Medium Fidelity Aeroelastic Wingbox Optimisation
Effect of Engine Location and Sweep on Wingbox Weight

MASTER OF SCIENCE THESIS

For obtaining the degree of Master of Science in Aerospace Engineering at Delft University of Technology

S.W. Hertgers

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Faculty of Aerospace Engineering · Delft University of Technology
GRADUATION COMMITTEE

Dated: 26-11-2015

Committee chairman: ______________________________________________________________________________________
Dr. C. Kassapoglou, MSc

Committee members: ______________________________________________________________________________________
Dr.Ir. R. De Breuker

_________________________________________________________________________________________________________
Ir. J.M.A.M. Hol

_________________________________________________________________________________________________________
Ir. N.P.M. Werter
In aeroelastic research the interaction between the structural wing and the aerodynamics is investigated. Currently the wingbox structures, as other components, are manufactured out of composite materials to replace the metallic airplanes. Composites have the ability to be designed by defining the orientation and thickness of a laminate. This property can be used in a process to minimize the wingbox weight under aeroelastic loads which is called aeroelastic tailoring or optimisation. The main advantage of a lighter wing(box) design is the lower fuel consumption of the aircraft during its lifetime.

Werter and Breuker (2015) have developed the Proteus tool to perform aeroelastic tailoring on a wingbox structure. An interface is created to export the wingbox model to Nastran using ModGen (Software developed by DLR). The Nastran model is further enhanced by using the internal gradient based optimizer to tailor the wingbox structure. Goal of this thesis is twofold. First the 1D Proteus model is verified using the extension to Nastran. Secondly the influence of engine position and sweep on the optimized wingbox weight of the One Engine Reference Model (OERM) is investigated. The designed Nastran interface performs the optimisation.

In the Nastran model the skin and spars are optimized for. Ribs should be present such that the cross-section does not deform too much. The ribs themselves are not optimized. No topology optimisation is performed, the dimension and locations of spars and ribs are inputs to the model. A linear analysis is performed in Nastran. Four different analysis types are defined: aeroelastic deflection, flutter, divergence and static point deflection. The tailoring will be performed under different optimisation constraints for each type of analysis. For the first type the strains should be below the allowable and the tip rotation may not exceed 12 deg. The second and third condition refer to flutter and divergence which may not occur for the given flight condition. The last case limits for example the engine displacement during landing. The strains are also limited for this load case.

To verify the correct implementation of the model a verification procedure has been performed. For a simple rectangular beam the deformation is compared to elementary mechanics. In the next verification step a wingbox structure is considered and the results are compared to Proteus (Werter and Breuker, 2015). For similar flight conditions under a given Mach number similar results are obtained. The divergence pressure between the two models is comparable.

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To investigate the influence of engine location and sweep the One Engine Reference Model (OERM) is taken as a reference. The Proteus model predicts analogous aeroelastic behaviour compared to the Nastran model. The Proteus model therefore is verified.

For the optimisation six load cases are defined. These are derived from the flight envelope as set by airworthiness authorities. The load cases consist out of four aeroelastic flight conditions: cruise, symmetric push down and two different symmetric pull up conditions. Next a stability load case is added as no divergence and flutter may occur. The various engine locations on the OERM are optimized with and without landing load case to see the influence of landing on the engine location. The lightest wingbox design was found for an engine located around 70% of the span. This design violates the landing load case. Considering the landing load case the lightest wingbox design is found at 50% of the span. Compared to the baseline model a wingbox weight reduction of 18% is achieved. If the engine is placed further towards the tip a lot of additional material is required for the engine to not hit the runway. An engine location further down the span is preferred as the engine has a relief effect on the wing lowering the (upward) deflection and thereby lowering the critical strains. The thickness and stiffness distribution creates a tip down deformation of the wing.

Next the quarter chord sweep of the OERM is varied. For a swept forward wing the wingbox weight is higher compared to the swept backward wingbox design. Additional stiffness is required to prevent wing divergence. The optimum wingbox weight is found for a sweptback wing of 50 deg. Comparing the design to the baseline model a weight reduction of only 4.3% is achieved. However sweeping the wing comes at the cost of a higher angle of attack during cruise as the produced lift reduces for higher sweep angles. As with the optimized engine location wingbox the stiffness and thickness distribution is as such that a tip down deformation of the wing results during the aeroelastic load cases.

Performing an aeroelastic gradient based optimisation using Nastran is possible. Great caution is advised with the Nastran optimizer as the end result is dependent on the type of optimisation performed. The model can be further extended to take ribs and buckling into account. An external optimizer could be considered to have larger control over the optimisation process.
Acknowledgements

I have always shown an interest in technology. The pinnacle in my opinion is aerospace engineering. Today I still get goose bumps if I see an aircraft lifting off the earth’s surface. It is a complex piece of technology where each component needs to interact with others in a perfect way. An aircraft is a true piece of interdisciplinary engineering. Part of this interdisciplinary technology is aeroelasticity where the structure and aerodynamics interaction is studied. Because of my above average interest in multidisciplinary technology I decided to perform my MSc thesis in the Morphing and Aeroelasticity group in the Aerospace Structures and Materials department in Delft.

First of all I would like to thank my supervisors Dr.Ir. R. De Breuker and Ir. N. Werter for their daily support to finish my thesis. Likewise acknowledgment goes to the department in which I was able to do my work and ask questions at any time.

Without the encouragement of my parents I would never have been able to complete my thesis. Many thanks go to them for supporting me. I would also like to thank my sister for her support to get through the difficult days. Also friends and family that have inspired me to make the journey to become an engineer should not be forgotten. Besides those people I would like to thank everyone who made my time in Delft never to be forgotten. From start to finish I have enjoyed my life as a student and without you I would not have made it.

Now I am ready for my professional engineering career. My study is over, but I am not done learning. I am ready as an engineer to take off into the future and see where it brings me.

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26-11-2015

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## Nomenclature

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<th>Description</th>
<th>Unit</th>
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<tr>
<td>$\alpha$</td>
<td>Angle of attack</td>
<td>deg</td>
</tr>
<tr>
<td>$\alpha_{\text{camber}}$</td>
<td>Correction angle of attack to take comber into account</td>
<td>deg</td>
</tr>
<tr>
<td>$\alpha_{\text{ini}}$</td>
<td>Initial twist distribution of the wing</td>
<td>deg</td>
</tr>
<tr>
<td>$\alpha_{\text{twist}}$</td>
<td>Individual correction angle of attack for twist</td>
<td>deg</td>
</tr>
<tr>
<td>$\alpha_{\text{tip}}$</td>
<td>Local angle of attack at the tip</td>
<td>deg</td>
</tr>
<tr>
<td>$\alpha_{W2GJ}$</td>
<td>Correction angle of attack for the DLM method to take airfoil camber and twist into account</td>
<td>deg</td>
</tr>
<tr>
<td>$\Gamma$</td>
<td>Material invariant Matrix</td>
<td>GPa</td>
</tr>
<tr>
<td>$\Gamma$</td>
<td>Wing dihedral</td>
<td>deg</td>
</tr>
<tr>
<td>$\gamma_{xy}$</td>
<td>Engineering Shear strain</td>
<td>–</td>
</tr>
<tr>
<td>$\delta_{FM}$</td>
<td>Tip deflection due to a force F or moment M</td>
<td>m</td>
</tr>
<tr>
<td>$\delta_{yFM}$</td>
<td>Continues deflection in span direction due to a force F or moment M</td>
<td>m</td>
</tr>
<tr>
<td>$\delta_{xyz}$</td>
<td>Structural displacement in chord (x), span(y) or height (z) direction</td>
<td>m</td>
</tr>
<tr>
<td>$\epsilon$</td>
<td>Strain</td>
<td>–</td>
</tr>
<tr>
<td>$\theta_{FMT}$</td>
<td>Tip torsional deformation due to a force F, a moment M or a torque T</td>
<td>Nm</td>
</tr>
<tr>
<td>$\theta_{yFM}$</td>
<td>Continues torsional deformation in span direction due to a force F, moment M or torque T</td>
<td>m</td>
</tr>
<tr>
<td>$\theta_{xyz}$</td>
<td>General torsional deformation, with subscript deformation in chord (x), span(y) or height (z) direction</td>
<td>rad</td>
</tr>
<tr>
<td>$\Lambda_{c1/4}$</td>
<td>Quarter chord sweep</td>
<td>deg</td>
</tr>
<tr>
<td>$\lambda$</td>
<td>Taper ratio</td>
<td>–</td>
</tr>
<tr>
<td>$\lambda_{\text{eig}}$</td>
<td>Divergence pressure eigenvalue Nastran</td>
<td>–</td>
</tr>
<tr>
<td>$\lambda_{ij}$</td>
<td>sensitivity coefficient</td>
<td>–</td>
</tr>
<tr>
<td>$\nu_{12}$</td>
<td>Poisson’s ratio</td>
<td>–</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Material density</td>
<td>kg/m$^3$</td>
</tr>
<tr>
<td>$\rho_{\text{air}}$</td>
<td>Air density</td>
<td>kg/m$^3$</td>
</tr>
<tr>
<td>$\sigma$</td>
<td>Stress</td>
<td>N/m$^2$</td>
</tr>
</tbody>
</table>

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\[ \sigma_d \] Damping eigenvalue of a flutter system
\[ \tau \] Shear stress \[ N/m^2 \]
\[ \omega \] Eigenfrequency \[ rad/s \]
\[ \omega_f \] Flutter eigenfrequency \[ rad/s \]

