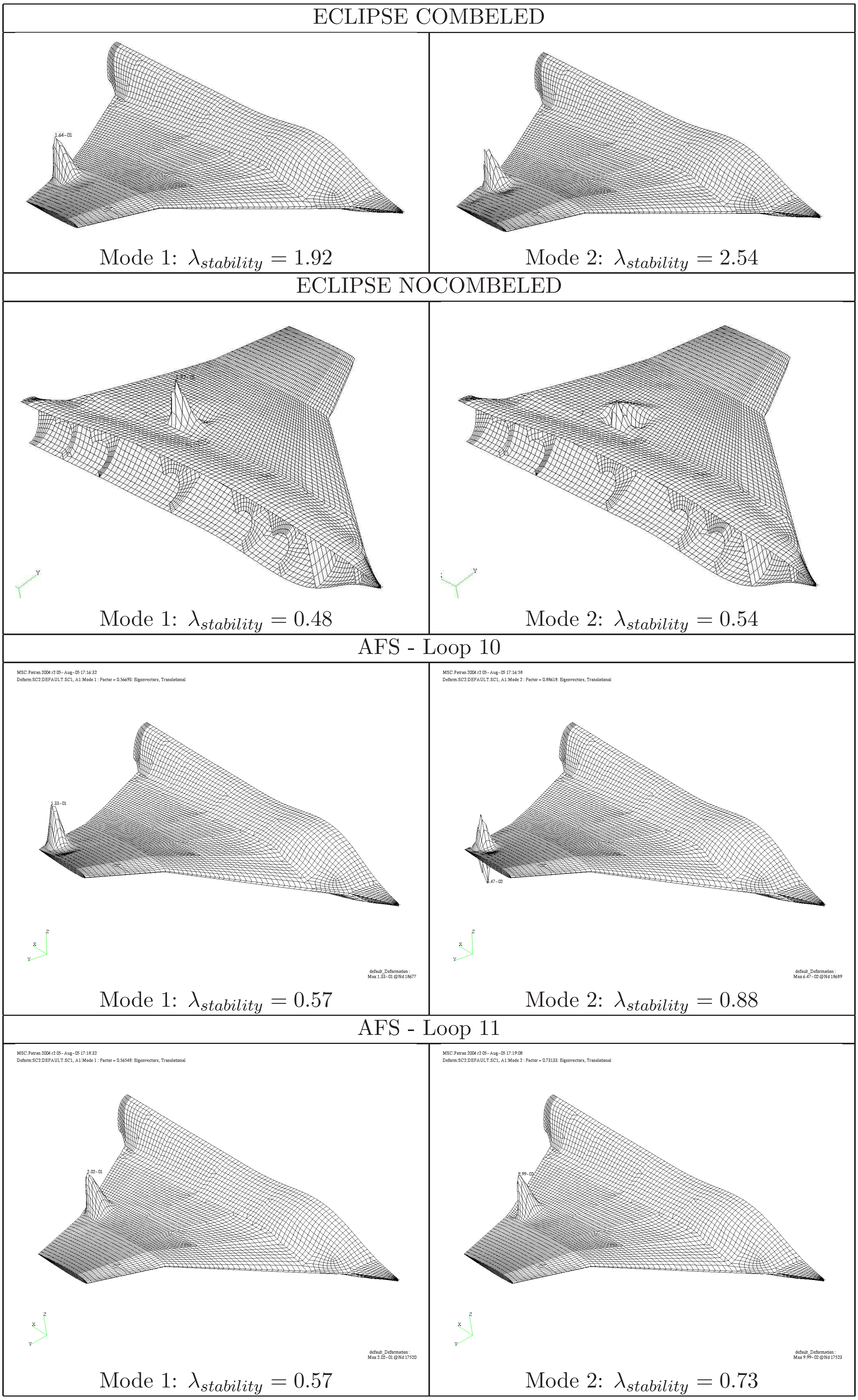



FIGURE 5.20: Verification of structural stability of the sized solutions - target: $\lambda_{stability} \geq 1.0$)

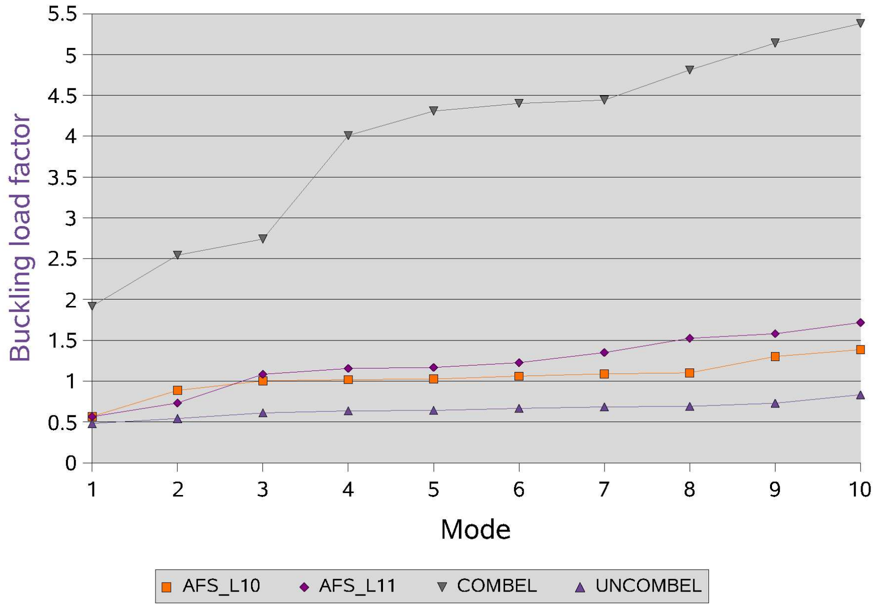



Figure 5.21: First ten buckling modes of solutions

**Enhancements to the basic integration process**

Each component wrapper in this system included a standardised interface with input and output variables describing the 'top-level' process information. This included the current iteration number and paths to files containing the medium and low-level data for that iteration such as the structural definition file and FE model. The AFS COM object was wrappered using one ScriptWrapper written in VBScript to enable the standard interface to be changed using only one file rather than using a separate script for each AFS function call. Module functions were accessed using an enumerated type variable that specified the AFS function to be performed by a particular instance of the AFS component in the process. The NASTRAN component uses the FileWrapper type to execute a local script.

Working in a development environment meant that a number of 'workarounds' were used because the installation of the PSE was not mature. One major problem was the dependence on the availability of specific AnalysisServers since the process needed reconfiguring to the local environment of each server. This was mitigated by the use of a shared central data store and constructing paths to this using the top-level variables. This is shown in the bottom dialog in Figure 5.10, which contains paths to the shared area for the server and the local machine. A common path to the model data is then augmented onto these variables.

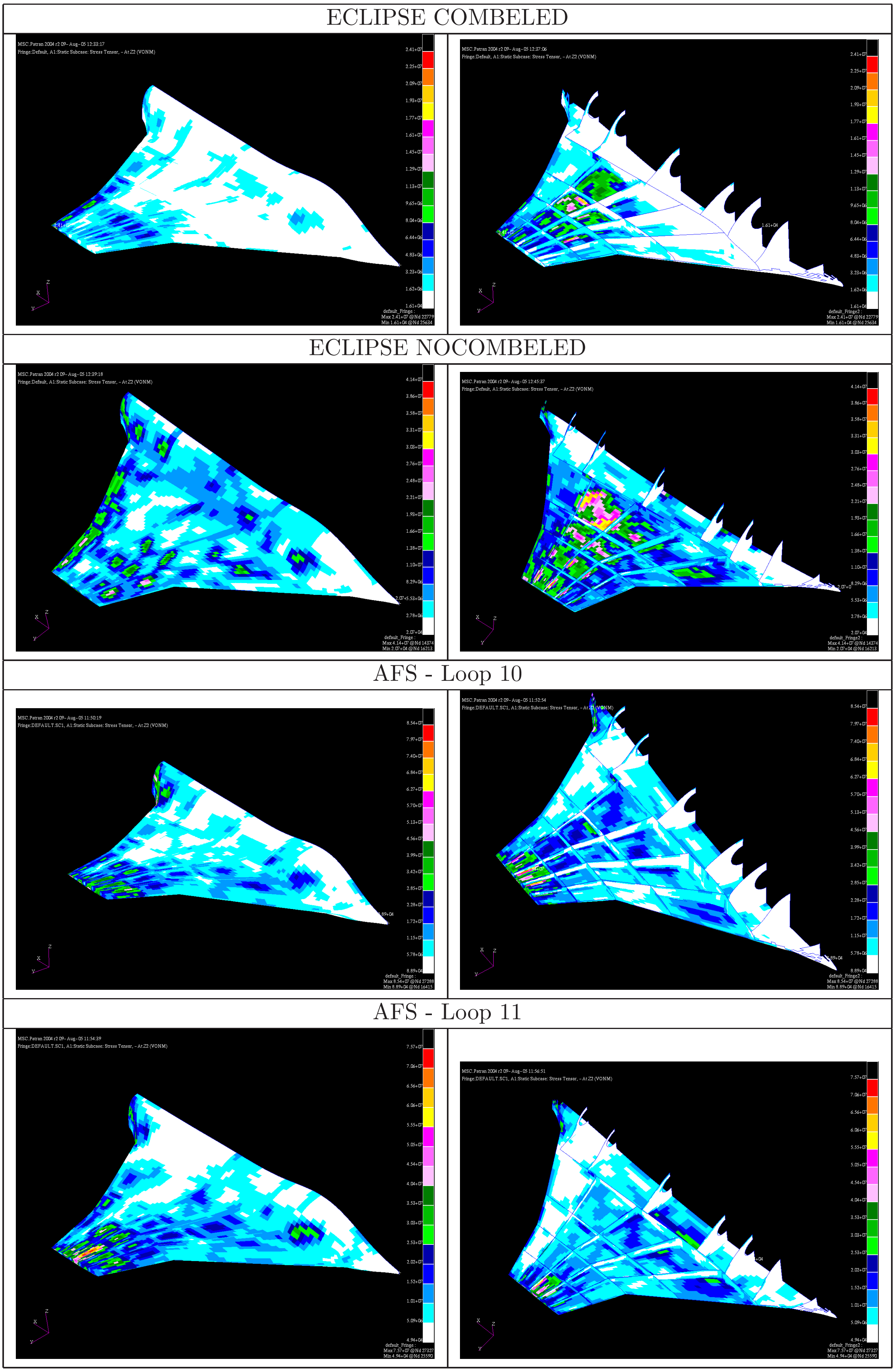



FIGURE 5.22: Verification of AFS UAV solution for strength after 10 loops - target: $\lambda_{strength} \geq 1.0$)

### Manual processes required to define the sizing problem

Table 5.23 shows an overview of the actions necessary to create the UAV model. The 'likely current process' describes the actions in the process used to size the UAV using ECLIPSE. Geometry is meshed using IGES data taken from a CATIA model. Loadings and boundary conditions are then applied. Because the ECLIPSE definition file requires the features to be identified manually the mesh is usually renumbered to place elements in particular features within defined ranges. These numbers are then transferred into the ECLIPSE file, together with the element scale factors to use for the panel stability calculation. The 'proposed process' describes timings necessary to size the model using the AFS method and a model already meshed using CATIA. These timings are based on the experiences with both systems. It is clear that significant time savings have been achieved through eliminating the need to manually define the elements in a panel feature as well as the need to manually calculate panel dimensions.

| *Likely current process* | |
|---|---|
| **Stage** | **Time (man hours)** |
| Mesh geometry from IGES file | 8 |
| Apply loads | 4 |
| Group elements into 'features' | 4 |
| Renumber elements within features | 4 |
| Define features in ECLIPSE deck using new numbers | 8 |
| Manually specify panel scale factors for each feature | 8 |
| Run sizing code | 2 |
| Post-processing | 2 |
| **Total** | **40** |
| *Proposed process* | |
| **Stage** | **Time (man hours)** |
| Mesh geometry within CATIAV5 | 4 |
| Refine mesh within PATRAN | 4 |
| Apply loads | 4 |
| Group elements into 'features' | 4 |
| Automatically define features using unique property card IDs linked to elements within a feature | 0.5 |
| Run sizing code | 2 |
| Post-processing | 2 |
| **Total** | **20.5** |

FIGURE 5.23: Estimates of time taken for a user with 'basic' experience of structural sizing using old and new processes (based on UAV model testing)

# Chapter 6

# Sizing of structures for stiffness criteria

Chapter 5 described a multilevel sizing approach and system architecture. Sizing was performed on a rectangular wingbox and example UAV structures such that they met strength and stability criteria in local regions of the structure. This involved using a decomposition approach to reformulate the overall structural sizing problem into a series of smaller sub-problems which were then solved simultaneously and the results recombined. This approach works well for strength and stability criteria where the interaction between different sub-problems is relatively limited within the structure. However, for stiffness criteria the response (e.g. deflection) is often measured a large physical distance away from the region of structure which most affects the response (e.g. the stiffness of the wing root determines to a large extent the deflection at the tip). Thus a different type of sizing approach is required to size for such criteria. Section 6.1 describes the development and implementation of a sizing method for such global stiffness criteria which is then then combined in Section 6.2 with the existing local sizing methods for strength and stability criteria. This multilevel, 'global-local', structural sizing methodology is then used to size a structure to simultaneously meet strength, stability and stiffness criteria.

## 6.1 Sizing of Structures for Stiffness Criteria

The objective of a sizing process for stiffness criteria is to minimise the mass of the structure, $M$

$$M = \sum_{i=0}^{n} m_i \tag{6.1}$$

whilst satisfying the displacement constraints, which here are formulated as equality constraints $h_j(\mathbf{m})$

$$h_j(\mathbf{m}) = C(\mathbf{m})_j - C_{j_{target}} = 0 \tag{6.2}$$

where $C_j$ is the actual displacement, and $C_{j_{target}}$ the target displacement, of a node in the direction of a unit load vector, $L_{Dj}$ and $\mathbf{m}$ is a vector of the individual element masses, $m_i$.

Candidate sizing methods for stiffness criteria included the direct optimisation approach and the optimality criterion approach discussed previously in Section 3.5. A direct method was not chosen for this work because there was no readilly available optimisation method in the current toolset. The existing optimality criterion approach was chosen because it would help company understand if their current method was possible within the likely future toolset rather than the legacy toolset. One of the benefits of this would be to allow a company to reuse mature aeroelastic sizing processes. It should be noted that in a multilevel optimisation approach it should be possible to exchange sizers at the different levels providing the integration of the process can accommodate such changes. Therefore in the proposed process the optimality criterion approach could be exchanged without the need to redevelop the individual sizing codes, although the new sizing method would need to be integrated into the process. The optimality criterion method used in ECLIPSE was adapted and added to the methods currently in the AFS system described in Chapter 5.

### 6.1.1 Methodology

The methodology used here is adapted from the current ECLIPSE methodology as described in Appendix C. For the purposes of this study it was assumed that membrane strain energies were dominant in the structure and that bending and membrane-bending stiffnesses could be neglected. This assumption eliminated the need for the FE model to be reconfigured using duplicate elements superimposed on each other in order to extract membrane and bending strain energies (as described in Section 3.4). The resulting changes to these equations are detailed below.

The first stage of the sizing process was to estimate the initial values of the Lagrange multipliers for each constraint, $j$, using Equation C.29

$$\lambda_j = \frac{2C_{jv0}}{pC^2_{jv_{target}}} M_{v0} \tag{6.3}$$

where $M_{v0}$ is the initial mass of structure being varied, $p$ is the number of constraints, and the initial value of the constraint measured in those elements that can be varied,

$C_{jv0}$, is $2\sum_{i=1}^{N} E_{ji}$. $C_{jv_{target}}$ is the target value to be achieved in the structure that can be varied. Since all strain energy was considered to be due to membrane stiffnesses then in this case $E_{ij}$ was assumed to be equal to the total strain energy calculated for an element by NASTRAN for constraint case $j$ and loadcase $i$. The methodology specifies that element scale factors, $\alpha_i$, be found by solving Equation C.35

$$\alpha_i^4 m_{i_0} - \sum 2\lambda_j \left(E_{1_{ji}}\alpha_i^2 + 2E_{2_{ji}}\alpha_i + 3E_{3_{ji}}\right) = 0 \tag{6.4}$$

where $E_{1_{ji}}$, $E_{2_{ji}}$ and $E_{3_{ji}}$ represent the strain energy in an element due to membrane, membrane-bending and bending stiffness respectively. Using only element membrane strain energies this equation can therefore be simplified to

$$(m_{0_i})\alpha_i^2 - \left(\sum_{i=1}^{N} \lambda_j E_{ji}\right) = 0 \tag{6.5}$$

giving

$$\alpha_i = \sqrt{\frac{\sum_{i=1}^{N} \lambda_j E_{ji}}{w_{0_i}}} \tag{6.6}$$

New estimates of the Lagrange multipliers were created using Equation C.41

$$\lambda_{new} = \lambda_{old} + D^{-1}(C_j - C_{j_{target}}) \tag{6.7}$$

where

$$\frac{dC_j}{d\lambda_k} = \sum_{j=1}^{M}\sum_{k=1}^{M} \frac{dC_{j_{target}}}{d\alpha_i}\frac{d\alpha_i}{d\lambda_k} \tag{6.8}$$

$$\frac{dC_{j_{target}}}{d\alpha_i} = \frac{-2}{\alpha_i}\left(\frac{E_{ji}}{\alpha_i}\right) \tag{6.9}$$

$$\frac{d\alpha_i}{d\lambda_k} = \frac{E_{ki}\alpha_i^2}{(4m_{i_0}) - \sum 2\lambda_j E_{ik}\alpha_i} \qquad (6.10)$$

Element scale factors in the model were then recalculated using Equation 6.6 and the element resizing process repeated until convergence.

### 6.1.2 Implementation

This method was implemented within the existing AFS sizing system described in Chapter 5. The benefits of this approach were that the existing NASTRAN file handling routines could be reused, together with the existing ModelCenter wrappers. However, within the overall process the sizing methods for global stiffness criteria and local strength and stability criteria were treated as mutually exclusive. Figure 6.1 shows the process for sizing the FE model for stiffness criteria. Stages A to B define the FE model and parameterise it using the same feature definition approach used for strength and stability criteria. Appendix F.1 shows an example definition of the stiffness criteria and a wing skin panel feature. Whilst free-body diagrams of the features are not created, the elements associated with the feature are sized based on a mean thickness calculated from the suggested individual element sizes within the feature. Stages C and D are carried out to create the virtual strain energies required by the sizing process. Unit forces were added to the FE model at the nodes, and in the direction of, displacements to be measured. NASTRAN case control decks were manually created to define the order of the analyses such that the unit load for a given stiffness constraint would be ‘crossed’ with the applied load case for that constraint (See Equation C.6 in Appendix C). The NASTRAN DMAP alterations previously discussed in Section 3.4 were added to this case control deck. This model was then analysed in NASTRAN. The resultant strain energies were read from the NASTRAN F06 output file into AFS and used to calculate an initial estimate of the Lagrange multiplier, $\lambda_j$, for each constraint $j$ using Equation 6.3. Element scale factors were calculated using Equation 6.6 and new estimates of the Lagrange multipliers calculated using Equation 6.7. Once the inner loop had converged then the elements thicknesses in the FE model were resized using the scale factors, $\alpha_i$. The elements were resized using an average of the scale factors for individual elements in the group. For stiffness only sizing cases the resized structure was reanalysed using NASTRAN and the overall sizing process repeated for a fixed number of loops.

### 6.1.3 Testing

The metallic wingbox model tested previously in Chapter 5 was adapted in order to test the sizing process for stiffness criteria. Figure 6.2 shows the two cases considered. Case

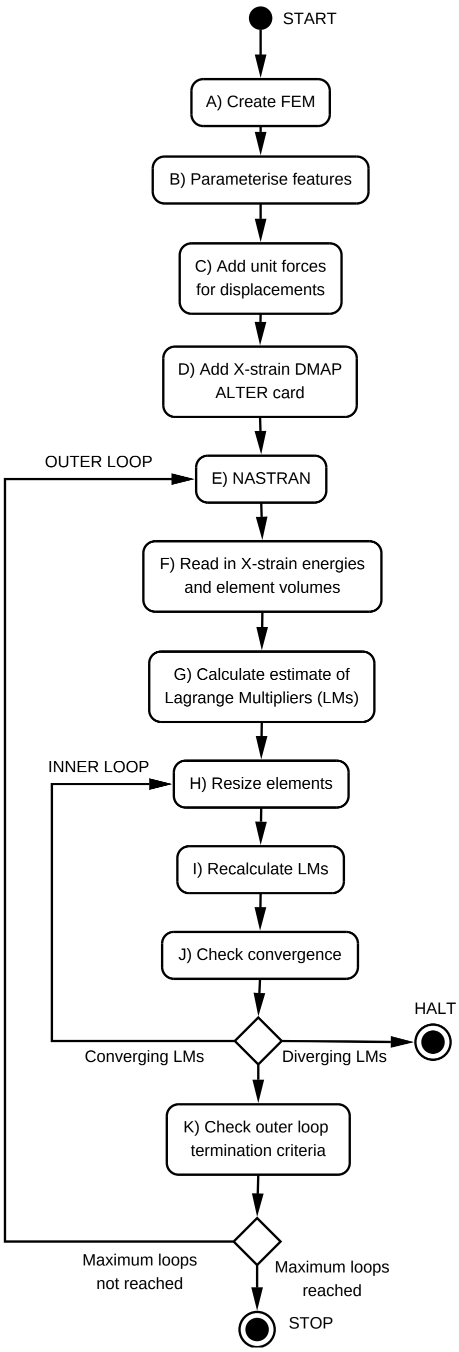
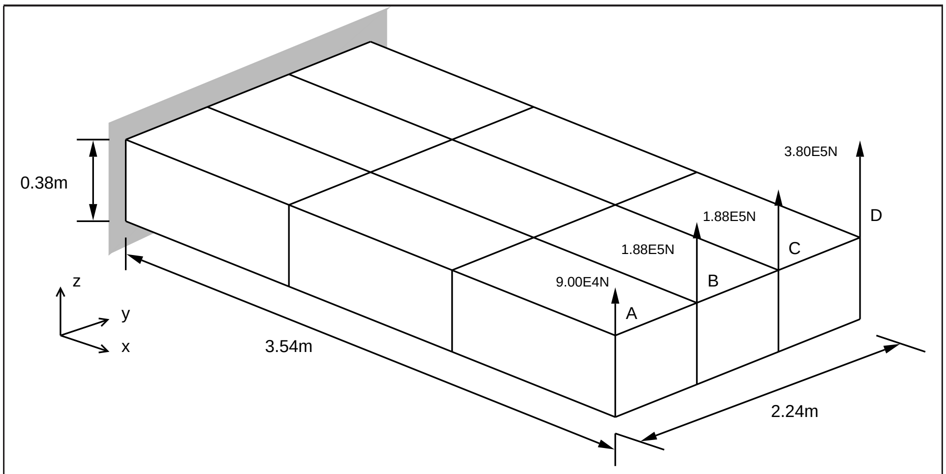


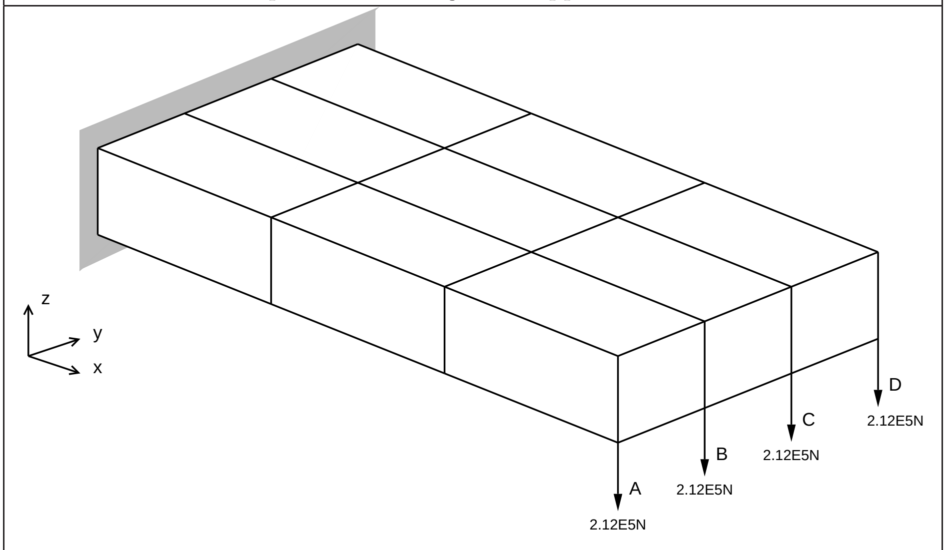

Figure 6.1: Stiffness method used in ECLIPSE

1 was the upward twist load case used previously in Chapter 5. Case 2 was a uniformly distributed downwards load of the same overall magnitude as Case 1, but applied to the lower edge of the wingbox tip. In case 1 the tip deflection at the upper corner of the box was constrained to a deflection of 50mm, whereas for case 2 the tip was constrained at the bottom right corner to a deflection of 50mm.

The AFS sizing process for stiffness criteria was tested against the existing ECLIPSE process. As for the tests carried out with ECLIPSE in Chapter 5 the features were modelled in ECLIPSE with and without the element thicknesses 'COMBEL' together (see Section 5.4). Initial panel sizes were set to 1mm for both the AFS and ECLIPSE models. Panel thicknesses were constrained to between 1mm and 20mm. Results described in these sections are compared against the ECLIPSE results after 20 iterations since the represents the best optima likely to be found.




*Case 1: Upwards bending with applied twist loadcase*




*Case 2: Downwards bending loadcase*

FIGURE 6.2: Boundary conditions for wingbox stiffness problem (internal ribs and spars are present where indicated by lines)

Figure 6.1 shows the masses of the solutions found for this problem using ECLIPSE and AFS. The agreement between these solutions was, unsurprisingly, close. The heaviest mass of 1075.4kg, found by the ECLIPSE COMBEL method, was 60.4kg, or 6%, heavier than the lightest solution of 1014.9kg, found by the ECLIPSE NOCOMBEL solution. The convergence history for the three sizing methods is shown in Figure 6.3. The ECLIPSE COMBEL solution method took 4 iterations to converge to within 1% of the final solution mass, whereas the NOCOMBEL solution converged to within 1% of the final solution mass by loop 15. The AFS solution converged to within 1% of the mass of the solution by loop 4. The final solutions found by ECLIPSE are plotted alongside those found by AFS for loop 5 and 6. The AFS solutions show that the solution has converged, with little redistribution of load taking place between iterations.

| | Masses (kg) | | | | |
|---|---|---|---|---|---|
| **Sizer** | **Spars** | **Ribs** | **Lower Skin** | **Upper Skin** | **Total** |
| ECLIPSE (COMBELS) | 193.4 | 14.6 | 433.4 | 434.0 | 1075.4 |
| ECLIPSE (NOCOMBELS) | 184.3 | 10.2 | 409.9 | 410.6 | 1014.9 |
| AFS | 171.5 | 15.1 | 420.4 | 421.0 | 1028.1 |

Table 6.1: Masses of sized wingbox structural components

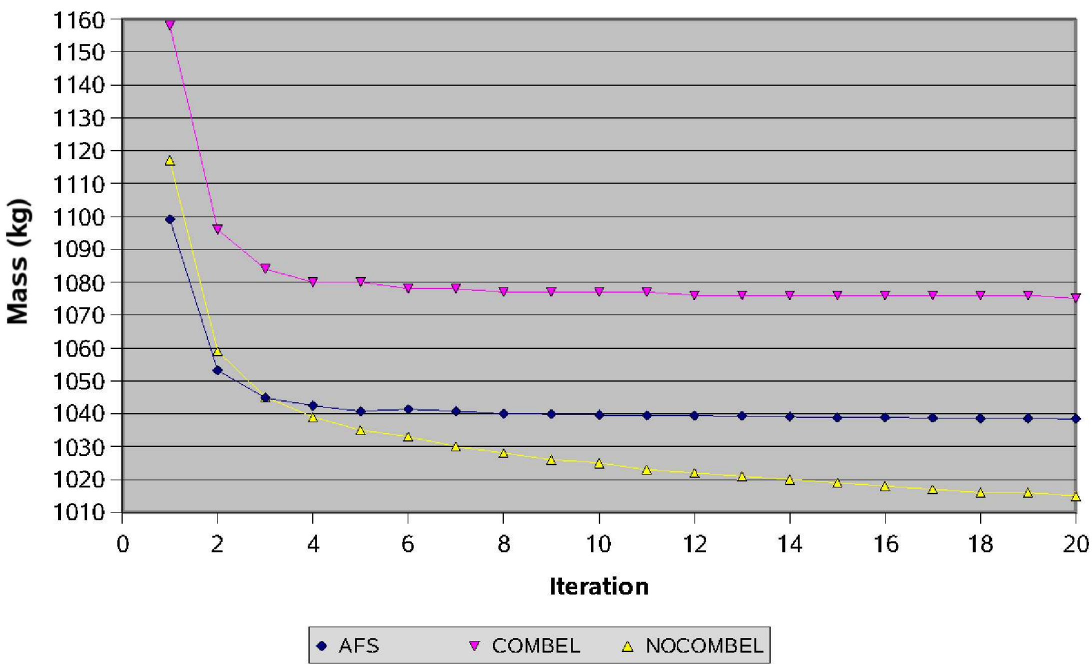
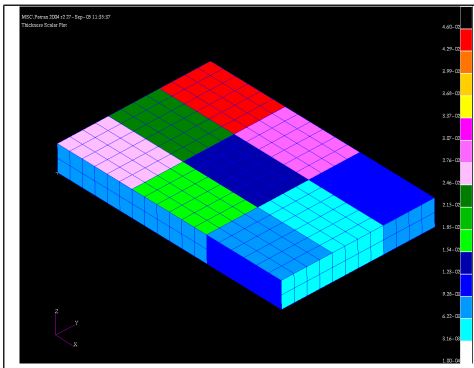
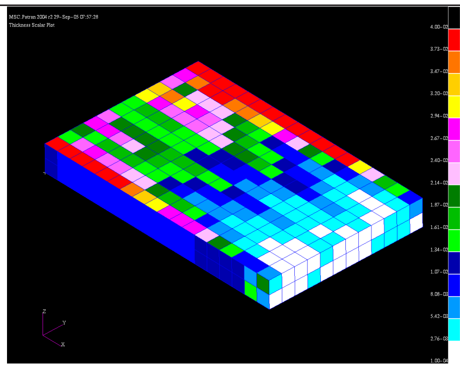



Figure 6.3: Masses of the solutions found during the sizing process for stiffness only constraint

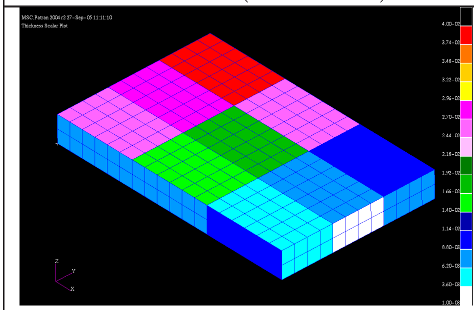
Figure 6.6 shows the response of the structure to the applied loads as the structure is sized for the two stiffness criteria. The deflections of the initial structure under the applied loads were +1.12m and -1.04m for the upwards and downloads bending loadcases respectively. After the first outer sizing loop the magnitude of the deflections decreased



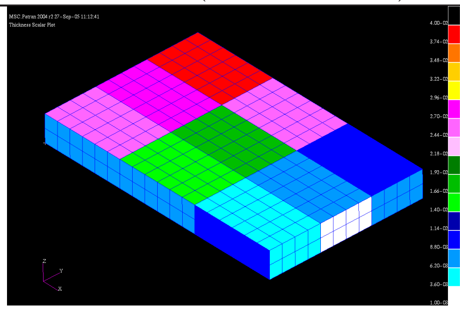
ECLIPSE (COMBEL)



ECLIPSE (NOCOMBELS)



AFS - Loop 5



AFS - Loop 6

FIGURE 6.4: Thicknesses of different solutions to the problem

to +0.097m (+97mm) and -0.088m (88mm). After 6 sizing loops the deflections had reduced to +0.057m (+57mm) and -0.054m (-54mm) respectively, a difference of 7mm and 4mm from the target displacement values of 50mm. Figure 6.1.3 shows that the ECLIPSE solutions converged to within 0.5mm of the target solution.