**Latin Symbols**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>( A_{ij} )</td>
<td>Components of the in-plane stiffness matrix</td>
<td>( N/m )</td>
</tr>
<tr>
<td>( A_m )</td>
<td>Enclosed area by a cross-section</td>
<td>( m^2 )</td>
</tr>
<tr>
<td>( AR )</td>
<td>Aspect ratio</td>
<td>–</td>
</tr>
<tr>
<td>( B_{ij} )</td>
<td>Components of the coupling matrix</td>
<td>( N/m )</td>
</tr>
<tr>
<td>( b )</td>
<td>Wing span</td>
<td>( m )</td>
</tr>
<tr>
<td>( C_{L_a} )</td>
<td>Lift coefficient</td>
<td>–</td>
</tr>
<tr>
<td>( C_{M_0} )</td>
<td>(Aerodynamic) Moment coefficient</td>
<td>–</td>
</tr>
<tr>
<td>( c )</td>
<td>Chord</td>
<td>( m )</td>
</tr>
<tr>
<td>( c_i )</td>
<td>constraints</td>
<td>–</td>
</tr>
<tr>
<td>( c_{ref} )</td>
<td>Reference chord</td>
<td>( m )</td>
</tr>
<tr>
<td>( c_{root} )</td>
<td>Root chord</td>
<td>( m )</td>
</tr>
<tr>
<td>( D )</td>
<td>Damping Matrix</td>
<td>( kg )</td>
</tr>
<tr>
<td>( D )</td>
<td>Components of the out-of-plane stiffness matrix</td>
<td>( N/m )</td>
</tr>
<tr>
<td>( EAS )</td>
<td>Equivalent Air Speed</td>
<td>( m/s )</td>
</tr>
<tr>
<td>( E_{11} )</td>
<td>Ply longitudinal stiffness</td>
<td>( GPa )</td>
</tr>
<tr>
<td>( E_{22} )</td>
<td>Ply transverse stiffness</td>
<td>( GPa )</td>
</tr>
<tr>
<td>( EI )</td>
<td>Beam stiffness inertia</td>
<td>( Nm^2 )</td>
</tr>
<tr>
<td>( ec )</td>
<td>Distance between the aerodynamic axis of the wing and the elastic torsional axis of the wing</td>
<td>( m )</td>
</tr>
<tr>
<td>( F_{xyz} )</td>
<td>Force, subscript indicates direction</td>
<td>( N )</td>
</tr>
<tr>
<td>( G )</td>
<td>Shear stiffness</td>
<td>( Pa )</td>
</tr>
<tr>
<td>( G_{12} )</td>
<td>Ply shear stiffness</td>
<td>( GPa )</td>
</tr>
<tr>
<td>( h )</td>
<td>Height of the rectangular beam</td>
<td>( m )</td>
</tr>
<tr>
<td>( K )</td>
<td>Stiffness matrix</td>
<td>( N/m )</td>
</tr>
<tr>
<td>( K_{theta} )</td>
<td>Torsional stiffness</td>
<td>( N/m )</td>
</tr>
<tr>
<td>( L )</td>
<td>Lift</td>
<td>( N )</td>
</tr>
<tr>
<td>( L_e )</td>
<td>Beam length</td>
<td>( m )</td>
</tr>
<tr>
<td>( M )</td>
<td>Mass Matrix</td>
<td>–</td>
</tr>
<tr>
<td>( M )</td>
<td>Mass</td>
<td>( kg )</td>
</tr>
<tr>
<td>( M_{xyz} )</td>
<td>Moment, subscript denotes axis of application</td>
<td>( Nm )</td>
</tr>
<tr>
<td>( M_0 )</td>
<td>Aerodynamic Moment</td>
<td>( Nm )</td>
</tr>
<tr>
<td>( m_{fuselage} )</td>
<td>Fuselage mass</td>
<td>( kg )</td>
</tr>
<tr>
<td>( N_{ij} )</td>
<td>Distributed force. Subscript denotes direction</td>
<td>( N/m )</td>
</tr>
<tr>
<td>( Q_{ij} )</td>
<td>Ply stiffnesses</td>
<td>( GPa )</td>
</tr>
<tr>
<td>( q )</td>
<td>Dynamic pressure</td>
<td>( N )</td>
</tr>
<tr>
<td>( q_c )</td>
<td>Generalized Coordinates</td>
<td>–</td>
</tr>
<tr>
<td>( q_{div} )</td>
<td>Divergence dynamic pressure</td>
<td>( N/m^2 )</td>
</tr>
<tr>
<td>( r_j )</td>
<td>Structure response</td>
<td>–</td>
</tr>
<tr>
<td>Symbol</td>
<td>Description</td>
<td>Unit</td>
</tr>
<tr>
<td>--------</td>
<td>--------------------------------------------------</td>
<td>-------</td>
</tr>
<tr>
<td>$S$</td>
<td>Wing surface area</td>
<td>$m^2$</td>
</tr>
<tr>
<td>$T$</td>
<td>Torsion</td>
<td>$Nm$</td>
</tr>
<tr>
<td>$t$</td>
<td>Laminate thickness</td>
<td>$m$</td>
</tr>
<tr>
<td>$t_{max}$</td>
<td>Maximum allowed laminate thickness</td>
<td>$m$</td>
</tr>
<tr>
<td>$U_i$</td>
<td>Material invariants</td>
<td>$GPa$</td>
</tr>
<tr>
<td>$V$</td>
<td>Flight velocity</td>
<td>$m/s$</td>
</tr>
<tr>
<td>$V_{1234AD}$</td>
<td>Lamination Parameters</td>
<td></td>
</tr>
<tr>
<td>$V_f$</td>
<td>Velocity at which a wing starts to flutter</td>
<td>$m/s$</td>
</tr>
<tr>
<td>$w$</td>
<td>Width of the rectangular beam</td>
<td>$m$</td>
</tr>
<tr>
<td>$x$</td>
<td>Chord direction</td>
<td>$m$</td>
</tr>
<tr>
<td>$y$</td>
<td>Span direction</td>
<td>$m$</td>
</tr>
<tr>
<td>$z$</td>
<td>Height direction</td>
<td>$m$</td>
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## Definitions

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<th>Definition</th>
<th>Explanation</th>
</tr>
</thead>
<tbody>
<tr>
<td>bdf</td>
<td>Input file for Nastran (FEM analysis or optimization)</td>
</tr>
<tr>
<td>card</td>
<td>Text line in the inp (ModGen) and bdf (Nastran) file that describes a property e.g. grid point or mass property</td>
</tr>
<tr>
<td>DMI</td>
<td>Aerodynamic correction matrix produced by ModGen to take into account camber and twist</td>
</tr>
<tr>
<td>Medium Fidelity</td>
<td>Aeroelastic optimization performed by Nastran</td>
</tr>
<tr>
<td>inp</td>
<td>Input file for ModGen to create FEM elements</td>
</tr>
<tr>
<td>Low Fidelity or Proteus</td>
<td>Code as developed by Breuker et al. (2011) which performs aeroelastic tailoring in Matlab</td>
</tr>
<tr>
<td>Matlab</td>
<td>Environment developed by Mathworks for engineering programming, used for the low fidelity optimization</td>
</tr>
<tr>
<td>ModGen</td>
<td>Software developed by the DLR to create the Nastran FEM elements</td>
</tr>
<tr>
<td>Nastran</td>
<td>NASA structural analysis; FEM solver used for structural analysis and optimization</td>
</tr>
<tr>
<td>Python</td>
<td>Open source programming language, used to read the result files</td>
</tr>
<tr>
<td>rib</td>
<td>Member of an aircraft wingbox either in flow direction or perpendicular to the spars</td>
</tr>
<tr>
<td>segment</td>
<td>Structure between two neighbouring ribs</td>
</tr>
<tr>
<td>SOL 101</td>
<td>Nastran static analysis</td>
</tr>
<tr>
<td>SOL 103</td>
<td>Nastran modal (eigenfrequency) analysis</td>
</tr>
<tr>
<td>SOL 144</td>
<td>Nastran aeroelastic analysis</td>
</tr>
<tr>
<td>SOL 145</td>
<td>Nastran flutter analysis</td>
</tr>
<tr>
<td>SOL 200</td>
<td>Nastran optimization</td>
</tr>
<tr>
<td>spar</td>
<td>Structural member of the wingbox running in spanwise direction between rib elements</td>
</tr>
<tr>
<td>stringer</td>
<td>Structural element to stiffen the wingbox</td>
</tr>
<tr>
<td>wingbox</td>
<td>Main structural element in an aircraft wing</td>
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Chapter 1

Introduction

Ever since the first plane was designed engineers have tried to create lighter structures. Every kilogram saved not only reduces fuel consumption, but less structure is required to support the weight further reducing the mass. This is the well-known, positive, knock-on effect. To design the most efficient structure optimization procedures may be used to calculate the optimal structural design. Also the wings of aircraft are subjected to optimization. The optimization often leads to flexible wings which are vulnerable for aeroelastic effects. Aeroelasticity is the interaction of the structure with the surrounding air. As the structure deforms the wing deforms which changes the aerodynamic loads. In the entire flight envelope the aeroelastic effects should not hamper the safety of the flight. Airplane manufactures are currently developing airplanes, like the Airbus A350 and the Boeing 787, with more composite materials. A benefit of composite materials is the ability to tailor the properties of structures. Airplane wing structures can be tailored to make sure aeroelastic effects are minimized during flight. This field of research is better known as aeroelastic tailoring.

Werter and Breuker (2015) have developed a Matlab code called Proteus which optimizes the wing box under aeroelastic constraints. This procedure will be referred to as low fidelity as it is a simplified 1D beam model of the wingbox structure. The first goal of this thesis is to extend the Proteus model by exporting the model to Nastran. DLR (the German Aerospace Research Centre) has developed ModGen which can create a Nastran FEM 2D shell model of the wingbox. Nastran is recognized FEM software to calculate stresses and strains in aircraft structures. Due to the larger number of elements and a more accurate representation of the wingbox structure the Nastran optimization procedure will be referred to as medium fidelity. With the extension the Proteus 1D model will be verified. The second goal of the thesis is to investigate the effect of engine location and sweep on the optimization of a wingbox. As a reference the One Engine Reference Model (OERM), developed by DLR, will be used. The optimization will be performed using the internal gradient optimizer in Nastran.
First in Chapter 2 the goals of the research will be explained. The created extension and the theory behind the aeroelastic calculations in Nastran are explained in Chapter 3. Also the optimization procedure in Nastran will be explained. In Chapter 4 the various aeroelastic effects are verified. The OERM full wingbox structure is verified in Chapter 5. The result of the engine and sweep optimization is given in Chapter 6. Finally the conclusions and recommendations are given in Chapter 7.

Performing a wingbox optimization with Nastran is a delicate task. The correct combination of Nastran functions has to be used to have a wingbox model which can be tailored. In Appendix A some tips are given on how to get a functioning optimization in Nastran. Appendix B contains a detailed description of the extension code to export the Proteus model to Nastran. The results of all wingbox optimization with a varying engine location are shown in Appendix C, while the swept wingboxes are in Appendix D.
Chapter 2

Research Methodology

The objective of this research is twofold. First an interface will be developed to export a low fidelity 1D beam model developed by Werter and Breuker (2015) called Proteus to a 2D Nastran shell model. To create a Nastran model ModGen, made by DLR (German Aerospace Research Centre), will be used. ModGen uses an input file with wing parameters and structural definitions to create the shell model. The Nastran model can only deal with linear problems. This means that the a linear relation is assumed between the deformations and the strains. Hooke’s law is applied to the material with a linear relationship between strains and stresses. The material property is non-isotropic. Secondly the Nastran model will be used to investigate the influence of engine location and wing sweep on the optimization of a composite wingbox. The One Engine Reference Model (OERM) developed by DLR will be used to perform the research. Figure 2.1 shows how the research objective is broken down into various subtasks (via a so-called Work Breakdown Structure).

At the top the two objectives are given: Verification of the 1D Proteus model and the optimization of the OERM. First the 2D shell model needs to be implemented in Nastran. The calculation of aeroelastic effects like divergence and flutter in Nastran is studied. For the second objective, optimization of the OERM, the optimization module in Nastran is examined. Box two gives the multifidelity verification. First the deformation of a simple rectangular beam under a tip load is investigated to verify a correct FEM model in Proteus and Nastran. The model is expanded to a wingbox structure to verify aeroelastic effects. The OERM serves as a complex model to perform the verification on a more aircraft like wingbox structure. The Nastran software serves as verification of the 1D Proteus model. The third and final step is to optimize the wingbox with a varying engine location and quarter chord sweep. The OERM is taken as a baseline model to compare the results with. The main research question is:

*Verification of the 1D Proteus beam model by extending the model to a 2D Nastran shell model and investigating the influence of engine location and wing sweep on the optimization of the One Engine Reference Model by performing linear aeroelastic tailoring using Nastran.*

S.W. Hertgers

MSc Thesis
Chapter 1: Verification of 1D Proteus model

1. Nastran Implementation
   - Aeroelastic effects
   - Optimization

2. Multifidelity Verification
   - Rectangular Beam
   - Wingbox structure
   - OERM

Chapter 2: Optimization of OERM

3. Wingbox Optimization
   - Engine location
   - Sweep

Figure 2.1: Work Breakdown Structure
To answer the main question and to check the application of the interface, the following research questions will be investigated:

1. How to implement aeroelastic tailoring in Nastran?
   (a) How are the various aeroelastic properties like flutter and divergence calculated?
   (b) Which optimization procedure is used by Nastran?
   (c) How can tailoring be performed in Nastran?

2. Verification of the 1D Proteus beam model and the 2D Nastran shell model
   (a) Check FEM model implementation by comparison of the Proteus and Nastran model with elementary mechanics
   (b) Verify aeroelastic effects with a simple wingbox structure
   (c) How do the Proteus and Nastran model compare for the OERM (full wingbox structure)?

3. What is the best wingbox laminate design for various aircraft designs?
   (a) What is the influence of engine location on the aeroelastic tailoring of a composite wing?
   (b) How do forward swept wing optimization compare with conventional backward swept wings?
   (c) How does a landing load case influence the optimization of the wingbox structure for various engine locations?

The input parameters that are currently used in the optimization processes are: wing geometry, material properties, laminate properties, mesh size, external forces, non-structural masses (e.g. landing gear, engine, etc.) and the flight profile. The optimization objective is the wing weight. The output is the layup for skins, stiffeners and skin of the wing (Dillinger (2014) and Werter and Breuker (2015)).

The optimization is performed on laminates in a given wingbox configuration. No topology optimization is implemented. The spar and rib locations are fixed and predetermined. The performed structural analysis is linear and no gust loads are applied to the wing. As a constraint the strains in the laminates are used as well as the tip rotation. Furthermore the designed laminate has to be manufacturable. A realistic orientation and skin thickness has to be found by the optimizer.

With the extension the 1D OERM in Proteus will be exported to Nastran. The Nastran optimizer will be used to investigate a different engine location and a different quarter chord sweep. The Nastran model serves as a higher (medium) fidelity model as it is a more accurate 2D shell representation of the wing.
Chapter 3

Medium Fidelity Model Development

To create an interface between the low fidelity Matlab code and the mid fidelity Nastran software ModGen will be used. The flow of information between Matlab, ModGen and Nastran is displayed in Figure 3.1.

Matlab will be used to develop an input model for ModGen. The input file defines the wing properties and the FEM model for the wingbox which are obtained from the low fidelity model. Additional parameters to describe the high fidelity model are required. The input file is described in Section 3.1.

Secondly ModGen is run. ModGen creates multiple text files containing the FEM model. Depending on the kind of analysis these files are combined, using Matlab, into a single bdf file used as an input for Nastran. A Nastran bdf file is an execution file which contains the FEM model and type of analysis to be performed by Nastran. The user can select if a single analysis is executed or an optimisation. For the optimisation the bdf files are manipulated to perform a Nastran optimisation. This is clarified in Section 3.2. Finally Python, the most right section in Figure 3.1, is used to read the .f06 results file generated by Nastran.

To perform an optimisation extra cards need to be added which is further outlined in Section 3.3. This section also contains background information on the optimisation procedure in Nastran. Appendix A contains a detailed description of the used cards and some general tips and tricks to get a working optimisation in Nastran. In Appendix B a more extensive explanation is given on the interface and which additional parameters are required to export the low fidelity model to Nastran.
3.1 ModGen Input File

The input file for ModGen has a similar layout compared to the Nastran bdf files it creates. So-called cards define the elements to be created. The first entity of a card contains the property name, the next lines (depending on the property) define the characteristics of that specific property. Each input entity is restricted to a 8 character format. A card may be continued with on the next line. An example to create a grid point at (13,1,90) is given below:

```
1 GRID 300000 13.0 1.0 90.0
```

In the input file first the wing structure is prescribed. Grid points specify where airfoil profiles start and end. The normalized airfoil data is read from a separate text document. In between profiles segments are constructed. A segment is defined to be the wingbox region between...
3.1 ModGen Input File

two spanwise airfoil data points.
After the wing is formed, its wingbox structure is formulated. ModGen requires the spars, ribs and stringers to be described. Note that the FEM elements themselves are not present yet, but will be constructed later on. The stringers may be used to create FEM elements in chord direction. Only if FEM elements are generated for the stringer it will be existing in the analysis and optimisation.

Although the wing has been described by the airfoil profile locations the aerodynamic panels, required for the DLM analysis, are not present. After the aerodynamic panels are specified the reference axis is constructed. This axis will be used to couple the aerodynamic loads to the structure. Next masses are quantified using CONM2 elements. The masses are connected to the structure using three dimensional rigid body elements (RBE3). Thereafter the coordinate systems for the laminates are prescribed.

The following step is to set up the FEM elements for the wingbox structure. Only if requested by the user the elements are formulated. A midspar (if required) is created throughout the wing. With ModGen a midspar has to be initiated throughout the full wing. If a segment has no mid spar, the FEM elements will be removed manually before the bdf file will be submitted for analysis.

For the top and bottom skin the user can choose either to use a fixed amount of equally spaced mesh elements or to use the stringers as a reference for the meshing of the skin. Nastran requires an equal mesh for the ribs, skin and spars to be able to perform an analysis. The grid points of the mesh all elements should coincide.

The final and last step is to specify the materials and laminates. The material data is normalized by the thickness \( t \) of the laminate as will be further clarified in Section 3.3.2.

For each attribute of the input file (creation of spars, masses, aerodynamic panels, etc.) a separate function has been written in Matlab. The functions are used to convert the low fidelity wing structure to Nastran. Other functions include writing a property card to a text, inp or bdf file and a function to restrict the input to the 8 digit format.

A detailed description on how the interface works can be seen in Appendix B. It contains a description of the additional parameters required besides the 1D Proteus model to have a Nastran model. The input file structure for ModGen can be summarized as:

1. Prescribe wing shape by
   
   (a) Wing grid points consisting of leading and trailing edge xyz coordinates for each defined airfoil in the wing. The chord length is captured with this grid point definition.
   
   (b) Text files with normalized airfoil data between the above described leading and trailing edge grid points.
   
   (c) Create segments between two neighbouring airfoil inputs

2. Create wingbox by:
   
   (a) Spars, start and end point is entered as percentage of the chord.
   
   (b) Ribs, start and end point in span direction between the segments.
   
   (c) Stringers, relative position in chord direction with a direction by angle in each segment.
3. Wing planform formulation for aerodynamic mesh based on the prescribed wing.

4. Generate a load reference axis to couple the aerodynamic mesh to the structural mesh.

5. Concentrated mass elements to simulate engine and landing gear or any other mass component in the wing.

6. Coordinate system for composite material in the structural elements.

7. Meshing of the structural elements:
   
   (a) Skin
   (b) Spar
   (c) Rib
   (d) Stringer

8. Material properties of the structural elements used above.

### 3.2 Aeroelasticity in Nastran

A Nastran bdf file consists out of so-called cards, like the ModGen input file. These cards define the attributes of a certain property. Next to cards describing the FEM model, keywords are required to define the type of analysis and the desired output. The interface can construct five different type of analysis files:

1. SOL 101: Static analysis
2. SOL 103: Modal (eigenvalue) analysis
3. SOL 144: Static aeroelastic analysis
4. SOL 145: Flutter analysis
5. SOL 200: Optimisation, with as sub-cases:
   
   (a) Static
   (b) Modal
   (c) Aeroelastic
   (d) Flutter

Depending on the type of analysis the ModGen output text files are combined. Note that the optimisation can be performed for any requested analysis. The optimisation will be described in Section 3.3.

ModGen does not create any cards which define forces and moments. Therefore these cards are produced by Matlab. Besides the force cards also a clamp needs to be defined. Rigid body elements are present in the input file to relate the clamp, forces and moments to the FEM model. The user can choose between 2D rigid body elements at rib locations or 3D
3.2 Aeroelasticity in Nastran

The aeroelastic analysis requires additional trim cards. With the trim card an aeroelastic analysis can be performed for a fixed angle of attack or for a trimmed steady flight depending on the combination of free motions and restrictions used in the trim card. The free motions are described with AESTAT cards. The aeroelastic analysis also needs an AEROS card which denotes wing parameters like reference chord, span and area.

In tailoring divergence and flutter are important parameters, next to the aerodynamic loads, which determine the aeroelastic properties of the wing. Nastran uses special methods to calculate these properties which will be explained next.

3.2.1 Aerodynamic Load Computation

Nastran uses the Doublet-Lattice Method (DLM) to calculate the aerodynamic loads. In the DLM the wing is assumed to be flat. A W2GJ correction factor is applied to each individual panel to take the local camber and twist into account as can be seen in Figure 3.2. The W2GJ correction tilts each panel by $\alpha_{W2GJ}$ to correct for camber and twist. The black panels represent the panels in the DLM method. In red the rotated panels are indicated.

The twist distribution is known as a function of the span. The mid-point of the panel, indicated with a dot in Figure 3.2, is a reference to calculate the change in local angle of attack due to wing twist.

Figure 3.3 displays how the correction angle is calculated to include camber. The camber line, highlighted in blue, is partitioned along with the aerodynamic mesh elements in black. As the height from the flat plate is known the correction angle $\alpha_{W2GJ}$ can be calculated. The red line represents the corrected mesh element.

The total correction therefore is:

\[ \alpha_{W2GJ} \]
3.2.2 Divergence Prediction

In a static condition the structural deformation and the aerodynamic load cancel each other. The deformation is in equilibrium with the aerodynamic loads. A critical pressure exists at which the equilibrium does not exist anymore and structural failure can occur (Rodden, 2011). This pressure is called the divergence pressure.

A two dimensional wing, as sketched in Figure 3.4, has a torsional resistance \( K_\theta \), creates an aerodynamic moment \( M_0 \), and a lift \( L \). The distance between the aerodynamic centre of the wing and the elastic torsional axis of the wing is \( ec \). The moment equilibrium is (Megson, 2007):

\[
M_0 + Lec = \frac{1}{2}\rho_{air}V^2ScC_{M_0} + \frac{1}{2}\rho_{air}V^2S(\alpha + \theta)C_{L_0}ec = K_\theta \theta
\]  

(3.2)

\( V \) is the airspeed, \( c \) the chord length, \( S \) is the wing area and \( C_{M_0} \) and \( C_{L_0} \) are the moment and lift coefficient. For equilibrium \( \theta \) is:

\[
\alpha_{W2GJ} = \alpha_{\text{twist}} + \alpha_{\text{camber}}
\]  

(3.1)
3.2 Aeroelasticity in Nastran

Figure 3.5: Effect of sweep on the normalized divergence speed (Wright and Cooper, 2007)

\[ \theta = \frac{\frac{1}{2} \rho_{\text{air}} V^2 Sc (C_{M_0} + \alpha \varepsilon C_{L_{\alpha}})}{K_\theta - \frac{1}{2} \rho_{\text{air}} V^2 Sec C_{L_{\alpha}}} \]  

(3.3)

Divergence, when Equation 3.3 becomes unbounded, occurs if the torsional stiffness of the wing is equal to:

\[ K_\theta = \frac{1}{2} \rho_{\text{air}} V^2 Sec C_{L_{\alpha}} \]  

(3.4)

Instead of a simple two-dimensional wing, Nastran will compute the torsional stiffness of the entire structure and calculates the divergence pressure directly. Nastran uses a complex eigenvalue analysis to obtain the divergence speed. The DIVERG card is used to define the number of divergence pressures requested and the Mach number for which the divergence will be calculated. The divergence dynamic pressure \( q_{\text{div}} \) can be obtained from the results file with: (Rodden and Johnson, 2004)

\[ q_{\text{div}} = -\lambda_{\text{eig}}^2 \]  

(3.5)

As an eigenvalue analysis is performed an additional card is required to define the complex eigenvalue extraction method. The method is decided using the EIGC card. In the input the selected method should refer to this EIGC card.

Part of the research will include the investigation of swept forward wings. For swept forward wings an increase in lift results in a nose up twist (increase in \( \theta \)). The result is a lower divergence speed for swept forward wings as can be seen in Figure 3.5 (Wright and Cooper, 2007). Wings with a positive (backward) sweep are less prone to divergence.
3.2.3 Flutter Calculation

Whereas divergence is a static interaction between the aerodynamics and structure, flutter is a dynamic interaction. Flutter is the vibration of a structure due to the synergy. At the flutter speed a stable oscillatory motion exist. If the speed is increased above the flutter speed the stable motion will diverge which could lead to ultimate failure. Typically flutter occurs if two stable eigenmodes combine (Rodden, 2011).

In Nastran multiple analysis methods exist to calculate flutter. The methods available are the k and p-k method (with some sub-options). In an optimisation procedure the K-method is not available, therefore the P-K method has been chosen (Rodden and Johnson, 2004).

To calculate flutter the following equations of motion are needed:

\[
[M] \{\ddot{q}_c\} + [D] \{\dot{q}_c\} + [K] \{q_c\} = \{Q_A\} \quad (3.6)
\]

Where \(M\) is the mass matrix of the wing, \(D\) is the damping matrix, \(K\) is the stiffness matrix, \(q_c\) contains the generalized coordinates and \(Q_A\) are the generalized aerodynamic forces.