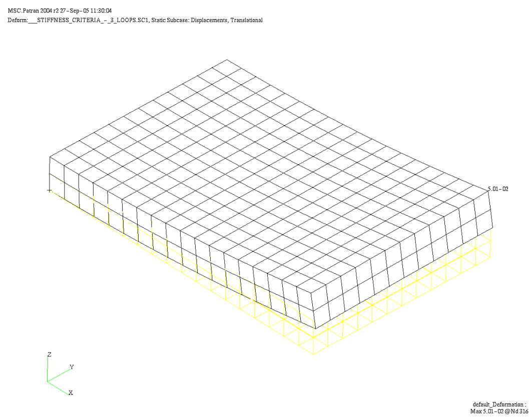
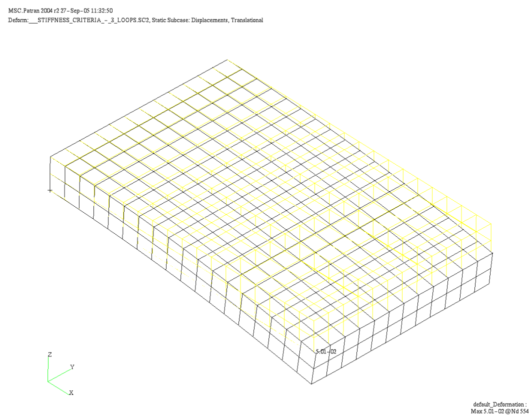
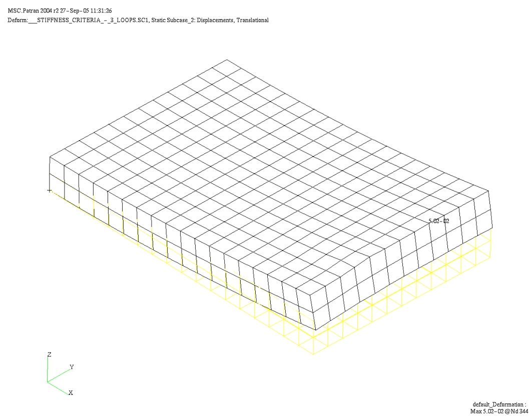
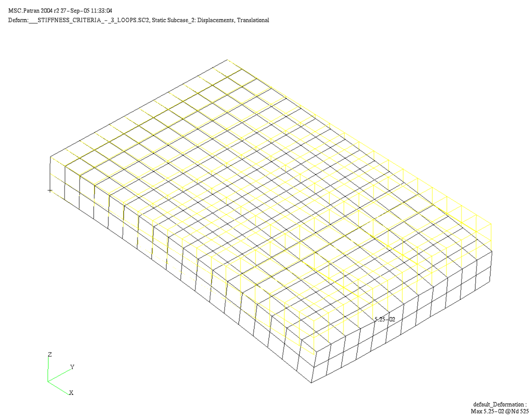
| **Upwards Twisting** | **Uniform Download** |
|---|---|
| **ECLIPSE (COMBEL)**<br>**Loop 20 -** $\delta_z = +0.050m$<br> | **Loop 0 -** $\delta_z = -0.050m$<br> |
| **ECLIPSE (NOCOMBEL)**<br>**Loop 20 -** $\delta_z = +0.050m$<br> | **Loop 0 -** $\delta_z = -0.050m$<br> |

FIGURE 6.5: Deflection of solutions sized for stiffness criteria using ECLIPSE

## 6.2 Sizing of Structures for Strength, Stability and Stiffness Criteria

This section describes the implementation of a basic, global-local sizing methodology which sizes metallic slab panel structures for strength, stability and stiffness criteria. The sizing processes previously described for stiffness criteria and strength and stability criteria were used for global and local sizing analyses respectively.

### 6.2.1 Methodology

The purpose of the methodology used here was to size the structure for stiffness criteria as well as strength and stability criteria. In order to link the sizing of the two structures a 'move limit' constraint was added to the AFS module. It ensured features within the structure could not be sized greater or less than a factor of the original size. Thus by

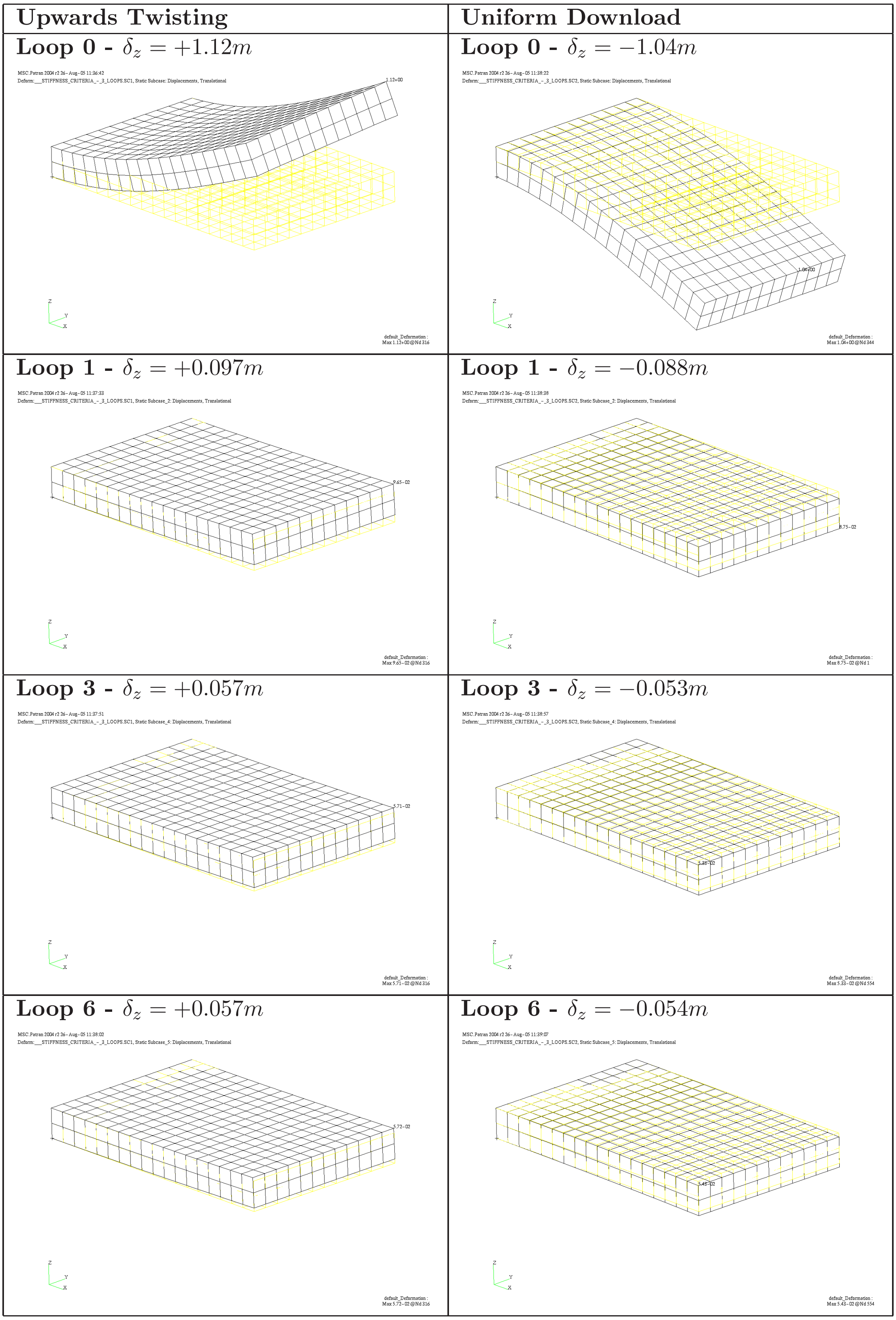



FIGURE 6.6: Deflection of solutions sized for stiffness criteria using ECLIPSE

setting the lower move limit to a factor of 1, this would constrain the sizer from moving below the original size of the feature.

For the purposes of this sizing process a simple move limit constraint was added to the AFS library and used in conjunction with the local sizing methods. Since it was known that the solution for the wingbox problem was heavier when sized for stiffness constraints than strength or stability constraints, it was assumed that the stiffness sizer should dictate minimum sizes of the structure. The subsequent local sizing of the structure would serve to ensure the stiffness sized structure met these local criteria. Thus the move limit constrained the sizer to a feature scale factor $S$ within the range of $1 \leq S \leq 1{*}10^{100}$. No movelimit constraint was imposed on the sizing for stiffness criteria. An automated method of determining the critical constraint and linking it to the sizing process in other levels would be required for a production system (e.g. the linking of smeared design properties for stiffened panels to be included in the local optimisation (Ragon et al. (2003))).

Figure 6.7 shows the multilevel sizing process. The unsized FE model is augmented with a case control deck that allows strain energies to be calculated for the different stiffness constraints. The case control deck is removed and the results of the FE analysis fed into the sizing process for stiffness criteria. A case control deck for strength criteria is then added to the FE model and the structure sized using the strength, stability and movelimit constraints. This produces a structure sized to meet strength, stability and stiffness constraints. The process is repeated for a fixed number of iterations. Figure 6.8 shows the implementation of this process within the ModelCenter framework. The process was split into separate assemblies to simplify the navigation of the process, although this increased the complexity of the connections between components.

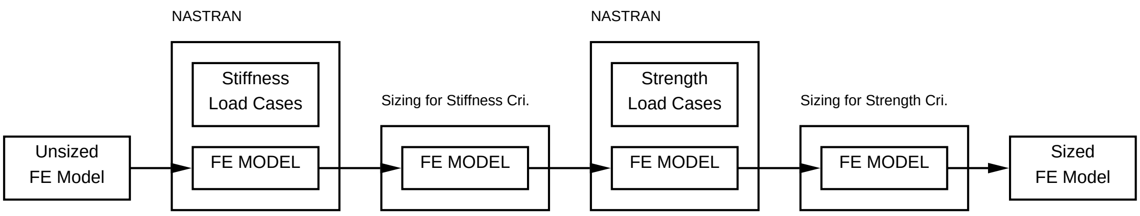



FIGURE 6.7: Sizing process for strength, stability and stiffness constraints

The structure was sized for stiffness criteria at the global level using the method described in Section 6.1. The structure was then sized to meet local strength and stability criteria using the method demonstrated in Chapter 5.

This methodology creates a sizing framework which could then be extended to include local stiffened panel sizing using approaches such as that described by Ragon et al. (2003). In addition it would be more straightforward to incorporate COTS codes such as Hypersizer whilst also using proprietary capability such as the stiffness method analysed previously. For the multilevel approach to utilise ‘off-the-shelf’ sizing processes a set of rules would be required to define the most active constraints in the sizing process overall, and ensure that the different sizing processes minimised the mass of the structure whilst meeting all design criteria. For the purposes of this work it was assumed that the stiffness criterion was the dominant factor in the sizing process as this had produced the heaviest mass structure.

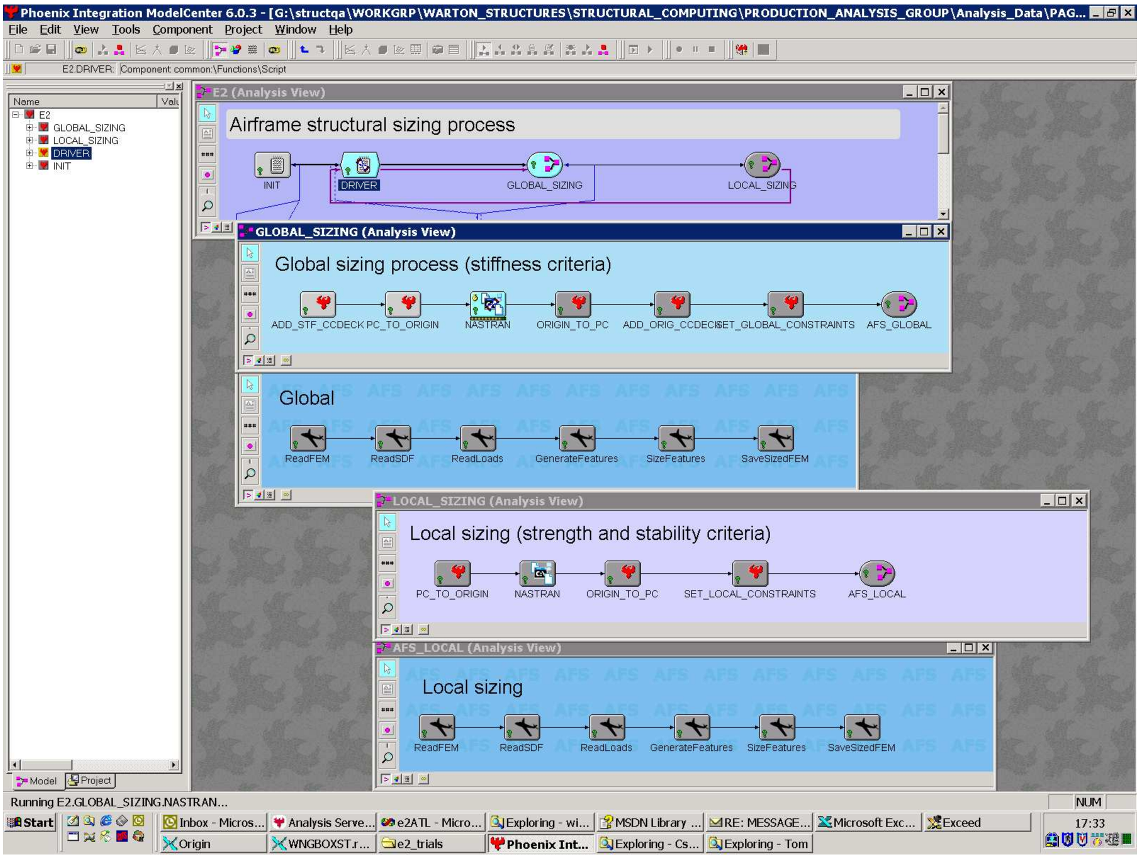



FIGURE 6.8: Multilevel sizing process integrated within ModelCenter for strength, stability and stiffness constraints

### 6.2.2 Testing

The metallic wingbox structure was sized using a combination of the design criteria previously tested. The displacement of the two corner points was constrained using the same parameters described in the previous section. In addition the wingbox was constrained to prevent failure through strength or stability for the upbending loadcase as described in Chapter 5.