A solution is assumed where \(\{q_c\} = \{\dot{q}_c\} e^{pt}\). The solution for \(p\) is assumed to be:

\[
p = \frac{b}{U} (\sigma_d + i\omega) = \delta + ik \quad (3.7)
\]

With the assumed solution in the P-K method the aerodynamics are approximated only for the harmonic aerodynamic results. The equation of motion can be written as:

\[
\left( \frac{U^2}{b^2} [M] p^2 + \frac{U}{b} [D] p + [K] \right) \{\dot{q}_c\} = \frac{1}{2} \rho U^2 [A(k, M_{\infty}, ...)] \{q_c\} \quad (3.8)
\]

The first guess for \(p\), \(p_1\), is derived from the previous flight condition or the first guess is derived from the natural frequency of the mode. By adding a small value of damping a second guess is made for \(p_2\). For those two values of \(p\) the aerodynamic forces are calculated and the force determinants. \(p\) is updated using:

\[
p_3 = \frac{p_2 F_1 - p_1 F_2}{F_1 - F_2} \quad (3.9)
\]

This process to calculate the solution \(p\) is repeated until a converged solution is reached for \(p\) for the given flight conditions. Then the next flight condition is considered. Flutter occurs if the real part of the solution for \(p\) is positive with a positive eigenfrequency \(\omega\).

In the p-k method the aerodynamic stiffness and damping are directly coupled to the structural stiffness and damping. A key feature in the method is the iteration in the aerodynamic load based on the reduced frequency for the chosen velocity. The loads are calculated based on the oscillatory motion (Rodden, 2011). Using the p-k method not only the stability boundary can be calculated, but also an estimate is provided for the damping (Rodden and Johnson, 2004).

To reduce the degrees of freedom the Nastran calculation has to considered the user has to decide the amount of vibration modes used for the flutter analysis. The vibration modes can
be obtained from the modal analysis (SOL 103). The modal degrees of freedom need to be transformed into aerodynamic degrees of freedoms. This conversion only takes place for the entered Mach number and reduced frequencies defined in the MKAERO cards. 

As already described flutter occurs if two stable eigenmodes combine. Frequently these are the bending and twisting mode. Therefore enough eigenfrequencies should be used which include these bending and twisting node, else flutter will not be recognized by Nastran. Using the LMODES PARAM card the number of eigenfrequencies considered can be assigned. 

For the flutter analysis an extra AERO card is required with the flight conditions. Using the FLUTTER card the method is selected. The FLUTTER card requires FLFACT cards containing density ratios, Mach numbers and reduced frequencies. As the FLUTTER calculation needs a modal analysis an eigenvalue analysis method needs to be chosen. The properties in the AERO (non-steady aerodynamics) and AEROS (steady aerodynamics) card should be similar for a successful analysis.

### 3.3 Nastran Optimisation Strategy

In aeroelastic optimisation a minimum or maximum of a specific function is solved under aeroelastic constraints. The function, also called the objective, considered will be the minimum weight of the wing-box structure. In general an optimisation problem can be defined as:

\[
\begin{align*}
\text{minimize} & \quad f(x) \\
\text{subjected to} & \quad c_i(x) = 0 \\
& \quad c_i(x) \geq 0
\end{align*}
\]  

(3.10)

where \( f \) is the objective function and \( c_i \) represents the equality and inequality constraints. \( x \) represents the design variables. The design variables will be the lamination parameters to define the laminates as treated in Section 3.3.2. For the optimisation Nastran uses an interior point line search filter which is a gradient based optimizer. In a gradient based optimisation the gradient is used to define a search direction. The gradient can be computed with:

\[
\frac{df(x)}{dx} = \frac{f(x + \Delta x) - f(x)}{\Delta x}
\]  

(3.11)

The gradient used is the design sensitivity. The design sensitivity is the rate of change of the objective, or response, with respect to a change in the design variables (Rodden, 2011). The sensitivity coefficient \( \lambda \) can be mathematically described by:

\[
\lambda_{ij} = \left. \frac{\partial r_j}{\partial x_i} \right|_x
\]  

(3.12)

The subscripts \( i \) and \( j \) denote the \( j \)th response \( r \) of the structure to design variable \( i \). To calculate the response the gradient function can be used like for example the forward difference function shown in Equation 3.11. For each design variable such a gradient can be computed. As already stated Nastran will be used to perform the optimisation. Nastran uses a Interior (Line) Point Method (IPM) to perform the optimisation. Further details about the IPM will be explained in Section 3.3.3. Nastran requires special cards for the execution of a optimisation procedure as clarified in Section 3.3.1.
3.3.1 Implementation

To make the high fidelity optimisation similar to the low fidelity optimisation the strain in each element will be constrained. Compared to the low fidelity optimizer the high fidelity FEM model has more elements and is an actual 3D structure in which the strains can be calculated. Instead of a single strain value, the strain in the mid fidelity code is restricted for both the top and bottom of the laminate for each of the skin and spar elements.

The objective in Nastran is created using the DOPTPRM card. Design variables are defined with the DESVAR cards and constraints are in the DCONSTR cards. The design variables need to be related to the actual FEM model which will be optimized. This is done using the DVMREL and DVPREL cards. DVMREL relates the design variables to the material properties. The DEQATN card is used to calculated the material properties from the design variables. The DVPREL is used to relate the design variables to element properties. This card is used to describe the thickness of the laminate to the thickness of the shell element.

An overview of the relation of all Nastran optimisation cards is provided in Figure 3.6.

The FEM model however also need to communicate with the optimizer about the results of a FEM analysis. The DRESP1 converts the strains in the FEM model to values to be used in the optimizer for both the objective and the constraints. Depending on the type of analysis the model is subjected to the following constraints:

- strain on top and bottom of (PSHELL) elements
- divergence
3.3 Nastran Optimisation Strategy

- flutter
- tip twist $\alpha_{tip}$

The influence of the design on gust loads has not been investigated. Furthermore Nastran can only perform a linear analysis. In the linear analysis the flutter speed is not dependent on flight condition. Therefore only a single flutter analysis is required. A constraint exist on the tip twist to make sure the local angle of attack does not exceed the user defined maximum. For high angles of attack separation will occur, leading to a loss of lift. The Doublet- Lattice Method (DLM) which calculates the aerodynamic loads cannot take the separation into account and therefore the tip angle of attack $\alpha_{tip}$ is limited. The constraint is:

$$\alpha_{tip} = \alpha + \alpha_{ini} + \theta$$  \hspace{1cm} (3.13)

where $\alpha$ is the angle of attack, $\alpha_{ini}$ is the initial tip twist and $\theta$ is the deformation due to the aerodynamic load.

As already described Nastran uses a gradient based optimisation. To equalize the influence of the constraints on the optimisation process the constraints are normalized. The strains are normalized by dividing the constraint with the critical constraint. In Nastran only the strains on top and bottom of the laminate, in x, y and shear, can be retrieved in the optimisation process. To optimize for the mid-plane strain, the top and bottom strains are averaged. Hence a constraint is violated if the strain $\epsilon_i$ in an element:

$$-1.0 \geq \epsilon = \frac{\epsilon_t + \epsilon_b}{2} \leq 1.0$$  \hspace{1cm} (3.14)

A similar calculation is performed for the shear strain $\gamma$. The normal strain in x-direction, y-direction and in shear (xy) is optimized for.

The thickness of each shell element will determine the strains that occur in the element and the stiffness of the wing. The thickness has a direct influence on the final weight of the wingbox. As with the strains the thickness $t_i$ of the laminate is normalized by dividing the thickness with the maximum allowed thickness $t_{max}$. The thickness constraint is formulated as:

$$0 > t_i = \frac{t}{t_{max}} \leq 1.0$$  \hspace{1cm} (3.15)

The Nastran optimizer can optimize for any DRESP1 (response) card. Two goals can be specified. At the one hand the objective is weight minimization. The optimizer searches for the design with the minimum weight by varying all parameters. On the other hand a minimization of the Root Bending Moment (RBM) can be chosen which is another common objective in aeroelastic tailoring. Note that a reduction in RBM is related to a minimum weight design.

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3.3.2 Lamination Parameters

The properties of a laminate are determined by the number of layers (or plies) used and the orientation of these plies. For the optimisation these properties are described by the lamination parameters. In this section the relation between the optimisation parameters and the laminate properties is expressed.

Any material property can be described by a stress-strain relationship. For a single ply the relation between stresses and strains is characterized by:

$$
\begin{bmatrix}
\sigma_1 \\
\sigma_2 \\
\gamma_{12}
\end{bmatrix} =
\begin{bmatrix}
Q_{11} & Q_{12} & 0 \\
Q_{12} & Q_{22} & 0 \\
0 & 0 & Q_{66}
\end{bmatrix}
\begin{bmatrix}
\epsilon_1 \\
\epsilon_2 \\
\gamma_{12}
\end{bmatrix}
$$

(3.16)

$Q_{ij}$ represent the stiffness of the ply. These ply stiffnesses can be calculated using the ply longitudinal, $E_{11}$, transverse, $E_{22}$ and shear, $G_{12}$ stiffness and the Poisson’s ratio $\nu_{12}$:

$$
Q_{11} = \frac{E_{11}}{1 - \nu_{12}\nu_{21}}
$$

(3.17a)

$$
Q_{12} = \frac{\nu_{12}E_{22}}{1 - \nu_{12}\nu_{21}}
$$

(3.17b)

$$
Q_{22} = \frac{E_{22}}{1 - \nu_{12}\nu_{21}}
$$

(3.17c)

$$
Q_{66} = G_{12}
$$

(3.17d)

$$
\nu_{21} = \frac{\nu_{12}E_{22}}{E_{11}}
$$

(3.17e)

Material invariants will be used to calculate the properties of the rotated ply. The material invariants are defined as (Gurdal et al., 1999):

$$
U_1 = \frac{1}{8} (3Q_{11} + 3Q_{22} + 2Q_{12} + 4Q_{66})
$$

(3.18a)

$$
U_2 = \frac{1}{2} (Q_{11} - Q_{22})
$$

(3.18b)

$$
U_3 = \frac{1}{8} (Q_{11} + Q_{22} - 2Q_{12} - 4Q_{66})
$$

(3.18c)

$$
U_4 = \frac{1}{8} (Q_{11} + Q_{22} + 6Q_{12} - 4Q_{66})
$$

(3.18d)

$$
U_5 = \frac{1}{8} (Q_{11} + Q_{22} - 2Q_{12} + 4Q_{66})
$$

(3.18e)

In Nastran the laminate properties will be described using the ABD-matrix. With the ABD matrix the strains $\epsilon$ and curvatures $\kappa$ can be obtained in a laminate as a function of the applied loads on the laminate:
By multiplying the material invariant matrices $U_i$ with the lamination parameters $V_{iA}$ or $V_{iD}$ and the thickness $t$ the generalised $A$ and $D$ matrix can be obtained.

$$ A = t(\Gamma_0 + \Gamma_1 V_{iA} + \Gamma_2 V_{2A} + \Gamma_3 V_{3A} + \Gamma_4 V_{4A}) \tag{3.20a} $$

$$ D = \frac{t^3}{12} (\Gamma_0 + \Gamma_1 V_{1D} + \Gamma_2 V_{2D} + \Gamma_3 V_{3D} + \Gamma_4 V_{4D}) \tag{3.20b} $$

Nastran requires the thickness normalized ABD matrix as the thickness is defined by the element card, not by the material property card. To calculate the thickness normalized $A$ matrix Equation (3.20) needs to be divided by the thickness $t$. The end result to calculate the $A$ matrix including the material invariant matrices is:

$$ A = \begin{bmatrix} U_1 & U_4 & 0 \\ U_4 & U_1 & 0 \\ 0 & 0 & U_5 \end{bmatrix} + \begin{bmatrix} U_2 & 0 & 0 \\ 0 & -U_2 & 0 \\ 0 & 0 & 0 \end{bmatrix} V_{iA} + \begin{bmatrix} 0 & 0 & U_{2/2} \\ U_{2/2} & U_{2/2} & 0 \end{bmatrix} V_{2A} $$

$$ + \begin{bmatrix} U_3 & -U_3 & 0 \\ -U_3 & U_3 & 0 \\ 0 & 0 & -U_3 \end{bmatrix} V_{3A} + \begin{bmatrix} 0 & 0 & U_3 \\ U_3 & -U_3 & 0 \end{bmatrix} V_{4A} \tag{3.21} $$

The four $V_{iA}$ variables represent the (continuous) design variables with respect to the $A$ matrix. The laminates are assumed to be symmetric hence the $B$ part is zero. The $D$ matrix is multiplied with $12/t^3$ to obtain the thickness normalized $D$ matrix:

$$ D = \begin{bmatrix} U_1 & U_4 & 0 \\ U_4 & U_1 & 0 \\ 0 & 0 & U_5 \end{bmatrix} + \begin{bmatrix} U_2 & 0 & 0 \\ 0 & -U_2 & 0 \\ 0 & 0 & 0 \end{bmatrix} V_{1D} + \begin{bmatrix} 0 & 0 & U_{2/2} \\ U_{2/2} & U_{2/2} & 0 \end{bmatrix} V_{2D} $$

$$ + \begin{bmatrix} U_3 & -U_3 & 0 \\ -U_3 & U_3 & 0 \\ 0 & 0 & -U_3 \end{bmatrix} V_{3D} + \begin{bmatrix} 0 & 0 & U_3 \\ U_3 & -U_3 & 0 \end{bmatrix} V_{4D} \tag{3.22} $$

For a single laminate the design variables can be summarized as:

$$ x_i = \left\{ V_{iA} \; V_{2A} \; V_{3A} \; V_{4A} \; V_{1D} \; V_{2D} \; V_{3D} \; V_{4D} \; t \right\} \tag{3.23} $$

The laminate design variables are subjected to constraints, as not every combination yields a feasible laminate. Therefore the laminate design variables, for both the $A$ and $D$ matrix, are subjected to the following constraints (Pedersen et al., 1997):
\[ V_1^2 (1 - V_2) + 2V_3^3 (1 + V_2) + V_2^2 + V_4 - 4V_1V_3V_4 \leq 1 \]  
\[ V_1^2 + V_3^2 \leq 1 \]  
\[ -1 \leq V_i \leq 1 \]  

(3.24) \hspace{1cm} (3.25) \hspace{1cm} (3.26)

A laminate is created in Nastran using the PSHELL card. Three type of material identification types are distinguished namely: membrane, bending and membrane-bending coupling. The membrane is the A-part of the ABD matrix (MID1), bending is the D part (MID2) and membrane-bending coupling the B part of the ABD matrix (MID4). The PSHELL card also contains a thickness (t) and therefore the material properties are thickness normalized. An example of the PSHELL card is shown underneath:

```
$PSHELL<PID--><MID1--><T--><MID2->121/T3--><MID3>--><TS/T--><NSM-->
501001 100 1.000-3 101 1.0000 +
$------>--
Z1--><--Z2--><--MID4>--><------><------><------><------><------>
2 $PSHELL
501001 100 1.000-3 101 1.0000 +
3 $------>--
Z1--><--Z2--><--MID4>--><------><------><------><------><------>
4 $------>--
Z1--><--Z2--><--MID4>--><------><------><------><------><------>
```

The material is defined by the MAT2 card. The properties of this card are:

\[
\begin{bmatrix}
\sigma_x \\
\tau_{xy} \\
\sigma_y
\end{bmatrix} =
\begin{bmatrix}
G_{11} & G_{12} & G_{13} \\
G_{12} & G_{22} & G_{23} \\
G_{13} & G_{23} & G_{33}
\end{bmatrix}
\begin{bmatrix}
\epsilon_x \\
\gamma_{xy} \\
\epsilon_y
\end{bmatrix}
\]  

(3.27)

As a Nastran card the material properties are:

```
$MAT2--><MID1--><G11--><G12--><G13--><G22--><G23--><G33--><rho-->
1006.258+102.049+10 0.06.258+10 0.02.104+10 1800.0+
$------>--
ST--><--SC--><--SS-->
0.0045 0.0045 0.007
```

The material matrix properties \(G_{ij}\) can be clearly distinguished. Furthermore a density \(\rho\) is given as well as the strain limit for tensions (ST), compression (SC) and shear (SS). To relate the material properties to the design variables the DEQATN card will be used. The property \(G_{11}\) for the membrane properties \((A)\) can be calculated with:

\[ G_{11} = U_1 + U_2V_1A + 0V_2A + U_3V_3A + 0V_4A \]  

(3.28)

The other material properties are calculated in the same manner. The implementation in Nastran is shown underneath. Only the computation of \(G_{11}\) is given.

```
$DEQATN
3 Q11(V1,V3,E1,E2,nu12,G12) = E1/(1.0-nu12*nu12+(E2/E1)); +
2 + Q22-E2/(1.0-nu12*nu12+(E2/E1)); Q12-(nu12+E2)/(1.0- +
3 + nu12+nu12*(E2/E1)); U1=1.0/8.0*(3.0*Q11+3.0*Q22+2.0*Q12+ +
4 + 4.0*G12); U2=1.0/2.0*(Q11-Q22); U3=1.0/8.0*(Q11+Q22+ +
5 + 2.0*Q12-4.0*G12); G11=(U1+U2*(V1-1)+U3*(V3-1))
```
Equation 3.26 shows that each design variable for the laminate orientation is varied between -1 and 1. During the research it turned out that in Nastran it is better to have the design variables all in the positive domain. Instead of a design space from -1 to 1 a design space from 0 to 2 is used. In the equations, as shown above, one is deducted from the lamination parameters $V_i$ such that the material property calculation is still correct.

### 3.3.3 Interior Point Optimisation

Interior Point optimisation belongs to the convex optimisation class. Many engineering optimisation problems can be considered to be convex and any local optimum that will be found is also the global optimum (Yang, 2010). Nastran uses the IPOPT optimizer, which is an interior point line search filter method.

In the Interior line Point Method (IPM) the system needs to satisfy the interior-point condition. This condition is satisfied if a feasible solution exists that satisfies all constraints. The IPM also uses the gradient and a step size to go to the next design (Vial et al., 2006). Interior point methods provide an attractive strategy to deal with a large number of inequality constraints Wächter and Biegler (2006).

In the line search method barrier functions are used. A barrier function becomes unbounded or singular when the boundary of the feasibility region is approached. Although this might lead to mathematical difficulties, it can be used to impose constraints. The line search method using barrier functions belongs to the class of convex optimisation. The inequality constraints are formulated using a indicator function $I_-$: (Yang, 2010)

$$\text{minimize } f(x) + \sum_{i=1}^{N} I_-[c_i(x)]$$  \hspace{1cm} (3.29)

The IPOPT line search filter method actually optimizes for two goals, minimizing the objective function and the constraint violation $c_i(x)$ (Wächter and Biegler, 2006). In the line search method two loops exist, an inner and an outer loop. While the inner loop tries to find the best solution, the outer loop updates the solution. By using Barrier problems the optimisation problem as defined in Equation 3.10 can be rewritten as:

$$\text{minimize } tf(x) + \psi(x)$$
$$\text{subjected to } \mathbf{A}x = \mathbf{b}$$  \hspace{1cm} (3.30)

The constraints are rewritten such that they can be written in a matrix system. $\psi$ is the logarithmic barrier function to represent the inequality constraints. The matrix system can be solved in a single run. The interior-point method starts inside a feasible design and moves toward an optimal vertex (Chong and Zak, 2008). Multiple runs are required to converge towards the optimum. If the change of the objective compared to the previous run is under a user defined limit the optimizer will stop the process.

The multifidelity Nastran model has been explained in this chapter. By implementing the correct combination of Nastran cards a 2D shell model can be created in the optimisation environment. In the aeroelasticity module the strains in the wingbox are a constraint in the optimisation, while the stability constraint is implemented for divergence and flutter. The
next step is to verify the correct implementation of all theory as will be done in the next chapter. The verification will also be performed for the 1D Proteus model.
Chapter 4

Multifidelity Verification

According to Hirsch et al. (2004) verification is the process of determining if the model accurately represents the designers’ intent and if the solutions are correct. The verification will be twofold. First the mid fidelity Nastran model is compared with analytical expressions. Secondly the mid fidelity model is analysed with respect to the Proteus code.

Starting with a simple rectangular beam its shape will gradually change to represent a wing box structure. The verification of the rectangular beam is performed by a tip load comparison. The end point deflection is compared with simple mechanics. Secondly a stringer will be added, a multiple skin layup will be checked and a mid-spar is included as can be read in Section 4.1. The next step is to verify a more wing-like wingbox structure by verifying the tip deflection, the eigenmodes, the trimming properties and the influence of gravitational loads. Also the divergence will be reviewed as explained in Section 4.2.

To compare the low and medium fidelity calculation procedures it is important that a similar structure is compared. Therefore some changes were made to the low fidelity code. The low fidelity optimizer uses a text file to derive the cross-section, whereas in the mid fidelity optimizer the airfoil shape is used to define the cross-section. To make sure the wingbox shape is similar, the low fidelity optimizer has been adapted to create a cross-section in a similar fashion compared to the medium fidelity optimizer.

4.1 Rectangular Beam

For the first part of the verification the rectangular beam as given in Figure 4.1 is loaded with three separate loads: a force F, a moment M and a torque T. The deformation consisting of tip deflection and rotation will be compared with two reference models. The first is with respect to elemental mechanics. The second model is the Proteus low fidelity optimizer as developed by Werter and Breuker (2015). Thereafter the three loads are combined to create a single load case to verify the deflection and rotations for this more complex load. At last a stringer addition, a mid-spar inclusion and a multiple skin layup will be verified. The purpose
of this verification is twofold. The correct implementation of the Nastran model is proven and it serves as a verification for the 1D Proteus model.