Table 6.2 shows the masses of the solutions found for this structure. As with the stiffness only case the difference between the upper and lower bounds of the solution masses was small, with a 5% difference betwen the mass of the ECLIPSE (COMBEL) solution and the ECLIPSE (NOCOMBEL) solution. The AFS solution was between these two bounds as shown by Figure 6.9. This also shows a significant change in the sized mass of the solution. This was attributed to the redistribution of load causing a panel stability constraint to become inactive. Table 6.10 shows the range of thicknesses obtained from the three sizing methods.

| | Masses (kg) | | | | |
|---|---|---|---|---|---|
| **Sizer** | **Spars** | **Ribs** | **Lower Skin** | **Upper Skin** | **Total** |
| ECLIPSE (COMBELS) | 191.0 | 22.6 | 436.9 | 430.2 | 1080.7 |
| ECLIPSE (NOCOMBELS) | 182.6 | 17.7 | 415.1 | 414.8 | 1030.0 |
| AFS | 169.0 | 18.5 | 418.9 | 466.7 | 1073.2 |

TABLE 6.2: Masses of sized wingbox structural components

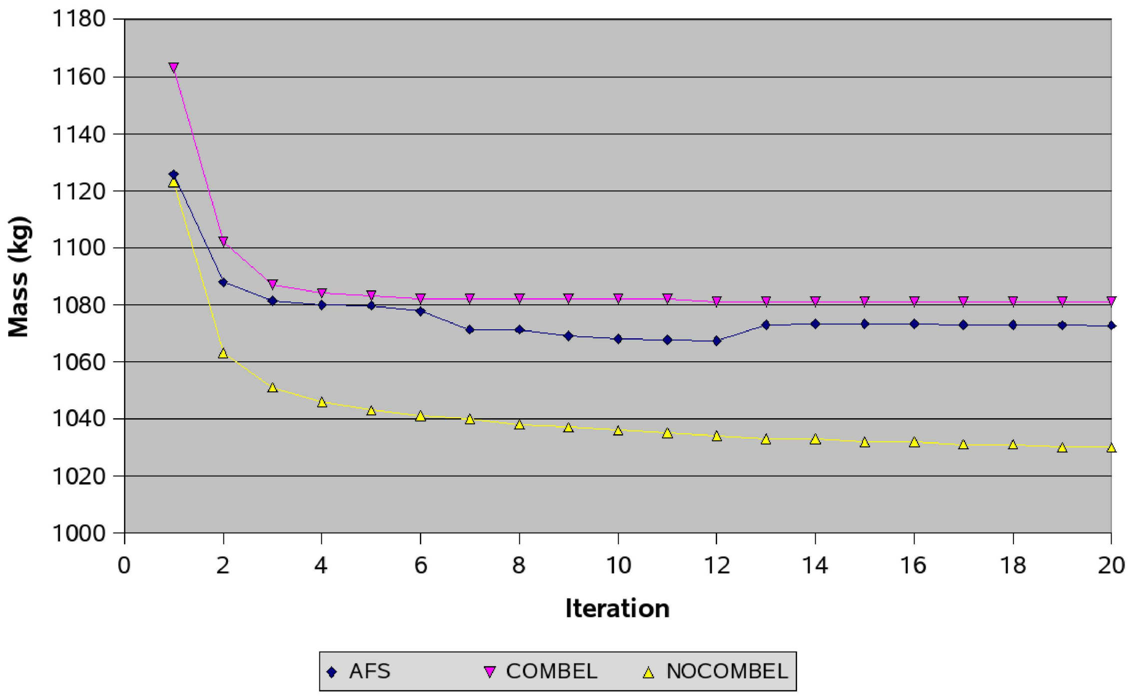



FIGURE 6.9: Masses of the solutions found during the sizing process for combined strength, stability and stiffness constraints

Table 6.5 shows the deflection of the sized solutions under the applied load. As with the stiffness only test case the ECLIPSE solutions best met the displacement criteria, each moving precisely 0.050m. The effect of superimposing the strength and stability calculation on the AFS stiffness solution was to stiffen the structure, reducing its displacement to ±0.051m. The stability load factor under which each solution buckled is given in Table 6.11. The ECLIPSE (COMBEL) and (NOCOMBEL) solutions did not meet the target buckling value of 1.0, buckling instead at load factors of 0.64 and 0.66 respectively. Both AFS solutions for loop 5 and loop 6 buckled at a load factor of 2.16. Table 6.12 shows that no solution experienced von-Mises stresses greater than $1.68 * 10^8$, within the allowable stresses defined in Chapter 5.

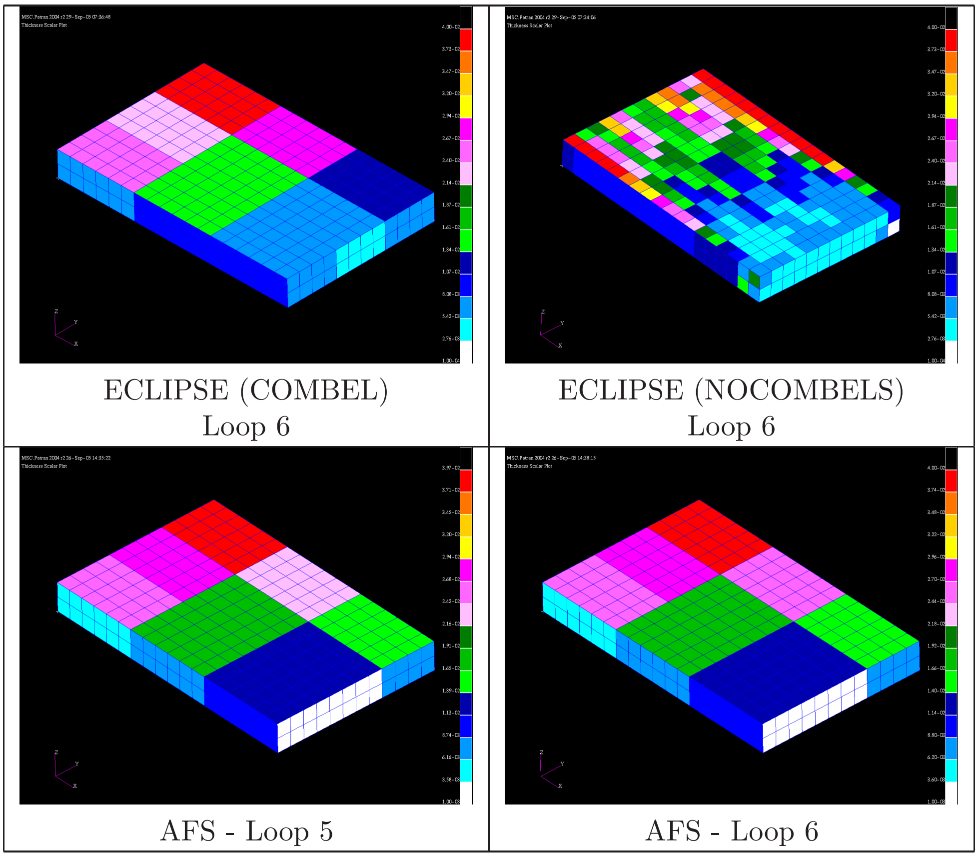



FIGURE 6.10: Thicknesses of solutions to the strength, stability and stiffness sizing problem

| **Sizing code** | **Upwards Twisting, $\delta_z$/m** | **Uniform Download $\delta_z$/m** |
|---|---|---|
| **ECLIPSE (COMBEL)** | +0.050 | -0.050 |
| **ECLIPSE (NOCOMBEL)** | +0.050 | -0.050 |
| **AFS** | +0.051 | -0.051 |

TABLE 6.3: Deflection of solutions sized for strength, stability and stiffness criteria using ECLIPSE

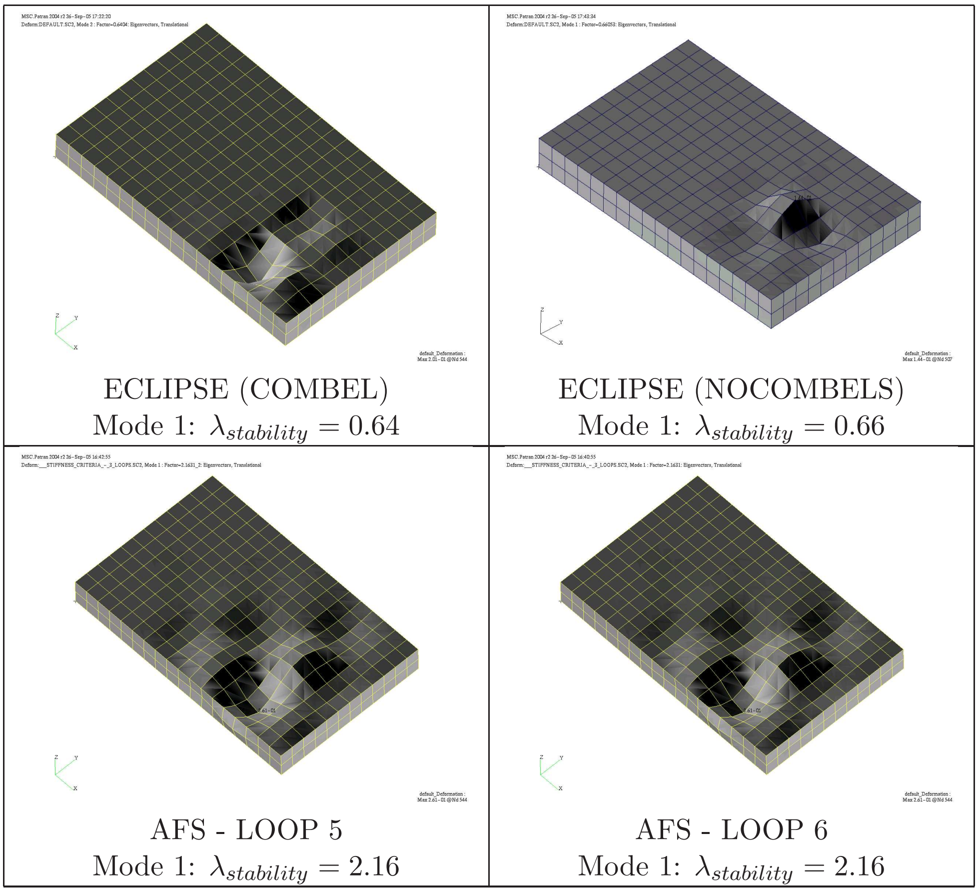



FIGURE 6.11: Verification of solution structural stability for a target value of $\lambda_{stability} = 1.00$

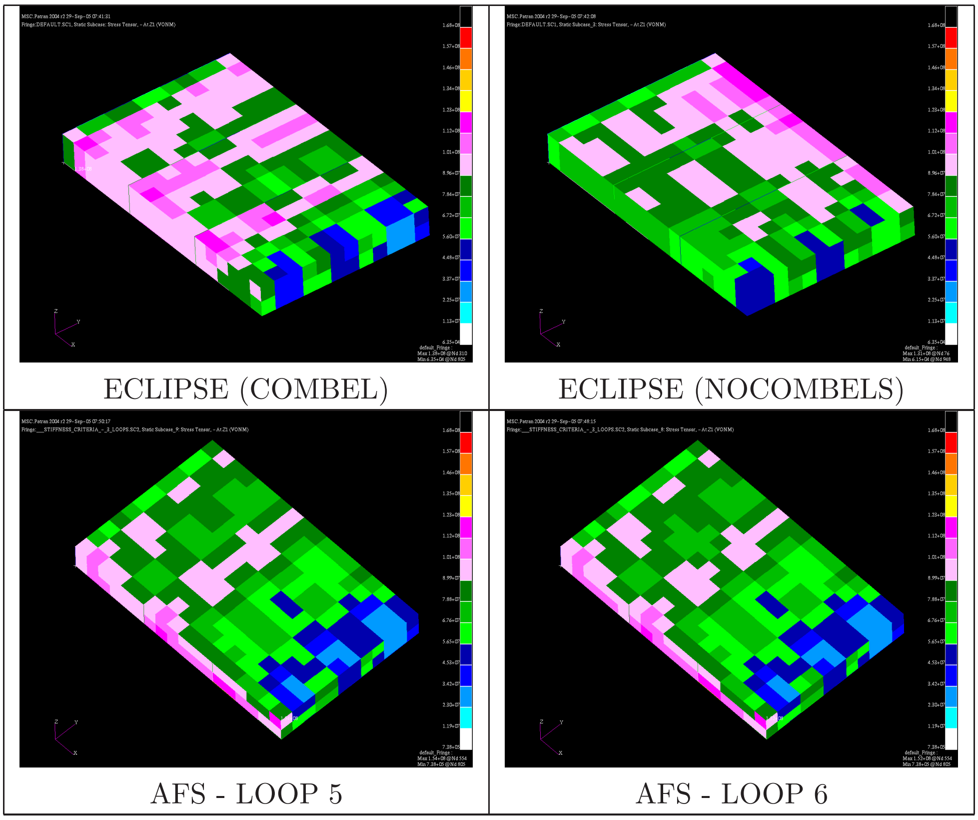



FIGURE 6.12: Verification of solution strength

# Chapter 7

# Discussion

In the introduction to this thesis the broad arguments were made for a change in the type of strategy defence companies employ to manage their capability and processes. Instead of developing the most advanced products, defence companies now need to produce advanced products affordably. This change in emphasis means the existing proprietary capability and processes are often no longer sufficient to be competitive. Moreover, the increasing degree to which 'risk' is shared between customer and supplier mean that such companies will find it harder to financially tolerate problems with the development of new products. The role of rapid design processes in providing earlier knowledge to manage product development should not be overlooked. A structural sizing process provides key mass and design information which forms the basis of the development process of the airframe structure.

## 7.1 Development and Demonstration of Sizing Technology

Chapters 4, 5 and 6 have shown examples of structural sizing capability derived from the existing toolset. 'ESO/ECLIPSE' shown in Section 4.1 wrappered the proprietary ECLIPSE code with an element sort and deletion process utilising in-house NASTRAN data access code, to size structures using a simplified version of the Evolutionary Structural Optimisation method. Test cases showed it produced similar answers to examples from the literature but that for example problems such as the wing leading-edge rib it did not converge to a manufacturable solution. Further work to improve the problem definition may have helped resolve the issues seen here. However, if a 'topological' sizing capability were to be added to ECLIPSE it would be more efficient to integrate it into the existing code to reduce the replication of post-processing of the FE model. It would also allow easier access to parameters required for analysis of constraints such as stiffness. Moreover, the optimality criterion method already within ECLIPSE already uses methods common to the homogenisation method.

Development of fixed stacking sequence and variable stacking sequence laminate sizing methods in Section 4.2 has shown the company's CITS technology used as part of sizing processes accessed through either a COM interface or as a library function. Access to the method through the ModelCenter interface allowed it to be used in MDO and structural trade studies. Coupling of the method to the genetic algorithm showed that it is also possible to obtain lighter mass solutions than the current fixed stacking sequence method in ECLIPSE.

The development of a system to size whole structures addressed a number of issues that the current sizing system faces. First the straightforward use of automatically defined FBDs demonstrated a significant reduction in the preparation time required to define the structural model to be sized. Using assumptions about the definition of panel geometry it was possible to link structural 'features' to the FE model in minutes rather than hours using the manual method currently necessary with ECLIPSE (see Section 5.4), reducing the approximate time of a user with 'basic' experience to size the UAV model from the equivalent of one man-week to half a man-week (Section 5.5). Experiences with the use of the CATIAV5 meshing capability showed that it is also possible to reduce the time required to generate detailed geometry, mesh it and size it by conducting most of the geometry definition and meshing process from within CATIAV5, although minor modifications of the mesh were still required in PATRAN. If these improvements were combined and refined even into the current process, then it is likely that the effort required to size a structure could be significantly reduced, enabling a greater number of structural configurations to be sized, thereby meeting the initial aim to reduce the resource overhead associated with the process.