The elementary tip deflection and tip rotation of a rectangular beam due to a tip load $F$ is (Hibbeler, 2011):

\begin{align}
\delta_F &= \frac{FL_e^3}{3EI} \\
\theta_F &= \frac{FL_e^2}{2EI}
\end{align}

where $L_e$ is the length of the beam, $EI$ is the stiffness times the inertia of the beam. Equation 4.1 only gives the tip displacement and rotation. The continues deflection $\delta_y$, and the angle $\theta_y$, of the beam as a function of $y$, the span direction, due to a tip load $F$ is:

\begin{align}
\delta_yF &= -\frac{Fy^2}{6EI} (3L_e - y) \\
\theta_yF &= -\frac{Fy}{2EI} (2L_e - y)
\end{align}

For a thin-walled rectangular beam of height $h$ and width $w$ the cross-sectional inertia $I$ is:

$$I = \frac{1}{12} th^3 + wt \left( \frac{h}{2} - \frac{t}{2} \right)^2$$

As tailoring will be performed the beam is made out of composite material. The layup considered is quasi isotropic ([0,45,-45,90]) which means that the properties are uniform in all directions. The thickness $t$ is 2.5 mm. An isotropic layup is chosen such that the analytical expressions still hold. The material properties are given in Table 4.1.
4.1 Rectangular Beam

<table>
<thead>
<tr>
<th>Strength Properties</th>
</tr>
</thead>
<tbody>
<tr>
<td>$E_{11}$ 148 GPa</td>
</tr>
<tr>
<td>$E_{22}$ 10.3 GPa</td>
</tr>
<tr>
<td>$G_{12}$ 5.9 GPa</td>
</tr>
<tr>
<td>$\nu_{12}$ 0.27</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Failure Properties</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\epsilon_{\text{max}}$ 4500 $\mu$strain</td>
</tr>
<tr>
<td>$\gamma_{\text{max}}$  7000 $\mu$strain</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Density</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\rho$ 1580 kg/m$^3$</td>
</tr>
</tbody>
</table>

Table 4.1: Material properties for the rectangular beam

<table>
<thead>
<tr>
<th>Property</th>
<th>Analytical</th>
<th>Proteus</th>
<th>Difference (%)</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>1.698</td>
<td>1.698</td>
<td>0.01</td>
<td>1.688</td>
<td>0.56</td>
</tr>
<tr>
<td>$\theta_x$ (deg)</td>
<td>9.118</td>
<td>9.081</td>
<td>0.41</td>
<td>9.046</td>
<td>0.79</td>
</tr>
</tbody>
</table>

Table 4.2: Tip deflection and rotation for a rectangular beam under a tip force $F$ of 10000 N compared to the analytical solution

The cross-sectional properties of a composite beam can be calculated with (Kassapoglou, 2010):

$$EI_i = E_i \left( \frac{w_i h_i^3}{12} + A_i d_i^2 \right)$$

(4.4)

The subscript $i$ denotes the section of the cross-section (top, bottom and sides). $E_i$ is the stiffness of each section, $w_i$ and $h_i$ are the width and height of the section. Finally $A_i$ and $d_i$ represent the area of the section and the distance to the neutral axis. $w_i$, $h_i$ and $d_i$ are dependent on the axis at which the load is applied. As sketched in Figure 4.1 the beam will be loaded in $z$ direction. The stiffness $E_i$ is directly derived from the ABD matrix:

$$E_i = \frac{1}{t a_{11}}$$

(4.5)

t represents the thickness of the laminate, while $a_{11}$ is a property obtained from the inverse ABD matrix. For a simple rectangular beam of height 0.46 m and width 0.4 m the tip deflection and tip rotation of the various models is given in Table 4.2.

Figure 4.2 shows the deflection $\delta_z$ and rotation $\theta_x$ along the span direction. The blue squares represent the Proteus low fidelity solution. The mid fidelity Nastran result is given by the red triangles. The analytical deflection and rotation as calculated with Equation 4.2 is represented by the black line. Both Table 4.2 and Figure 4.2 show perfect compliance with the analytical solution for the applied tip load $F$ with a difference below 1%. Note that the results are given for a converged mesh.
Figure 4.2: Verification of a rectangular composite beam under a tip load $F$ of 10000 N

<table>
<thead>
<tr>
<th>Property</th>
<th>Analytical</th>
<th>Proteus</th>
<th>Difference (%)</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>0.080</td>
<td>0.079</td>
<td>0.41</td>
<td>0.079</td>
<td>0.79</td>
</tr>
<tr>
<td>$\theta_x$ (deg)</td>
<td>0.570</td>
<td>0.568</td>
<td>0.41</td>
<td>0.565</td>
<td>0.79</td>
</tr>
</tbody>
</table>

Table 4.3: Tip deflection and rotation for a rectangular beam under a tip moment $M$ of 5000 Nm compared to the analytical expression

Moment $M$

Similarly a tip moment $M_x$ can be applied to the tip of the beam. The tip deflection and rotation is (Hibbeler, 2011):

\[
\delta_M = \frac{ML_x^2}{2EI} \quad (4.6a)
\]

\[
\theta_M = \frac{ML_x}{EI} \quad (4.6b)
\]

Identical variables appear as with the tip deflection due to a force $F$ as presented in Equation 4.1. As with the tip force the deflection and rotation can be expressed as a continues function of the span coordinate $y$:

\[
\delta_{yM_x} = \frac{M_x y^2}{2EI} \quad (4.7a)
\]

\[
\theta_{yM_x} = \frac{M_x y}{EI} \quad (4.7b)
\]

For a similar dimensioned beam with a tip moment of 5000 Nm the tip deflection and rotation are given in Table 4.3.

Figure 4.3 shows the deflection as a function of span. As with the tip force $F$ the blue squares show the low fidelity solution, the red triangles give the mid fidelity solution and the black
Figure 4.3: Verification of a composite beam under a tip bending moment of 5000 Nm

<table>
<thead>
<tr>
<th>Property</th>
<th>Analytical</th>
<th>Proteus</th>
<th>Difference (%)</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\theta_y$ (deg)</td>
<td>0.556</td>
<td>0.557</td>
<td>0.04</td>
<td>0.556</td>
<td>0.00</td>
</tr>
</tbody>
</table>

Table 4.4: Tip rotation for a rectangular beam under a tip torque $T$ of 2500 Nm compared to the analytical result

Torsion T
The third and final load that is applied is torsion. The rotation as a function of $y$ can be analytically calculated with (Megson, 2007):

$$\frac{\theta_T}{dy} = \frac{T}{4A_m^2G} \int \frac{ds}{t}$$  \hspace{1cm} (4.8)

where $A_m$ is the enclosed area, $G$ is the shear stiffness of the used material and the integral represents the length $s$ of each section divided by the thickness $t$. The tip rotation is

$$\theta_T = \frac{TL_e}{4A_m^2G} \int \frac{ds}{t}$$  \hspace{1cm} (4.9)

The shear stiffness $G$ is calculated with:

$$G = \frac{1}{\lambda_{3,3}}$$  \hspace{1cm} (4.10)

For a tip torque of 2500 Nm the tip rotation is given in Table 4.4 and the spanwise variation of the rotation as a function of span is given in Figure 4.4. Also for the tip torque the models predict the same tip rotation.
Combined Load

The last step is to combine the previous loads in a single load case. Using the superposition principle the analytical expression for tip deflection and rotations are combined to calculate the tip deformations. The result is given in Table 4.5. Figure 4.5 displays the deformation curves of the low fidelity and medium fidelity model. The applied force is 10000 N, the moment is 5000 Nm and the torque T is 2500 Nm.

In Figure 4.5 the continuous analytical solution is not present anymore. From Table 4.5 it can be seen that the tip deflection and rotations are within 0.5% of the analytical model. Hence the Nastran medium fidelity model for an isotropic beam is implemented correctly for static load cases as well as the 1D Proteus beam model.

Before aeroelastic loads will be applied and other aeroelastic effects are investigated for a wing-box cross-section first the effect of stringers, mid-spars and a multiple layup in chord direction is investigated.

Table 4.5: Tip deflection and rotations for a rectangular beam under a combined load compared to the analytical solution

<table>
<thead>
<tr>
<th>Property</th>
<th>Analytical</th>
<th>Proteus</th>
<th>Difference (%)</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>1.777</td>
<td>1.777</td>
<td>0.01</td>
<td>1.767</td>
<td>0.57</td>
</tr>
<tr>
<td>$\theta_x$ (deg)</td>
<td>9.688</td>
<td>9.649</td>
<td>0.41</td>
<td>9.611</td>
<td>0.79</td>
</tr>
<tr>
<td>$\theta_y$ (deg)</td>
<td>0.556</td>
<td>0.557</td>
<td>0.09</td>
<td>0.557</td>
<td>0.04</td>
</tr>
</tbody>
</table>
4.1 Rectangular Beam

Figure 4.5: Deformation curves for a combined load case of $F = 10000 \text{ N}$, $M = 5000 \text{ Nm}$ and $T = 2500 \text{ N}$
4.1.1 Stringer

In the design optimization stringers can be added to the wing-box. Stringers will make the wing-box more stiff resulting in a lower bending. Equivalent stringer properties will be used. The stringer is modelled as a simple straight section with a certain height and thickness. As already described in Section 3.3.2 the stresses in a laminate can be calculated using the ABD matrix. Due to the addition of stringers to the laminate both the A part and D part of the matrix will change, influencing the strains and stresses in the laminate. According to (Kassapoglou, 2010) $A_{11}$, $D_{11}$ and $D_{66}$ are influenced due to the stringer addition. The optimizer only calculates the equivalent properties of the skin. For the optimizer it does not matter if a laminate is used to obtain the calculated ABD values, or if a stringer stiffened laminate is used. The equivalent skin properties are calculated. Because the equivalent properties are calculated the influence of stringer addition is hard to research. However a separate stringer definition is created to have the possibility to add stringers to both the low and medium fidelity model and to investigate the influence it has. Due to the stringer addition the resulting skin thickness in the optimization will be more realistic. In the low fidelity code the stringer is modelled as an extra section with a reference to a laminate material. For the analytical calculation Equation 4.4 the stringer element is added as it is a simple extra section i. Figure 4.6 shows the rectangular beam with 8 stringers applied to the top and bottom skin. Simple straight stringers are used of 50 mm with a thickness of 1 mm.

To model the stringer in Nastran CBAR elements are created with a given property PBAR. PBAR refers to a material used for the stringer. Only the MAT1 card is accepted for the PBAR element, therefore the stringer material has to be homogeneous in the mid fidelity model. The low fidelity model can accept inhomogeneous anisotropic laminates. As with the
4.1 Rectangular Beam

<table>
<thead>
<tr>
<th>Property</th>
<th>Analytical</th>
<th>Proteus</th>
<th>Difference (%)</th>
<th>Nastran</th>
<th>Difference (%)</th>
<th>Nastran No stringer</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z \ (m)$</td>
<td>1.371</td>
<td>1.365</td>
<td>0.43</td>
<td>1.367</td>
<td>0.31</td>
<td>1.777</td>
</tr>
<tr>
<td>$\theta_x \ (\text{deg})$</td>
<td>7.475</td>
<td>7.424</td>
<td>0.69</td>
<td>7.429</td>
<td>0.62</td>
<td>9.688</td>
</tr>
<tr>
<td>$\theta_y \ (\text{deg})$</td>
<td>0.556</td>
<td>0.556</td>
<td>0.08</td>
<td>0.557</td>
<td>0.03</td>
<td>0.556</td>
</tr>
</tbody>
</table>

**Table 4.6:** Tip deflection and rotations for a rectangular beam with stringers under a combined load

A rectangular beam a tip load will be applied to verify the results. A tip force of 10000 N is applied, with a tip moment of 5000 Nm and a torsional load of 2500 Nm. For the analytical calculation Equation 4.4, 4.6 and 4.8 are used where the stringer is an additional section i. The results are given in Table 4.6. The deflection curves are shown in Figure 4.7.
As can be seen in Table 4.6 and Figure 4.7 the results of the models are similar to the analytical model. Stringers make the cross-section stiffer due to the additional material which contributes to the inertia of the beam. So compared to the previous verification case, which did not have any stiffeners, the deflection and rotations are expected to be smaller. The dotted line in Figure 4.7 and the last column of Table 4.6 shows the wingbox without stringers. The deflection and bending angle are 23.0% lower. The rotation angle of the beam has remained unchanged. Stiffeners do not contribute to the torsional stiffness of the beam as described by Equation 4.9 as the enclosed area is not enlarged nor the shear flow expression has changed (integral expression of Equation 4.9).
4.1 Rectangular Beam

<table>
<thead>
<tr>
<th>Property</th>
<th>Analytical</th>
<th>Proteus</th>
<th>Difference (%)</th>
<th>Nastran</th>
<th>Difference (%)</th>
<th>No mid spar</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>1.560</td>
<td>1.561</td>
<td>0.05</td>
<td>1.551</td>
<td>0.55</td>
<td>1.777</td>
</tr>
<tr>
<td>$\theta_x$ (deg)</td>
<td>8.504</td>
<td>8.479</td>
<td>0.29</td>
<td>8.444</td>
<td>0.70</td>
<td>9.688</td>
</tr>
<tr>
<td>$\theta_y$ (deg)</td>
<td>0.556</td>
<td>0.557</td>
<td>0.08</td>
<td>0.557</td>
<td>0.03</td>
<td>0.556</td>
</tr>
</tbody>
</table>

Table 4.7: Tip deflection and rotations for a rectangular beam with mid spar under a combined load

4.1.2 Mid Spar

The mid-spar can been seen as an additional long stringer connecting the top and bottom skin. A mid-spar is often added at the root to provide extra rigidity to the wing-box structure. The applied tip load is similar to the previous verification case. Both the deflection and rotations agree with the analytical calculation as shown in Table 4.7. Figure 4.8 shows the deformation and rotation of the low and mid fidelity model. The dotted line represent the wingbox without mid spar. Again comparing the rectangular beam with and without mid spar the rectangular beam with mid spar has a lower deflection due to the higher stiffness as is expected. The tip displacement has been reduced with 12.4%. The rotation is similar as a mid spar does not contribute to the shear stiffness of the beam.
4.1.3 Multiple Skin Layup

ModGen creates laminates in segments between ribs and (mid-) spars. An extension has been developed such that the laminate can be changed in chord direction for any number of design fields or laminates. To verify the extension the rectangular beam will have a changing laminate for the second half of the top and bottom skin as shown in Figure 4.9. Instead of a 2.5 mm skin thickness the region highlighted in red has a thickness of 5.0 mm.

As with the previous cases a similar tip load is applied with a force of 10000 N, a moment of 5000 Nm and a torque of 2500 Nm. The deformation results are given in Table 4.8 and Figure...
4.1 Rectangular Beam

Figure 4.9: Rectangular beam with a changing laminate in chord direction. Thickness in m.

<table>
<thead>
<tr>
<th>Property</th>
<th>Analytical</th>
<th>Proteus</th>
<th>Difference (%)</th>
<th>Nastran</th>
<th>Difference (%)</th>
<th>No variation</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>1.314</td>
<td>1.332</td>
<td>1.38</td>
<td>1.325</td>
<td>0.87</td>
<td>1.777</td>
</tr>
<tr>
<td>$\theta_x$ (deg)</td>
<td>7.161</td>
<td>7.226</td>
<td>0.90</td>
<td>7.203</td>
<td>0.58</td>
<td>9.688</td>
</tr>
<tr>
<td>$\theta_y$ (deg)</td>
<td>0.492</td>
<td>0.502</td>
<td>2.17</td>
<td>0.502</td>
<td>2.21</td>
<td>0.556</td>
</tr>
</tbody>
</table>

Table 4.8: Tip deflection and rotations for a rectangular beam with a variation top and bottom laminate under a combined load.

4.10. The deflection for the original beam is in the last column of Table 4.8 and highlighted in Figure 4.10 with the black dotted line.

Again as with the stringer addition and the mid spar inclusion the stiffness is increased which lowers the tip deflection and rotation by 26.4%. Notice that the torsional stiffness is enlarged resulting in a lower tip rotation. The shear flow expression, $s$ over $t$ in Equation 4.9, has changed which results in a lower rotation. Compared to the previous verification cases the error has slightly increased, however still within an acceptable 2.5%.
Both the 1D Proteus and Nastran 2D model are implemented correctly as the deformations agree with elementary mechanics. The next step is to verify the Proteus aeroelastic results with the Nastran model.
4.2 Swept Forward Wingbox

The next step is to go from the simple rectangular section of the previous section to an airfoil shaped wing-box. Only the low and medium fidelity code will be compared. First a general tip load case will be considered to show the correct implementation of the model. The laminate is quasi-isotropic and the skin thickness is constant at 25 mm. This value has been chosen to have aeroelastic properties that can be easily verified.

The wing properties can be seen in Table 4.9. The wing has a NACA 0012 symmetric airfoil. The wingbox starts at 30% of the chord and ends at 70%. The resulting cross-section is given in 4.11.

Table 4.9: Properties of the 0012 forward swept wingbox

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>b/2</td>
<td>16 m</td>
</tr>
<tr>
<td>S/2</td>
<td>32 m^2</td>
</tr>
<tr>
<td>\Lambda_{c_1/4}</td>
<td>-26.6 deg</td>
</tr>
<tr>
<td>\Gamma</td>
<td>0 deg</td>
</tr>
<tr>
<td>c_{ref}</td>
<td>2.0 m</td>
</tr>
<tr>
<td>\lambda</td>
<td>1.0</td>
</tr>
<tr>
<td>AR</td>
<td>16</td>
</tr>
<tr>
<td>c_{root}</td>
<td>2.0 m</td>
</tr>
</tbody>
</table>

Table 4.10 shows the deformation result for a tip load applied to both a straight (no sweep) and swept forward wing with a force F of 20000 N, a moment M of 10000 Nm and a torque T of 5000 Nm. For the straight wingbox the error is around 1%. For the swept forward wing an error of about 3% is observed in the results. The deviation is due to different assumptions with respect to the wingbox orientation both models have.

A generic forward swept wingbox has been sketched in Figure 4.12. The blue wingbox in Figure 4.12 represents the Nastran model. The Proteus model is given in red. The models differ in two aspects. First of all in Proteus each section of the wingbox is assumed to be
Multifidelity Verification

Figure 4.12: Wingbox cross-section for the forward swept wing. Front spar at 30% and the rear spar at 70% of the chord

Table 4.10: Deflection and rotation for a swept forward wingbox under a static tip load with $F = 20000\, N$, $M = 10000\, Nm$ and $T = 5000\, Nm$

<table>
<thead>
<tr>
<th>Property</th>
<th>Straight Wingbox</th>
<th>Swept Wingbox</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z (m)$</td>
<td>Proteus 1.057</td>
<td>Nastran 1.069</td>
</tr>
<tr>
<td></td>
<td>Proteus 1.060</td>
<td>Nastran 1.20</td>
</tr>
<tr>
<td>$\theta_x (deg)$</td>
<td>Proteus 5.746</td>
<td>Nastran 5.790</td>
</tr>
<tr>
<td></td>
<td>Proteus 6.133</td>
<td>Nastran 6.133</td>
</tr>
<tr>
<td>$\theta_y (deg)$</td>
<td>.104</td>
<td>.103</td>
</tr>
<tr>
<td></td>
<td>.1512</td>
<td>.1555</td>
</tr>
</tbody>
</table>

With the previous section and the above load application the FEM model for a static deflection is verified. However, as aeroelastic tailoring is considered, those effects should be verified as well. First the aeroelastic deformation is verified. Thereafter the divergence is investigated. The considered wing will be swept forward as forward swept wings are more prone to divergence. The next sections consider gravity and a trimmed flight.

4.2.1 Aeroelastic Deformation

Two aeroelastic cases are considered. First the aeroelastic deformation will be compared for a zero Mach number. Secondly a non-zero Mach number flight case is checked.

The analysis is performed for a wing with an angle of attack $\alpha$ of 5.0 deg at sea level. After a convergence study to make sure the models have converged to a solution the results of an aeroelastic analysis of a swept forward wing are given in Table 4.11 and Figure 4.14.

The total lift produced at $V = 100\, m/s$ and $M = 0$ is $1.30 \times 10^5\, N$ according to Nastran, while
4.2 Swept Forward Wingbox

Figure 4.13: Deflection of a swept forward wingbox section with $F = 20000\; N$, $M = 10000\; Nm$ and $T = 5000\; Nm$

<table>
<thead>
<tr>
<th>Property</th>
<th>Proteus</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z (m)$</td>
<td>2.26</td>
<td>2.29</td>
<td>1.43</td>
</tr>
<tr>
<td>$\theta_x (deg)$</td>
<td>9.83</td>
<td>10.01</td>
<td>1.80</td>
</tr>
<tr>
<td>$\theta_y (deg)$</td>
<td>3.36</td>
<td>3.43</td>
<td>2.24</td>
</tr>
<tr>
<td>Lift (N)</td>
<td>$1.33\cdot10^5$</td>
<td>$1.30\cdot10^5$</td>
<td>2.20</td>
</tr>
<tr>
<td>$q_{div} (N/m^2)$</td>
<td>$1.42\cdot10^4$</td>
<td>$1.43\cdot10^4$</td>
<td>0.22</td>
</tr>
</tbody>
</table>

Table 4.11: Analysis results for an aeroelastic analysis of a swept forward wing at alpha = 5 deg, $M = 0.0$, $V = 100\; m/s$
Figure 4.14: Deflection for a forward swept wing $V = 100 \text{ m/s}$ and $M = 0$
4.2 Swept Forward Wingbox

<table>
<thead>
<tr>
<th>Property</th>
<th>Proteus</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>δz (m)</td>
<td>4.82</td>
<td>5.03</td>
<td>4.16</td>
</tr>
<tr>
<td>θx (deg)</td>
<td>20.90</td>
<td>21.91</td>
<td>4.58</td>
</tr>
<tr>
<td>θy (deg)</td>
<td>6.96</td>
<td>7.31</td>
<td>4.79</td>
</tr>
<tr>
<td>Lift (N)</td>
<td>2.67·10⁵</td>
<td>2.56·10⁵</td>
<td>4.23</td>
</tr>
<tr>
<td>q\text{div} (N/m²)</td>
<td>1.03·10⁴</td>
<td>1.04·10⁴</td>
<td>1.26</td>
</tr>
</tbody>
</table>

Table 4.12: Analysis results for an aeroelastic analysis of a swept forward wing at alpha = 5\ deg, M = 0.8, V = 100 m/s

Proteus predicts 1.33·10⁵ N. The difference is 2.20%. The low fidelity code predicts divergence to take place at a dynamic pressure of 1.42·10⁴ N/m². According to Nastran q\text{div} equals 1.43·10⁴ N/m². The difference is 0.22 %, so the divergence pressure is calculated correctly. From Table 4.11 it can be seen that the deformation shows a difference of 1.5% for δz and θx, while θy shows difference of around 2%. The error is smaller compared to the static deflection.

The maximum deviation between the Proteus and Nastran model is 2.2 %. Therefore the implementation is considered to be correct. The next step is to take compressibility effects into account. Only the Mach number has been changed to 0.8 compared to the previous flight case. Table 4.12 gives the deflection, lift and divergence results. Figure 4.15 shows the deflection and rotation of the wing.

For M=0.8, under similar circumstances, the lift according to Proteus is 2.67·10⁵ N, while Nastran calculates a lift of 2.56·10⁵ N. The difference is increased to 4.23%. Because of the Prandtl-Glauert correction the lift is higher compared to the Mach zero load case. The low fidelity model also uses the Prandtl-Glauert correction factor to take compressibility into account.

The divergence takes compressibility of the air into account. The divergence pressure has slightly changed. The LF code calculates a divergence pressure of 1.03·10⁴ N/m² and the HF model predicts 1.04·10⁴ N/m². The difference is 1.26%. Hence the divergence is calculated correctly for a swept forward wing flying at Mach 0.8.

Due to the higher predicted lift the deformations are larger. As a difference in lift of 4.23% already exists in the lift, also the deformations have a 4% difference. The observed difference between the models is below 5% which is considered to be reasonably well. Therefore the static aeroelastic deformation of the Proteus model are verified.