Using a more explicit free-body-diagram approach to feature sizing enabled the reuse of company capability in the form of CITS. Chapter 5 showed that it is possible to size a metallic structure for strength and stability constraints by decomposing the global model into a series of local sizing problems and that the FBD method produces solutions which are within the upper and lower bounds of solutions produced by ECLIPSE. Section 6.1 and 6.2 subsequently showed that it is possible to use an optimality criterion method to include a stiffness constraint into this sizing process. Section 4.2 has shown that it is possible to link the local free-body-diagrams produced in the local sizing method in Chapter 5 to a composite panel sizing capability. To extend this method to size a composite structure for stiffness criteria would require the ability to extract strain energies for an element representing a composite material. It should be possible to extend this method to include the sizing of stringer stiffened panels as described by Ragon et al. (2003). Hypersizer/Pro has the capability both to decompose the global structural model into FBDs, size metallic and composite stiffened panels for strength and stability criteria, and link to the ModelCenter PSE through a COM interface. However, it lacks an in-built stiffness methodology and alternative strength and stiffness sizing methods would be required to confidently find minimum mass solutions. Based on the experiences with DOT/CITS the strength and stability sizer would need to be external to the PSE to ensure a reasonable response time.

The design of this system as shown in Section 5.1 means it should be possible to integrate this sizing process within an MDO process. Top-level sizing process parameters are contained within the PSE environment enabling a direct link to the CAD/CFD/FE generation process if this is automated within the PSE environment. A hierarchical structure exists between the data within the system. Detailed sizing criteria are contained in an object-orientated structural definition file enabling design criteria to be changed by accessing a single criterion rather than altering the FE model as would be required in ECLIPSE. Detailed loading information is available for each feature should this be required as part of a lower-level MDO calculation.

The development of this demonstration system has involved liaising with a number of component suppliers to understand and resolve issues. For the PSE there have been discussions on bug fixes, more efficient management of metacode and more efficient process operation. Discussions with the suppliers of NASTRAN have enabled existing issues with the ECLIPSE DMAP interface to be resolved and understood.

## 7.2 Competitive Sizing Process

The study of the current structural design process raised a number of generic issues relating to the management of capability and processes within large organisations. First, one of the recurring issues raised by engineers in study of the design lifecycle in Section 2.1, and recommended by other aerospace organisations (Section 2.3.1), was the delegation of responsibility to ensure that a manager has technical, financial and operational control over the technology and process. An example of this is the situation described by Thompson et al. (1999) in 1999, identifying, and warning of, support and availability risks of a sizing code, which had not since been mitigated or addressed in 2005. Section 2.1.5 showed that such processes and capabilities managed using an internal market in reality compete with external suppliers who have more control in managing and investing in their products. If a company is to maintain a sizing process then the internal manager of the process needs to understand what the cost of the process is, and to be able to charge a rate that allows them to invest in maintenance and upgrading of the internal process rather than just the direct cost of the work which uses it.

Second, Section 2.1.5 identified barriers to the use of structural optimisation capability already within the company. Ultimately a ‘vicious-circle’ of poor awareness and sharing of knowledge, combined with little strategic planning for investment in this type of technology, meant that its benefits were largely inaccessible across much of the lifecycle. That the company still had a sizing capability was due to the ability of individuals with clear personal visions, and technical understanding, working tactically within the company's systems to sustain it. Comments made by Thompson (Thompson et al. (1999)) show that proprietary sizing capability can only be relied upon within the current environment. The changing nature of the engineering environment effectively means that capability is transient and needs to be maintained and updated to work within that environment. Without a long-term plan such capability will be lost and with it the current process, in-house knowledge and a company's ability to structurally design minimum mass airframes. It would then be forced to react when the need to size an aircraft next arose. It is likely that the sizing capability and process acquired would not be familiar to those expected to use it. This would affect the development schedule of the product which had acquired it.

Section 2.3 noted that since different levels of product performance will be expected within different segments of the market, a 'competitive' structural sizing process is a relative concept. An airframe design organisation could take one of three approaches. A 'low-cost' approach would mean the company uses industry standard processes, whilst accepting that it might not be able to produce the most competitive structural designs if there are more advanced processes available. Industry standard processes are more likely to be commoditised and hence lend themselves to minimising the cost of the using that process. Conversely, the company could seek to clearly 'differentiate' itself in the marketplace using 'cutting-edge' structural design processes to produce structural designs that are highly competitive. In doing so it would need to accept that this process could be more expensive than the industry standard because it is less mature. A compromise approach would be to use a system that customised an industry standard process, creating a 'hybrid' process, to produce structural designs that focused on key areas of a product to improve its perceived value, whilst seeking to minimise the overall cost of the process. However, since a customised process would need to be maintained, for example by updating data extraction routines from the FE code, then the cost of the process would also have to include maintaining the custom features over its lifecycle.

### Competitive Structural Process

This work has considered the three basic structural sizing constraints of strength, stability and stiffness and assumed that these form the building blocks for a system which will also consider constraints such as aeroelastics. As well as slab panels a future sizing process will also need to size, or at least predict the mass of, equivalent stiffened panel structures. Slab panels can be modelled sufficiently well in existing sizers for both metallics and composites. However existing sizing processes for modelling stiffened panels are less mature, particularly in the case of composites. Ideally an automated sizing process for a structure to be made with stiffened panels would produce a detailed design including stiffener pitch, height and thickness, or laminate layup, as required.

Of the airframe sizing methods considered in Section 3.4 the basic optimality criterion method provides a solution for sizing slab panels, together with some basic representation of stiffened panels. These methods are mature with well established techniques that expanding their use into aeroelastic problems. However, the main disadvantage of this method, as shown in Section 6, is that it requires access to data most efficiently calculated within the FE analysis, (e.g. strain energy components for each element). Since such data is often not readily accessible in FE codes then the integration overhead of extracting this data complicates the development and maintenance of the process.

Direct optimisation methods offer the potential to link an optimisation code to the FE analysis without the need for access to such parameters. Chapter 5 included an example of NASTRAN's SOL200 direct optimiser linked to the NASTRAN linear statics analysis. In this case the optimiser found a feasible, heavier, design than most comparison methods. The more realistic problem of the UAV sizing showed the difficulties in formulating the problem such that it could be efficiently solved for models of increased computational expense and complexity. The large number of variables, in this case 103, meant multiple feasible loadpaths. This in turn implies that a direct optimiser would need to conduct a global search of the design space. However global search methods, such as genetic algorithms, typically require a large number of evaluations of the objective and constraint functions. Local gradient based searches of the design space would in themselves require objective and constraint function gradients to be calculated. Thus it is likely that some form of hybrid search would be required in order to search the relatively highly dimensional, and computationally expensive, design space, as compared to the optimality criterion methods. Further work would be required to understand how the FE model evaluations could be used more efficiently such that the problem is not prohibitively computationally expensive, how to efficiently search the global design space in general, and how to link in aeroelastic constraints into the problem. Increases in readilly available computational power will increase the feasibility of such methods.

The process proposed here has been demonstrated for strength, stability and stiffness design criteria and compared against a current in-house code. The main drawback to this particular system is the overhead of integrating a large number of components within the PSE. In reality this is unnecessary, and it should be possible to reduce the process to the key sizers and a method of implementing design rules.

The decomposition method used in this work is a compromise to reduce the computational expense of the overall process by reducing the need for highly detailed FE models and for eigenvalue buckling analyses of the whole structure. The global-local methodology allows it to be used with optimality criterion and direct optimisation methods of the global structure, whilst substituting local sizing models of the local structure. However, as the studies here have shown, these benefits are offset by the increase in the complexity of the sizing process and the maintenance effort it requires.

### Maintaining a Competitive Process

To maintain an equivalent technical capability to competitors it is necessary to define the industry benchmark, ensure that the company uses that standard and has a plan to continually meet the industry standard. Proprietary interfacing techniques such as those used for the Eurofighter Typhoon are less likely to be needed with the increased interoperability of CAD, CFD and FE codes, and the industry standard needs to be used to reduce issues such as those described in Chapter 5. It is likely that the benchmarks for design processes used to reduce the observability of an airframe will remain proprietary to retain a level of secrecy about its performance characteristics. However, the availability of COTS process integration frameworks, together with increasingly interoperable COTS CAD, CFD and FE analysis codes, will increase the potential to perform increasing amounts of the design process using a COTS toolset. Therefore it is likely that companies wishing to reduce fixed costs will move towards processes utilising COTS technology, looking to find competitive advantage from their employee's ability to understand and use the technology and processes better than their competitors.

In the specific case of sizing codes a company needs to understand what the capabilities of current COTS codes are, since these will increasingly form the basis of processes used by new-entrants into the market and competitors attempting to reduce fixed costs as described in Section 2.2. Exercises such as those by GARTEUR (ap C. Harris (1997b), ap C. Harris (1997a)) increased understanding of the competitive position of proprietary codes within the European marketplace, but as Section 2.1 showed, there can often be comparatively little understanding of the COTS optimisation capability that exists even within a company's current toolset. In addition, the levels of support available to engineers need to be understood to ensure that expertise not within the organisation can be reliably and routinely accessed.

## 7.3 Future Structural Design Process

The proposed structural design process in Chapter 5 represents a compromise between the interests of the four principle constituents in the process. By integrating different sizers in the same process the process engineer can adapt the process to meet the needs of specific projects. Moreover, integrating COTS and reused code within the process reduces the need to develop in-house code, which should reduce the cost of the process, thereby helping the process manager reduce the fixed cost of the process. However, the overhead of integrating different sizing codes has not been quantified and it is difficult to do so without developing a system to a specific set of needs. As noted in Section 2.1.5 these needs are rarely known until the project is due to commence, creating a dilemma for the process engineer. For a project engineer this process allows them to potentially use a variety of different codes available to them, including codes provided by partner organisations. Sharing a similar process, but different components, between projects would allow a project engineer to become relatively familiar with the sizing process, if not the toolset.

# Chapter 8

# Conclusions

This thesis has examined the structural sizing capabilities within the Air Systems division of BAE Systems. It has used this study to investigate possible technical enhancements and the overall technology and maintenance issues. More specifically it has proposed a structural design process aimed at increasing the amount of knowledge within a product design process in the early stages, whilst reducing the overhead of that process on the company. A demonstration system showed that the amount of manual intervention necessary in the process was reduced through the automated parameterisation of features from a structural model. This has demonstrated significant savings in the time required to size a design for strength, stability and stiffness criteria. In practice this would allow more designs to be analysed earlier in the lifecycle, possibly as part of automated multidisciplinary studies. The method used to achieve this was a decomposition approach to sizing reusing capability within the current company toolset. The free-body-diagram representation used here was shown to be a better representation of panel 'features' in a structural model than the equivalent method in ECLIPSE since it produced designs that were more likely to satisfy critical constraints such as stability.

Further development of this approach would allow the process to consist of a mix of components to size a variety of design criteria. Some limited work has been performed sizing a composite structure for strength and stability using the LAS/CITS sizer. Further work could include the inclusion of sizing criteria for stiffened panel designs using the toolset currently available within BAE, for example by using a local panel sizing calculation based on the ESDU stiffened panel calculation. In the longer term the ability of COTS sizers to perform the tasks currently carried out by proprietary software should be investigated, together with approaches for sizing high dimensional structural problems such as a structure made from stiffened panels.

A review of the current use of structural optimisation technology within a 'typical' aerospace company has shown that the trend towards integrated project teams can contribute to a 'vicious circle' of tactical investment in specialist capability and processes used within a company. This can be compounded by little sharing of knowledge between existing projects and a consequent lack of awareness of the potential benefits of a technology. The transient nature of technology means a company's constantly needs to update and maintain its toolset capabilities, thereby exacerbating the situation. Often it is only the ability of specialists within the company to understand what is required, and work tactically within the system, that capabilities can be maintained. This compensates for, but possibly masks, the problems associated with a lack of a general technical vision or strategy for specialist toolsets.

As the suppliers of computational engineering tools develop their products and work with suppliers of complementary products, they will increasingly be able to offer processes that can compete with existing in-house toolsets of major engineering manufacturers. Whilst it is likely that military aircraft will always contain some level of proprietary technology this will be confined to specific areas that offer a performance advantage to that product. Processes that are used irregularly are unlikely to remain as competitive as those that are used regularly since the body of explicit and tacit knowledge required to use them competitively will not be maintained. The study of the airframe lifecycle showed levels of knowledge were maintained because the company had ready access to its existing sizing code, which did not have a licence cost that needed to be justified yearly. It was therefore unsurprising to find that there was little enthusiasm for a COTS process by the people who had managed to maintain the existing process.

Ultimately, a competitive manufacturing company needs strategic technical leadership, with accountability and the necessary resources to maintain competitive capability, processes and people. The strategy needs to understand and value core competencies even if the understanding of these competencies is not within the leadership. All employees need to have a well developed, well understood vision of where the company wants to be in the long-term and how the company plans to get there.

## Further Work

Using the lessons from this work the next step would be to evaluate a system using COTS components for the global and local sizing and compare this to the current sizing system in terms of performance, maintenance and operating cost. Integration of the components would be likely to pose the biggest development overhead, although using the free body diagram capabilities of codes such as HyperSizer should reduce that.

Aeroelastic design criteria such as flutter would need to be considered for a full sizing system. It is likely that this would be achieved in the future through a MDO approach formally coupling the CFD and FE methods. To do this the global structural sizing method will need to be capable of sizing the model for frequency based stiffness criteria.

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# Appendix A

# Structural design software

Table A.1 gives an overview of a number of proprietary and COTS sizing codes. It lists the code, developer and FE package used. Where available it also lists the Optimality Criterion and Mathematical Programming methods used by the code as well as the design criteria which can be considered in the sizing process.

This table is based on more detailed studies given in Duysinux and Fluery (1993) and ap C. Harris (1997b). Additional information has been added from the following sources: STARS - (Bartholomew and Wellen (1990)); ASTROS - (Neill et al. (1990), Canfield and Venkayya (1990)) LAGRANGE (Schuhmacher et al. (2004)). The original names of the organisations discussed in these documents have been retained. However, RAE has been subsumed into QinetiQ; Dornier and MBB have become part of EADS; and British Aerospace is now part of BAE Systems.

**Acronyms - Optimisation Methods**

| | |
|---|---|
| BFGS | Broyden-Fletcher-Goldfard-Shanno quasi-Newton method |
| DOE | Design of experiment technique |
| EPF | Exterior penalty function |
| FR | Fletcher-Reeves |
| GRG | Generalised reduced gradient |
| IBF | Interior barrier function |
| MFD | Method of feasible directions |
| MMA | Method of moving asymptotes |
| MOM | Method of multiplicators |
| PNM | Pseudo-Newton Method |
| SCA | Sequential convex approximations |
| SED | Strain energy density method |
| SLP | Sequential linear programming |
| SIMP | Solid Isotropic Microstructure with Penalty |
| SRM | Stress ratio method |
| SQP | Sequential linear programming |

**Acronyms - Design Criteria**

| | |
|---|---|
| AER | Aeroelastic design criteria |
| STR | Strength related design criteria |
| STA | Stability (buckling) related design criteria |
| STF | Generalised deflection design criteria |

| **System** | **Developer** | **FE Package** | **OC Methods** | **MP Methods** | **Criteria** |
|---|---|---|---|---|---|
| *Proprietary Codes* | | | | | |
| OPTSYS | Saab Aerospace | ASKA / ABAQUS | | SCA (MMA) | STR, STA STF, AER |
| STARS | RAE / Deutsche Airbus | NASTRAN | SRM SED | PNM | STR, STA STF, AER |
| OPOS | Dornier | ASKA, PERMAS, SAP, BOSOL | | SLP,SQP, SCA,FR, BFGS,MFD | STR, STA STF, AER |
| LAGRANGE | MBB | NASTRAN | SRM | IBF,MOM, SLP,SQP SCA,GRG | STR, STA STF, AER |
| ELFINI | Dassault-Bregeut | ELFINI | | PCG, SLP SQP | STR, STA STF, AER |
| ECLIPSE | British Aerospace | NASTRAN | SRM, SED | SLP | STR, STA STF, AER |
| ASTROS | USAF | NASTRAN | SRM | SLP,SQP, BFGS,FR, MFD | STR, STA STF, AER |
| AS3 | Lockheed-Martin | NASTRAN | | Unknown | STR, STA |
| *Commercial-off-the-shelf codes* | | | | | |
| GENESIS | Vanderplaats Research | NASTRAN | | MFD, SLP SQP | STR, STA STF |
| Hypersizer | Collier Research | NASTRAN IDEAS | | DOE | STR, STA |
| Optistruct | ALTAIR | OWN NASTRAN | | SIMP | STR, STA STF |

Table A.1: Summary of structural sizing software capabilities, including optimality

# Appendix B

# Laminated Panel Design Criteria

This appendix describes the standard equations used to calculate panel strength and stability for the laminate panel sizing problem (Le Riche and Haftka (1993)). Material property matrices $A$ and $D$ were calculated using classical laminate theory (Zenkert (1995)).

## Panel Strength Criterion

$$\lambda_{strength} N_x = A_{11}\epsilon_x + A_{12}\epsilon_y \tag{B.1}$$

$$\lambda_{strength} N_x = A_{21}\epsilon_x + A_{22}\epsilon_y \tag{B.2}$$

$$\tag{B.3}$$

where the strains in each layer are calculated by:

$$\epsilon_1^i = cos^2\theta_i\epsilon_x + sin^2\theta_i\epsilon_y \tag{B.4}$$

$$\epsilon_2^i = sin^2\theta_i\epsilon_x + cos^2\theta_i\epsilon_y \tag{B.5}$$

$$\gamma_1^i 2 = sin^2(\epsilon_y - \epsilon_x) \tag{B.6}$$

where $A$ is the extensional stiffness matrix calculated using classical laminate theory (Zenkert (1995)).

## Panel Stability Criterion

$$\frac{\lambda_b(m,n)}{\pi^2} = \frac{[D_{11}(\frac{m}{a})^4 + 2(D_{12} + 2D_{66})(\frac{m}{a})^2(\frac{n}{b})^2 + D_{22}(\frac{n}{b})^4}{(\frac{m}{a})^2 N_x + (\frac{n}{b})^2 N_y} \tag{B.7}$$

where $D$ is the flexural stiffness matrix calculated using classical laminate theory (Zenkert (1995)).

# Appendix C

# Stiffness sizing

This section contains a derivation of the stiffness sizing method used by ECLIPSE. The derivation is based on Thompson (1999), but includes a more comprehensive derivation since it does not exist elsewhere. Care should be taken when comparing Thompson (1999) with this derivation since the notations are different.

## C.1 Derivation of optimality criterion

The purpose of this method is to minimise the mass of the structure, $M$,

$$M = \sum_{i=1}^{k} m_i \tag{C.1}$$

where $m_i$ is the mass of element $i$, subject to satisfying the equality constraints

$$h_j = C_j - C_{j_{target}} \leq 0 \tag{C.2}$$

where $C_j$ is the actual constraint value and $C_{j_{target}}$ is the target constraint value for constraint $j$ where $1 \leq j \leq p$.

Using the Lagrange multiplier method the objective function is augmented with the values of the equality constraints $h_j$

$$L(m, \lambda) = \sum_{i=1}^{k} m_i + \sum_{j=1}^{n} \lambda_j (C_j - C_{j_{target}}) \tag{C.3}$$

Assuming that the constraint value is influenced by the size of individual elements, represented here by mass $m_i$, then the stationary point will occur when

$$\frac{\partial L}{\partial m} = 1 + \sum \lambda_j \frac{\partial C_j}{\partial m_i} = 0 \tag{C.4}$$

$C_j$ is the generalised deflection value and is formed as follows:

$$\begin{aligned} C_j &= L_D^T F L_A && \text{(C.5)} \\ &= L_D^T K^{-1} L_A && \text{(C.6)} \end{aligned}$$

where $L_D$ is the unit load case, $L_A$ is the applied load case, $F$ is the flexural matrix and $K$ is the stiffness matrix ($F = K^{-1}$). Here $C_j$ is a virtual strain energy, which physically is the displacement of a given point, or points, in the direction of unit load vector $L_D$ for a structure subject to the applied load vector $L_A$.

Differentiating $C_j$ with respect to $m_i$ (and noting that the loads are assumed constant as $m_i$ varies) gives

$$\begin{aligned} \frac{dC_j}{dm_i} &= L_D^T \left( \frac{dK^{-1}}{dm_i} \right) L_A && \text{(C.7)} \\ &= -L_D^T F \left( \frac{dK}{dm_i} \right) F L_A && \text{(C.8)} \end{aligned}$$

since $\frac{dK^{-1}}{dm_i}$ can be rearranged to give $-F\left(\frac{dK}{dm_i}\right)F$ as shown:

$$\begin{aligned} \frac{dK^{-1}}{dm_i} &= \frac{d\left(KK^{-1}K^{-1}\right)}{dm_i} \\ \frac{dK^{-1}}{dm_i} &= \frac{d\left(KFK^{-1}\right)}{dm_i} \\ \frac{dK^{-1}}{dm_i} &= \frac{dK}{dm_i}FK^{-1} + K\frac{dF}{dm_i}K^{-1} + KF\frac{dK^{-1}}{dm_i} \\ 0 &= \frac{dK}{dm_i}FK^{-1} + K\frac{dF}{dm_i}K^{-1} \\ 0 &= F\frac{dK}{dm_i}F + (KK^{-1})\frac{dF}{dm_i} \end{aligned}$$

giving

$$\frac{dK^{-1}}{dm_i} = -F\frac{dK}{dm_i}F \tag{C.9}$$
$$\tag{C.10}$$

Assuming that the total stiffness matrix, $K_{TOT}$ is made from the summation of membrane, membrane-bending and bending stiffness matrices ($K_{1k}$, $K_{2k}$ and $K_{3k}$ respectively) then

$$K_{TOT} = \sum K_{1k} + K_{2k} + K_{3k} \tag{C.11}$$
$$= \sum \bar{K}_{1k}m_k + \bar{K}_{2k}m_k^2 + \bar{K}_{3k}m_k^3 \tag{C.12}$$

where $\bar{K}$ is the stiffness matrix for a unit value of mass. Using this relationship it is possible to relate a change in mass to a change in stiffness at the element level:

$$\frac{dK}{dm_i} = \bar{K}_{1k} + 2\bar{K}_{2i}m_i + 3\bar{K}_{3i}m_i^2 \tag{C.13}$$
$$= \frac{1}{m_i}(K_{1i} + 2K_{2i} + 3K_{3i}) \tag{C.14}$$
$$\tag{C.15}$$

which can therefore be related to a change in the constraint by using the relationship established in (C.10)

$$\frac{dC_j}{dm_i} = \frac{-1}{m_i}L_D^T F(K_{1i} + 2K_{2i} + 3K_{3i})F.L_A \tag{C.16}$$

or

$$\frac{dC_j}{dm_i} = \frac{-1}{m_i}U_D^T(K_{1i} + 2K_{2i} + 3K_{3i})U_A \tag{C.17}$$

where $L_D^T F$ is equivalent to the displacement vector $U_D^T$ and $FL_A$ is equivalent to the displacement vector $U_A$.

Assuming that $U_D^T(K_{1i} + 2K_{2i} + 3K_{3i})U_A = 2E_j$, where $E_j = E_{1_j} + 2E_{2_j} + 3E_{3_j}$, then

$$\frac{dC_j}{dm_i} = \frac{-2E_{ij}}{m_i} \tag{C.18}$$

and hence the optimality criterion can then be found using (C.4)

$$\frac{\partial L}{\partial m_i} = 1 + \sum \lambda_j \frac{\partial C_j}{\partial m_i} \tag{C.19}$$
$$= 1 + \sum \lambda_j \frac{-2E_{ij}}{m_i} \tag{C.20}$$

since at the stationary point of the Lagrangian $\frac{\partial L}{\partial m_i} = 0$

$$\sum \left( \lambda_i \frac{2E_{ji}}{m_i} \right) = 1 \tag{C.21}$$

which is the optimality criterion.

NB: The original displacement equation can also be expressed in terms of the energies

$$C_j = L_D^T F L_A = 2 \sum_i E_{1ij} + E_{2ij} + E_{3ij} \tag{C.22}$$

## C.2 Initial values of Lagrange multipliers, $\lambda_j$

Assuming all criteria are equally effective, and that all of the structure can be represented as a single element, then (C.21) can be written as

$$2p\lambda_j E_j = 1 \tag{C.23}$$

Using (C.22)

$$C_{j_{target}} = 2(E_{jv} + E_{jc}) \tag{C.24}$$

and

$$\frac{E_{jv}}{M_v} = \frac{C_{j_{target}} - 2E_{jc}}{2M_v} = \frac{C_{jv_{target}}}{2M_v} \tag{C.25}$$

where the subscript $v$ indicates the regions of structure that can be sized, and $c$ the regions of structure which are of fixed size. The initial mass of structure which can be sized is $M_{v0}$, corresponding to an initial criterion value of $C_{j_{v_0}}$. On meeting the target criteria $C_{jv_0} = C_{jv_{target}}$. If membrane effects are assumed dominant then an increase in $M_{v0}$ will produce a linear decrease in $C_{j_{v_0}}$. Therefore

$$M_v = \frac{C_{jv_0}}{\bar{C}_{jv_{target}}} M_{v_0} \tag{C.26}$$

Substituting (C.25) and (C.26) into (C.23) gives

$$2p\lambda_j \frac{E_j}{M_v} = 1 \tag{C.27}$$

$$\lambda_j \frac{C_{jv_{target}}}{2M_v} = \frac{1}{2p} \tag{C.28}$$

$$\lambda_j = \frac{C_{jv_0}}{p {C_{jv_{target}}}^2} M_{v0} \tag{C.29}$$

## C.3 New element sizes

The optimality criterion stated that

$$\sum \lambda_j \frac{2E_j}{m} = 1 \tag{C.30}$$

where $E_j = E_{1_j} + 2E_{2_j} + 3E_{3_j}$

It is assumed that in the 'inner loop' loads in the structure remain constant and are not re-distributed as the elements are sized. If $\alpha_i$ is the ratio of the new element sizes to their original sizes then the energy corresponding to the new sizes is:

$$E_{ji} = \frac{E_{1_{ji}}}{\alpha_i} + 2\frac{E_{2_{ji}}}{\alpha_i^2} + 3\frac{E_{3_{ji}}}{\alpha_i^3} \tag{C.31}$$

and $m_i = \alpha_i m_{i_0}$

Combining these terms with the optimality criterion (C.21) gives

$$\sum \left( 2\lambda_j \frac{\frac{E_{1_{ji}}}{\alpha_i} + 2\frac{E_{2_{ji}}}{\alpha_i^2} + 3\frac{E_{3_{ji}}}{\alpha_i^3}}{\alpha_i m_{i_0}} \right) = 1 \tag{C.32}$$

$$\sum 2\lambda_j \left( \frac{E_{1_{ji}}}{\alpha_i} + 2\frac{E_{2_{ji}}}{\alpha_i^2} + 3\frac{E_{3_{ji}}}{\alpha_i^3} \right) = \alpha_i m_{i_0} \tag{C.33}$$

$$\sum 2\lambda_j \left( E_{1_{ji}}\alpha_i^2 + 2E_{2_{ji}}\alpha_i + 3E_{3_{ji}} \right) = \alpha_i^4 m_{i_0} \tag{C.34}$$

$$\alpha_i^4 m_{i_0} - \sum 2\lambda_j \left( E_{1_{ji}}\alpha_i^2 + 2E_{2_{ji}}\alpha_i + 3E_{3_{ji}} \right) = 0 \tag{C.35}$$

Assuming that membrane-bending coupling effects can be ignored then this reduces to a quartic equation that can be solved for $\alpha_i$.

## C.4 New values of Lagrange multipliers, $\lambda_j$

A first order expansion of $C_j$ in terms of $\lambda_k$ gives

$$\Delta C_j = \sum_k \frac{dC_j}{d\lambda_k} \Delta\lambda_k \tag{C.36}$$

where $\Delta C_j$ is the change necessary in $C_j$ to meet the target constraint value of $C_{j_{target}}$.

$$\Delta C_j = C_{j_{target}} - C_j \tag{C.37}$$

$$C_{j_{target}} - C_j = \sum_k \frac{dC_j}{d\lambda_k} \Delta\lambda_k \tag{C.38}$$

$$= \sum_k \frac{dC_j}{d\lambda_k} \left( \lambda_{k_{old}} - \lambda_{k_{new}} \right) \tag{C.39}$$

which can be written in matrix form as

$$C_{j_{target}} - C_j = D \left( \lambda_{k_{old}} - \lambda_{k_{new}} \right) \tag{C.40}$$

where $D_{jk} = \dfrac{dC_j}{d\lambda_k}$ and new values of $\lambda_k$ are given by the equation

$$\lambda_{new} = \lambda_{old} + D^{-1} \left( C_{j_{target}} - C_j \right) \tag{C.41}$$

From (C.31) it can be seen that $C_j$ is a function of $\alpha_i$ and from (C.35) that $\alpha_i$ is a function of $\lambda_k$, therefore

$$\frac{dC}{d\lambda_k} = \sum \left(\frac{dC_j}{d\alpha_i}\right)\left(\frac{d\alpha_i}{d\lambda_k}\right) \tag{C.42}$$

The assumptions used to determine $\frac{dC_j}{d\alpha_i}$ must be consistent with those used to determine $C_j$. Therefore for a generalised deflection criteria

$$\frac{dC_j}{d\alpha_i} = \frac{-2}{\alpha_i}\left(\frac{E_{1_{ij}}}{\alpha_i} + \frac{2E_{2_{ij}}}{\alpha_i^2} + \frac{3E_{3_{ij}}}{\alpha_i^3}\right) \tag{C.43}$$

Differentiating the equation used to calculate $\alpha_i$ with respect to $\lambda_k$ gives

$$\frac{d\alpha_i}{d\lambda_j} = \frac{E_{1_{ij}\alpha_i^2} + 2E_{2_{ij}\alpha_i} + 3E_{3_{ij}}}{4m_{i_0} - 2\left(\sum \lambda_j E_{1_{ji}}\right)\alpha_i - 2\sum \lambda_j E_{2ij}} \tag{C.44}$$

# Appendix D

# Rib Layout Example

This appendix defines the rib structure referenced in Section 4.1. Geometry and loads were ficticious but chosen to represent a more practical example than the other tests also described in Section 4.1. Figure D.1 shows the geometry of the wing rib with selected points on the perimeter outlined in Table D.1. The main web is bordered by a flange around much of the perimeter (shown with a thicker line in Figure D.1). The initial thickness of the web was 5mm, whilst the thickness and height of the flange material was 6mm and 12mm respectively.

Figure D.2 shows the unstructured, 2-d, finite element mesh used to model the rib web. The mesh contained approximately 32,000 degrees of freedom. Elements describing the flange were kept at fixed thickness and were not deleted during the sizing process. An isotropic material with typical aluminium properties was used ($E = 72000.0N/mm^2$, $G = 27692.0N/mm^2$, $\nu = 0.3$, $\sigma_{max} = 380.0N/mm^2$, $\sigma_{min} = -380.0N/mm^2$, $\tau_{max} = 250.0N/mm^2$)

Figure D.3 shows the boundary conditions applied to the mesh. It shows that the rib was constrained in all degrees of freedom at the flanged portion of the root (indicated by the arrows). Loads from externally applied pressures of $0.5N/mm^2$ were applied over the flanges in the direction indicated by the arrows ($P_A$ and $P_B$). An evenly distributed total load, $F_B$, of 860.32N in the y-axis direction, was applied along the tip of the rib. A load vector, $F_A$, of (-300,1000,0)$N$ was applied at the bottom right corner of the rib as indicated by the arrow. A moment, $M_A$, of 100,000 $N/mm$ was also applied at this point.

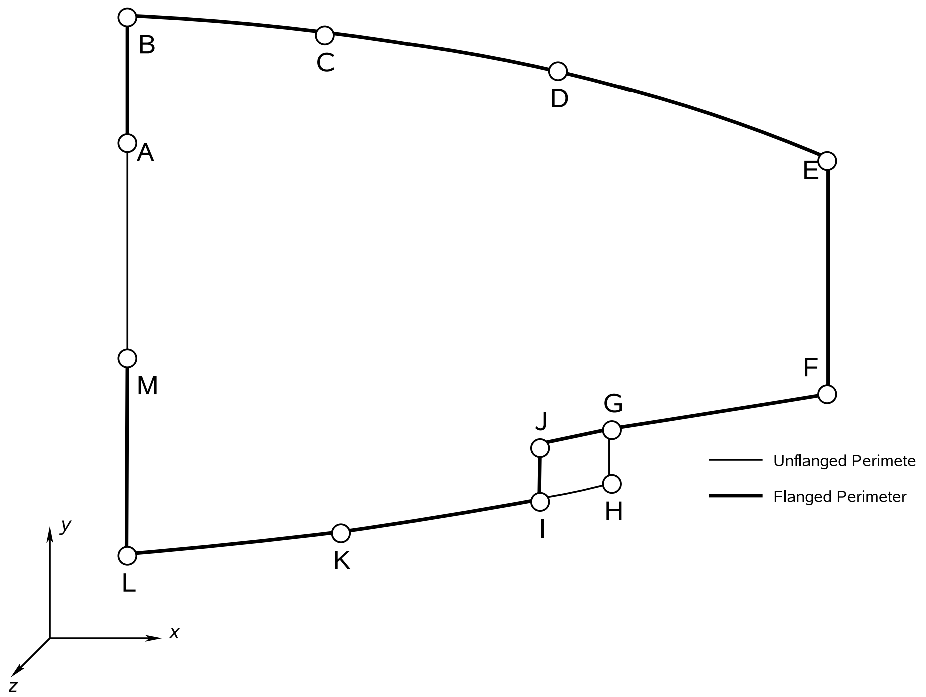


FIGURE D.1: Geometry of wing rib

| | *Location* | | | Point | *Location* | | |
|---|---|---|---|---|---|---|---|
| **Point** | **x (mm)** | **y (mm)** | **z (mm)** | **Point** | **x (mm)** | **y (mm)** | **z (mm)** |
| A | 0.0 | 255.0 | 0.0 | H | 289.2 | 25.6 | 0.0 |
| B | 0.0 | 330.0 | 0.0 | I | 248.1 | 18.9 | 0.0 |
| C | 128.4 | 311.5 | 0.0 | J | 248.5 | 53.8 | 0.0 |
| D | 278.5 | 274.7 | 0.0 | K | 119.4 | 4.6 | 0.0 |
| E | 420.0 | 218.2 | 0.0 | L | 0.0 | 0.0 | 0.0 |
| F | 420.0 | 79.7 | 0.0 | M | 0.0 | 120.0 | 0.0 |
| G | 289.4 | 59.9 | 0.0 | | | | |

TABLE D.1: Coordinates for rib geometry

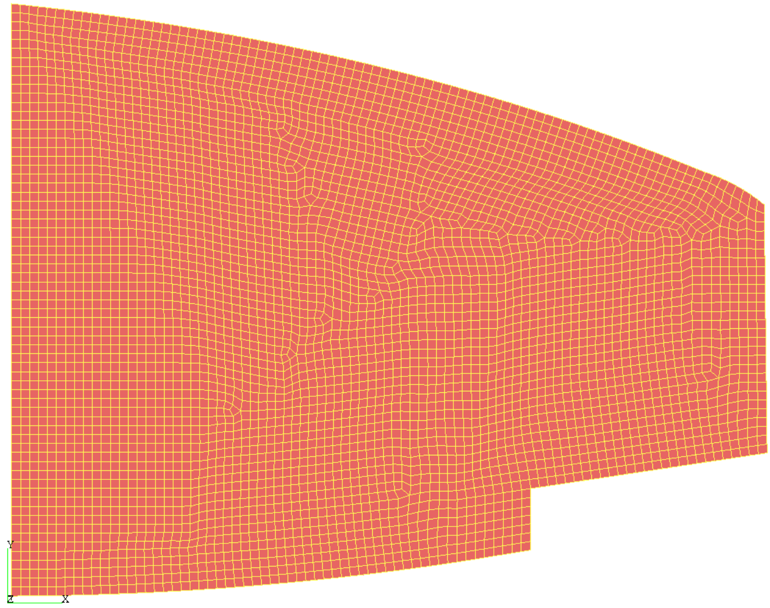


FIGURE D.2: Initial mesh domain

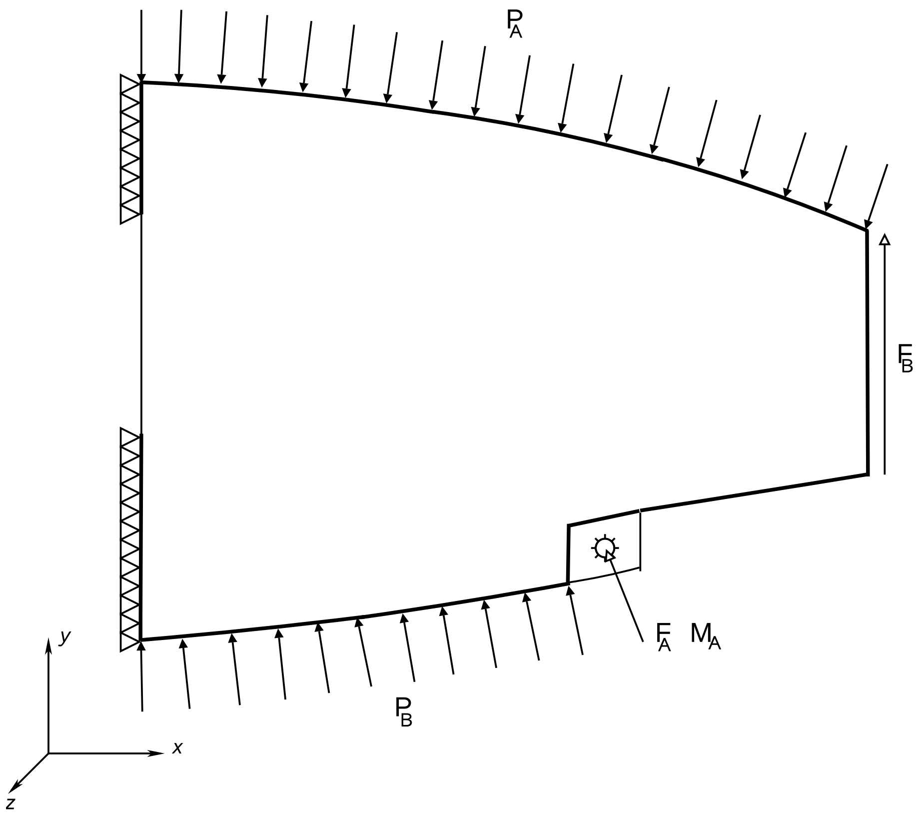


FIGURE D.3: Boundary conditions applied to rib

# Appendix E

# Structural test models

| *Name* | *Length* *a(m)* | *Width* *b(m)* | $N_{x0}$ *(N/m)* | $N_{x_b}$ *(N/m)* | $N_{y0}$ *(N/m)* | $N_{y_a}$ *(N/m)* | $q$ (N/m) |
|---|---|---|---|---|---|---|---|
| rib_2 | 2.24 | 0.381 | 1.817 | 1.817 | -3.186*10$^{4}$ | -3.186*10$^{4}$ | -3.927*10$^{4}$ |
| rib_3 | 2.24 | 0.381 | 7.976*10$^{1}$ | 7.976*10$^{1}$ | -1.668*10$^{4}$ | -1.668*10$^{4}$ | 2.148*10$^{3}$ |
| rib_4 | 2.24 | 0.381 | -5.854*10$^{4}$ | -5.854*10$^{4}$ | -1.088*10$^{5}$ | -1.088*10$^{5}$ | 1.885*10$^{5}$ |
| skin_lo_11 | 1.181 | 0.747 | -2.731*10$^{6}$ | -2.731*10$^{6}$ | -2.692*10$^{5}$ | -2.692*10$^{5}$ | 4.217*10$^{4}$ |
| skin_lo_12 | 1.181 | 0.747 | -2.593*10$^{6}$ | -2.593*10$^{6}$ | -4.208*10$^{5}$ | -4.208*10$^{5}$ | 1.495*10$^{5}$ |
| skin_lo_13 | 1.181 | 0.747 | -2.491*10$^{6}$ | -2.491*10$^{6}$ | -2.533*10$^{5}$ | -2.533*10$^{5}$ | 2.377*10$^{5}$ |
| skin_lo_21 | 1.181 | 0.747 | -1.611*10$^{6}$ | -1.611*10$^{6}$ | -6.112*10$^{4}$ | -6.112*10$^{4}$ | 2.598*10$^{5}$ |
| skin_lo_22 | 1.181 | 0.747 | -1.563*10$^{6}$ | -1.563*10$^{6}$ | -1.011*10$^{5}$ | -1.011*10$^{5}$ | 1.996*10$^{5}$ |
| skin_lo_23 | 1.181 | 0.747 | -1.550*10$^{6}$ | -1.550*10$^{6}$ | -6.160*10$^{4}$ | -6.160*10$^{4}$ | 9.671*10$^{4}$ |
| skin_lo_31 | 1.181 | 0.747 | -5.249*10$^{5}$ | -5.249*10$^{5}$ | 1.340*10$^{4}$ | 1.340*10$^{4}$ | -2.828*10$^{5}$ |
| skin_lo_32 | 1.181 | 0.747 | -5.131*10$^{5}$ | -5.131*10$^{5}$ | 1.803*10$^{4}$ | 1.803*10$^{4}$ | 1.962*10$^{5}$ |
| skin_lo_33 | 1.181 | 0.747 | -5.126*10$^{5}$ | -5.126*10$^{5}$ | 1.623*10$^{4}$ | 1.623*10$^{4}$ | -7.574*10$^{4}$ |
| skin_up_11 | 1.181 | 0.747 | 2.731*10$^{6}$ | 2.731*10$^{6}$ | 2.691*10$^{5}$ | 2.691*10$^{5}$ | -4.194*10$^{4}$ |
| skin_up_12 | 1.181 | 0.747 | 2.593*10$^{6}$ | 2.593*10$^{6}$ | 4.206*10$^{5}$ | 4.206*10$^{5}$ | 1.495*10$^{5}$ |
| skin_up_13 | 1.181 | 0.747 | 2.491*10$^{6}$ | 2.491*10$^{6}$ | 2.532*10$^{5}$ | 2.532*10$^{5}$ | -2.379*10$^{5}$ |
| skin_up_21 | 1.181 | 0.747 | 1.611*10$^{6}$ | 1.611*10$^{6}$ | 6.038*10$^{4}$ | 6.038*10$^{4}$ | -2.584*10$^{5}$ |
| skin_up_22 | 1.181 | 0.747 | 1.564*10$^{6}$ | 1.564*10$^{6}$ | 1.001*10$^{5}$ | 1.001*10$^{5}$ | -2.000*10$^{5}$ |
| skin_up_23 | 1.181 | 0.747 | 1.550*10$^{6}$ | 1.550*10$^{6}$ | 6.123*10$^{4}$ | 6.123*10$^{4}$ | -9.776*10$^{4}$ |
| skin_up_31 | 1.181 | 0.747 | 5.536*10$^{5}$ | 5.536*10$^{5}$ | -8.445*10$^{3}$ | -8.445*10$^{3}$ | 2.847*10$^{5}$ |
| skin_up_32 | 1.181 | 0.747 | 5.230*10$^{5}$ | 5.230*10$^{5}$ | -1.151*10$^{4}$ | -1.151*10$^{4}$ | -1.955*10$^{5}$ |
| skin_up_33 | 1.181 | 0.747 | 5.221*10$^{5}$ | 5.221*10$^{5}$ | -9.953*10$^{3}$ | -9.953*10$^{3}$ | 7.458*10$^{4}$ |
| spar_11 | 1.181 | 0.381 | -4.832*10$^{4}$ | -4.832*10$^{4}$ | -8.182*10$^{4}$ | -8.182*10$^{4}$ | -6.924*10$^{5}$ |
| spar_12 | 1.181 | 0.381 | -1.823*10$^{3}$ | -1.823*10$^{3}$ | 9.900*10$^{3}$ | 9.900*10$^{3}$ | -6.748*10$^{5}$ |
| spar_13 | 1.181 | 0.381 | -1.710*10$^{2}$ | -1.710*10$^{2}$ | 3.867*10$^{4}$ | 3.867*10$^{4}$ | -5.680*10$^{5}$ |
| spar_21 | 1.181 | 0.381 | -1.789*10$^{4}$ | -1.789*10$^{4}$ | -2.531*10$^{4}$ | -2.531*10$^{4}$ | -6.266*10$^{5}$ |
| spar_22 | 1.181 | 0.381 | 9.573*10$^{2}$ | 9.573*10$^{2}$ | 1.333*10$^{4}$ | 1.333*10$^{4}$ | -6.509*10$^{5}$ |
| spar_23 | 1.181 | 0.381 | 1.659*10$^{2}$ | 1.659*10$^{2}$ | 3.486*10$^{4}$ | 3.486*10$^{4}$ | -8.174*10$^{5}$ |
| spar_31 | 1.181 | 0.381 | -1.711*10$^{4}$ | -1.711*10$^{4}$ | 2.545*10$^{4}$ | 2.545*10$^{4}$ | -8.194*10$^{2}$ |
| spar_32 | 1.181 | 0.381 | 3.659*10$^{2}$ | 3.659*10$^{2}$ | 1.327*10$^{4}$ | 1.327*10$^{4}$ | -5.844*10$^{5}$ |
| spar_33 | 1.181 | 0.381 | 1.125*10$^{2}$ | 1.125*10$^{2}$ | 3.275*10$^{4}$ | 3.275*10$^{4}$ | -6.966*10$^{5}$ |
| spar_41 | 1.181 | 0.381 | -1.253*10$^{4}$ | -1.253*10$^{4}$ | -1.938*10$^{4}$ | -1.938*10$^{4}$ | -3.167*10$^{5}$ |
| spar_42 | 1.181 | 0.381 | -2.359*10$^{3}$ | -2.359*10$^{3}$ | 9.636*10$^{3}$ | 9.636*10$^{3}$ | -3.093*10$^{5}$ |
| spar_43 | 1.181 | 0.381 | -1.429*10$^{2}$ | -1.429*10$^{2}$ | 3.278*10$^{4}$ | 3.278*10$^{4}$ | -1.304*10$^{5}$ |

TABLE E.1: Panel test cases used when testing GA parameters

# Appendix F

# Files used by AFS and LAS

The following pages show standard input file used by the AFS modules.

## F.1 Input - Structural definition files

```
#VERSION: SDF20050421
//
//////////////////////////////////////////////////////////
//
// AFS Structural definition file created by AFS_setup
//
// CONSTRAINTS constraints CONSTRAINTS constraints
//
//////////////////////////////////////////////////////////
//
//
CONSTRNT: STRENGTH
CNSTNAME: STRENGTH1
RF_VALUE: 1.0
SUBCSEID: 1
DEBUGFLG: FALSE
//
//
CONSTRNT: STRENGTH
CNSTNAME: STRENGTH2
RF_VALUE: 1.0
SUBCSEID: 2
DEBUGFLG: FALSE
//
//
CONSTRNT: STABILITY
CNSTNAME: STABILITY1
RF_VALUE: 1.0
SUBCSEID: 1
DEBUGFLG: FALSE
//
//
CONSTRNT: STIFFNESS
CNSTNAME: STIFFNESS1
TARGETVL: 0.01
SUBCSEID: 1
DEBUGFLG: TRUE
//
//
CONSTRNT: STIFFNESS
CNSTNAME: STIFFNESS2
TARGETVL: 0.01
SUBCSEID: 2
DEBUGFLG: TRUE
//
//
CONSTRNT: MANUFACTURING
CNSTNAME: MANUFACTURING1
MIN_THCK: 0.001
MAX_THCK: 0.04
//
//
```

```
CONSTRNT: EXTERNAL
CNSTNAME: LAS1
SIZERMTH: LAS
EXTCONST: STR STA CONTIG
EXT_OPTS:
SUBCSEID: 1
//
//////////////////////////////////////////////////////////
//
// FEATURES features FEATURES features FEATURES features
//
//////////////////////////////////////////////////////////
//
FEATTYPE: PANEL_METALLIC
FEATNAME: skin_upperpanel_1
CNSTLIST: MANUFACTURING1 STIFFNESS1 STABILITY1 STRENGTH1
STIFFNESS2 STABILITY2 STRENGTH2
MATERIAL: METALLIC
ELEMS_E1: 148 147 146 145 156 155 154 153 164 163 162 161
NODES_E1: 43 256 255 254 112 271 270 269 169 286 285 284 238
ELEMS_E2: 161 165
NODES_E2: 238 245 252
ELEMS_E3: 165 166 167 168 157 158 159 160 149 150 151 152
NODES_E3: 252 294 295 296 183 279 280 281 126 264 265 266 57
ELEMS_E4: 152 148
NODES_E4: 57 50 43
ELEMS_EC: -1
//
[FURTHER FEATURES DEFINED HERE]
```

## F.2 Input / Output - Free body diagram files

```
# Free Body Diagram File
FBDVERSION = 20050710
NAME = testpanel.fbd
TYPE = PANEL

# Panel geometry
a = 0.508
b = 0.127
offsetangle = 0.0

# Loads   MEAN,    MIN,    MAX
Nx0 = 1750953.    0    0
Nxb = 1750953.     0    0
Ny0 = 875471.    0    0
Nya = 875471.     0    0
q = 2298.2    0    0
strainenergypercentage = 0

# Materials
# Layup
# Material - Angle - Thickness
1  0  0.01
# End Layup

# Material properties
# ID   E1  E2   G12   NU12   RHO   XC   XT   YC   YT   S
1 127.59e9 13.03e9 6.41e9 0.3 1630.0 4.2e+008 4.2e+008 4.2e+008
4.2e+008 2.56e+008
# End Material Properties
```

# Appendix G

# Solution Convergence Histories

Figures G.1 and G.2 on the following pages show the solution histories for the UAV sized for strength and stability design criteria described in Chapter 5. Figure G.3 shows the iteration history for the wingbox sized for strength, stability and stiffness criteria as described in Chapter 6.

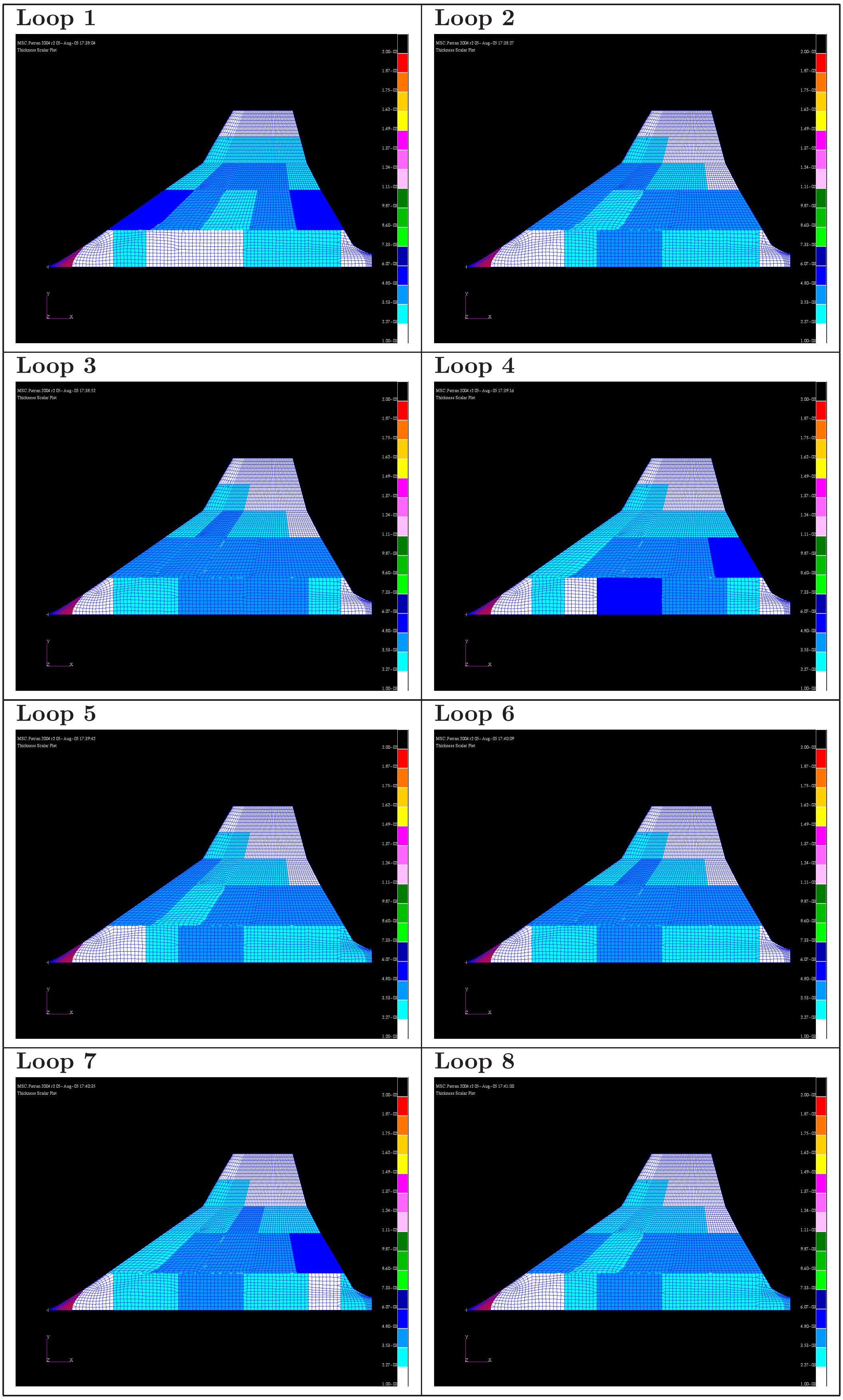


FIGURE G.1: Change of material thickness with iteration - loops 1 - 8

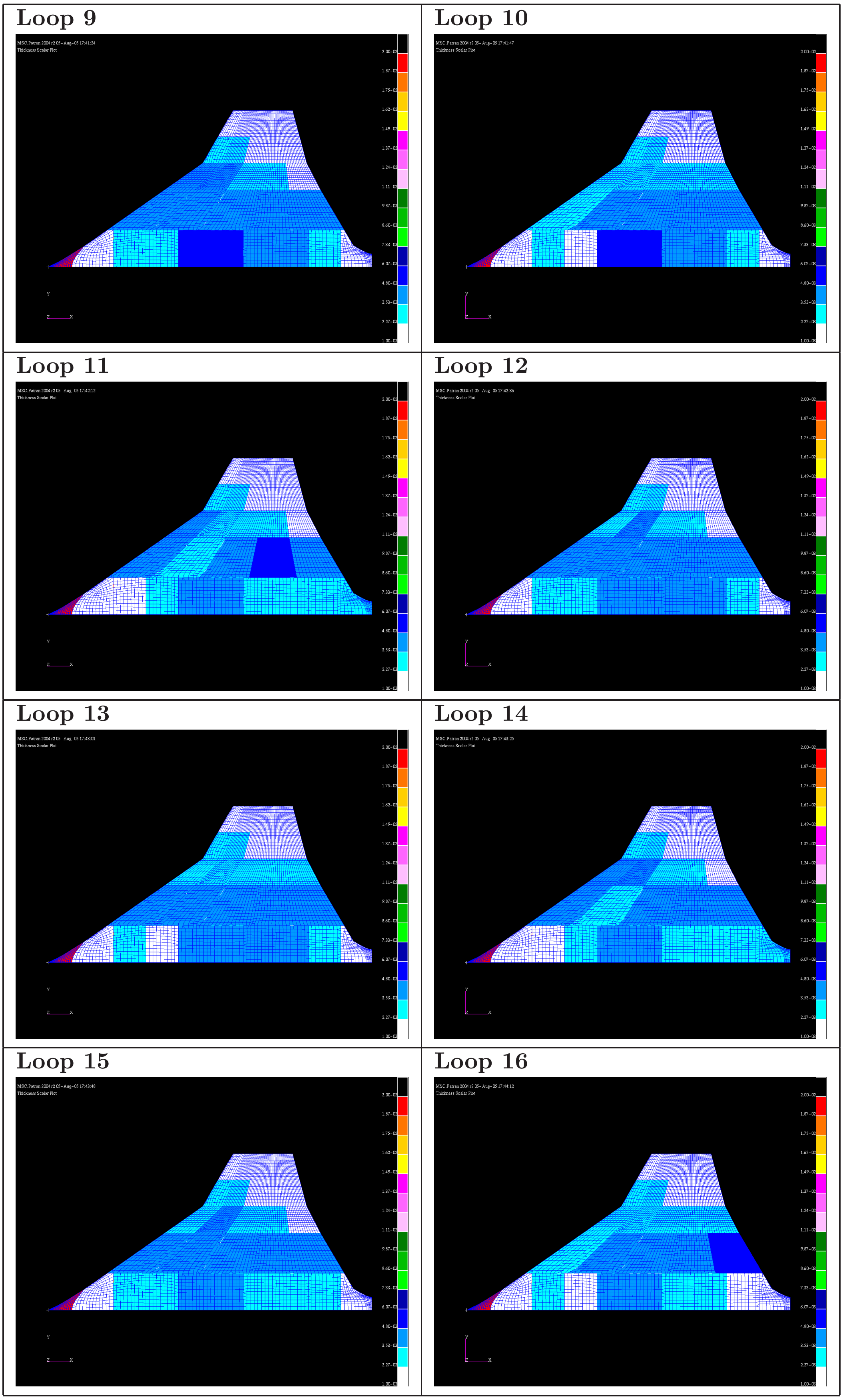


FIGURE G.2: Change of material thickness with iteration - loops 9 - 16

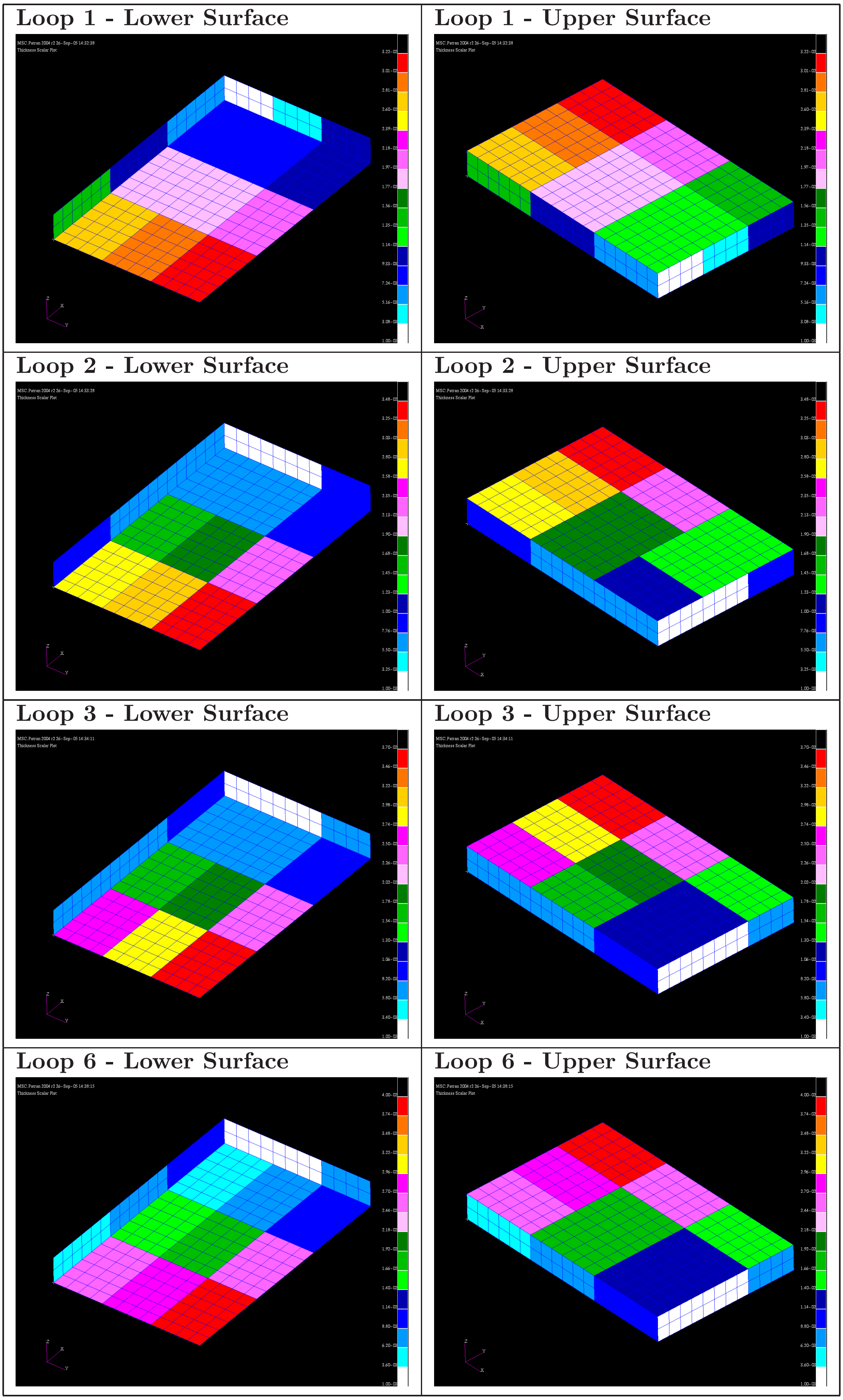


FIGURE G.3: Change of material thickness with iteration - loops 1 - 6