4.2.2 Strain Comparison

To verify the strains the velocity is set to 75 m/s such that the strains are in the allowable range (for optimization). Gravity is taken into account and the Mach number is 0. In Figure 4.16 the maximal principal strain is given. On the left hand side the strain according to Proteus is given, while on the right hand side the Nastran strain is given. In Figure 4.17 the minimum principal strain is given. The range of the colour bars is similar.
Figure 4.15: Deflection for a forward swept wing \( V = 100 \ m/s \) and \( M = 0.8 \)
4.2 Swept Forward Wingbox

Figure 4.16: Maximum Principal strain for the top skin, bottom skin and spars
Figure 4.17: Minimum Principal strain for the top skin, bottom skin and spars
4.2 Swept Forward Wingbox

(a) Top skin, top view, Proteus

(b) Top skin, top view, Nastran

(c) Bottom skin, bottom view, Proteus

(d) Bottom skin, bottom view, Nastran

(e) Front spar (top) and Rear spar (bottom), Proteus

(f) Front spar (top) and Rear spar (bottom), Nastran

Figure 4.18: Shear strain for the top skin, bottom skin and spars
### Table 4.13: The maximum, minimum and shear principal strain for the top skin, bottom skin and spars

<table>
<thead>
<tr>
<th>Property</th>
<th>Top Skin</th>
<th>Bottom Skin</th>
<th>Spars</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \varepsilon_{\text{max}} ) (( \mu \text{strain} ))</td>
<td>1160</td>
<td>3270</td>
<td>2920</td>
</tr>
<tr>
<td>( \varepsilon_{\text{min}} ) (( \mu \text{strain} ))</td>
<td>-3640</td>
<td>-1040</td>
<td>-3310</td>
</tr>
<tr>
<td>( \varepsilon_{\text{shear}} ) (( \mu \text{strain} ))</td>
<td>2400</td>
<td>2160</td>
<td>2180</td>
</tr>
</tbody>
</table>

The shear strain definition in Proteus and Nastran is different. Proteus calculates the engineering shear strain \( \gamma_{xy} \), Nastran however uses the strain \( \varepsilon_{xy} \). The relation between \( \gamma_{xy} \) and \( \varepsilon_{xy} \) is:

\[
\gamma_{xy} = 2\varepsilon_{xy} = 2\left(\frac{1}{2}\left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y}\right)\right)
\]  

(4.11)

The Proteus engineering shear strain is divided by two to compare the shear strain results of Figure 4.18.

In Figure 4.16 the maximum principal strains are plotted for the top skin (a and b), bottom skin (c and d) and the spars (e and f). On the left the Proteus results are given, while on the right the Nastran results are shown. The general trend for the maximum principal strains is similar for Proteus and Nastran. The principal shear strain is maximum at the root of the bottom skin which is observed for both models. Towards the tip the strain reduces. The minimum principal strains are plotted in Figure 4.17. Again the Proteus results are on the left, while the Nastran results are on the right. The minimum is, in blue, found at the root of the top skin. In both models the minimum occurs at the left of the top skin. Towards the tip the minimum principal strains go to zero. At last the shear strains are compared in Figure 4.18. A similar trend is shown by Proteus and Nastran with a maximum shear strain at the left of the top skin (a and b) and at the right of the bottom skin (c and d). In Table 4.13 the maximum strain values are given for each section.

In Nastran the strain in the middle of the element is used to obtain the maximum value. At the sides of the element, as it deforms, the strain value can be higher. The Nastran results are symmetric, while Proteus shows a larger variation in the strain results. Although the general trend is similar, due to the different strain calculation methods the maximum, minimum and shear principal strains cannot be direct compared, especially for the spars.
### 4.2 Swept Forward Wingbox

<table>
<thead>
<tr>
<th>Mode #</th>
<th>Proteus freq. (rad/s)</th>
<th>Nastran freq. (rad/s)</th>
<th>difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1st mode</td>
<td>6.97</td>
<td>6.98</td>
<td>0.20</td>
</tr>
<tr>
<td>2nd mode</td>
<td>19.50</td>
<td>19.03</td>
<td>2.40</td>
</tr>
<tr>
<td>3rd mode</td>
<td>43.18</td>
<td>43.28</td>
<td>0.21</td>
</tr>
<tr>
<td>4th mode</td>
<td>120.42</td>
<td>117.45</td>
<td>2.47</td>
</tr>
<tr>
<td>5th mode</td>
<td>119.90</td>
<td>120.30</td>
<td>0.33</td>
</tr>
<tr>
<td>6th mode</td>
<td>188.07</td>
<td>181.53</td>
<td>3.48</td>
</tr>
</tbody>
</table>

**Table 4.14:** First six eigenfrequencies of the swept forward wingbox

#### 4.2.3 Eigenfrequencies

As described in Chapter 3.2.3 flutter is a dynamic phenomenon where the eigenmodes of the structure combine. To correctly calculate the flutter results both the eigenmodes and the eigenfrequencies of both models should be similar. Table 4.14 shows the first six eigenfrequencies of the swept forward wing.

The eigenfrequencies show a variation of 3% which is considered to be good due to the difference in assumptions with respect to the wingbox orientation. The eigenmodes are visualized in Figure 4.20. From Table 4.14 it is observed that the Proteus frequencies are not ascending. From the eigenmode analysis it was observed that the fourth and fifth mode should be switched to observe equal modes. As the two eigenfrequencies are close to each other and due to the different computation methods a small difference in eigenfrequency explains the difference between the two modes.

The first eigenmode, according to Figure 4.19b (cyan line) is the first bending frequency in z (up-down) direction. The second eigenmode (blue) is the first bending in x (chord) direction as visualized in Figure 4.19a. The third (black) and fifth (red) eigenmode are the second and third bending in z eigenmodes (Figure 4.19b). The second bending in x direction is highlighted in green (Figure 4.19a). Finally the first torsion eigenmode is found for the sixth eigenfrequency in Figure 4.19c (magenta line). As flutter is often a combination of the first bending (z) and torsion mode, at least six eigenfrequencies should be considered for this wing for the flutter analysis. A flutter analysis however cannot be performed for this wing design as it suffers from divergence before it starts to flutter. Therefore the flutter verification will be performed with the One Engine Reference Model in the next chapter.

#### 4.2.4 Gravity

So far all performed load cases have been executed without gravitational forces. The next step is to include gravity into the verification. A mass is added, to represent the engine, in the middle of the wing to influence the deflection of the wing. In the LF model both a force and mass should be added to model gravity. In Nastran an extra card is added to describe a gravitational acceleration. The verification is performed with a static analysis. Table 4.15 gives the deflection of the wing under gravitational loads. The difference is lower than 1.5% and therefore the gravitational acceleration has been implemented correctly. The deflection curves are given in Figure 4.20.
Figure 4.19: First six eigenmodes for the swept forward wingbox as given in Table 4.14 by Proteus (dashed line +) and Nastran (solid line ○).

<table>
<thead>
<tr>
<th>Property</th>
<th>Proteus</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>-0.494</td>
<td>-0.487</td>
<td>1.45</td>
</tr>
<tr>
<td>$\theta_y$ (deg)</td>
<td>-0.502</td>
<td>-0.500</td>
<td>0.49</td>
</tr>
</tbody>
</table>

Table 4.15: Tip deflection and rotation for a swept forward wingbox under 1g acceleration
4.2 Swept Forward Wingbox

Figure 4.20: Deflection and rotation for a swept forward wingbox under 1g acceleration

<table>
<thead>
<tr>
<th>Property</th>
<th>Proteus</th>
<th>Nastran</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>1.30</td>
<td>1.30</td>
<td>0.31</td>
</tr>
<tr>
<td>$\theta_x$ (deg)</td>
<td>5.68</td>
<td>5.68</td>
<td>0.09</td>
</tr>
<tr>
<td>$\theta_y$ (deg)</td>
<td>2.02</td>
<td>2.03</td>
<td>0.61</td>
</tr>
<tr>
<td>Lift (N)</td>
<td>8.56 $\times 10^4$</td>
<td>8.53 $\times 10^4$</td>
<td>0.41</td>
</tr>
<tr>
<td>$q_{div}$ (N/m²)</td>
<td>1.42 $\times 10^4$</td>
<td>1.43 $\times 10^4$</td>
<td>0.22</td>
</tr>
</tbody>
</table>

Table 4.16: Aeroelastic results for a trimmed flight with a fuselage weight of 10000 kg, the wingbox weight and an engine of 1000 kg

4.2.5 Trim

Depending on the input both the Proteus and Nastran model can determine the angle of attack given a certain flight condition such that the aircraft remains stable and at a similar height. This is the so-called trimmed flight condition in which the acceleration in z-direction is zero. In the optimization the goal is reduce the wingbox weight. As the weight reduces, the angle of attack can be reduced as the lift can be lowered.

Table 4.16 gives the aeroelastic results for a 1g trimmed flight at Mach zero. The weight of the aircraft has been chosen as such that a realistic flight condition is achieved. A small difference of 0.18% in lift exists. The difference exists due to the different weight of the models. The 1D beam model has a wingbox weight of 1265 kg, while the 2D Nastran shell model weights 1296 kg. In the previous section for a similar angle of attack an error of 2.20% occurred (M=0, Table 4.11), hence the aerodynamic lift result of both methods differ slightly. The combination of different aerodynamic loads and a difference in wingbox weight causes the difference of 0.18% for the lift in trimmed configuration.

Also the deflections show only a small deviation. The deflection curves are given in Figure 4.21. The angle of attack according to Proteus is 3.36 deg, while Nastran trims the wing at 3.44 deg which is a difference of 2.4 %. In Chapter 5 a more complex trimmed wing will be verified with an higher g load and for a non-zero Mach number.
Figure 4.21: Deflection curves for a trimmed flight with a fuselage weight of 10000 kg, the wingbox weight and an engine of 1000 kg
Goal of this chapter was first of all to verify the correct implementation of a model in Nastran. By comparing the tip deflection with elementary mechanics both the low fidelity and medium fidelity models have been verified. Secondly Nastran has served as a verification of the separate aeroelastic effects for the Proteus model. As the effects are correctly computed, with errors only up to 5%, the next step is to analyse a full wingbox structure.
To investigate the influence of engine location and wing sweep in the next chapter the One Engine Reference Model (OERM), courtesy of the DLR (German Aerospace Research Centre), will be used. In the previous chapter various parts of the aeroelastic toolbox of Proteus and Nastran have been considered separately. In this chapter all effects are verified for the more complex reference model.

The wing properties of the OERM are summarized in Table 5.1. Figure 5.1 gives the wing planform (a) and wingbox design (b). In Figure 5.2 the twist distribution is plotted. The material properties are given in Table 5.2. These properties will also be considered for the optimisation in Chapter 6.

To check the correct implementation of the model in both the Proteus and mid fidelity Nastran software first a tip load is applied to the model for a simple derivative of the OERM wing. Subsequently elements are added until the full wing is simulated. Results of this process are omitted from the report. In both the Proteus and Nastran model ribs will be simulated as concentrated mass elements. Although the 2D Nastran model has the potential for the implementation of actual ribs the goal is to verify the Proteus code. Therefore a similar wing structure will be analysed.

First the results of the mesh convergence study are shown in Section 5.1. Secondly the deformation curves are given for a certain flight condition to compare the aeroelastic deformation of Proteus and Nastran in Section 5.2. In Section 5.3 the flutter computations will be verified as this has not been performed so far. Finally in Section 5.4 an optimisation is performed to serve as a baseline model for the engine and sweep variation that will be performed in the next chapter.

## 5.1 Mesh Convergence

The OERM will be used throughout the remainder of the research. A mesh convergence study is performed to make sure the obtained results are independent of the mesh size considered. The OERM consists out of a structural mesh and an aerodynamic mesh. A structural mesh
Wing Properties

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>$b/2$</td>
<td>29</td>
<td>$m$</td>
</tr>
<tr>
<td>$S/2$</td>
<td>190.4</td>
<td>$m^2$</td>
</tr>
<tr>
<td>$\Lambda_{c/4}$</td>
<td>29.4</td>
<td>$^\circ$</td>
</tr>
<tr>
<td>$\Gamma$</td>
<td>4.6</td>
<td>$^\circ$</td>
</tr>
<tr>
<td>$c_{ref}$</td>
<td>7</td>
<td>$m$</td>
</tr>
<tr>
<td>$\lambda$</td>
<td>0.23</td>
<td>$-$</td>
</tr>
<tr>
<td>$AR$</td>
<td>8.83</td>
<td>$-$</td>
</tr>
<tr>
<td>$c_{root}$</td>
<td>11.2</td>
<td>$m$</td>
</tr>
<tr>
<td>$m_{fuselage}$</td>
<td>90000</td>
<td>$kg$</td>
</tr>
</tbody>
</table>

Table 5.1: Wing properties of the One Engine Reference Model

Strength Properties

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
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<tbody>
<tr>
<td>$E_{11}$</td>
<td>83</td>
<td>GPa</td>
</tr>
<tr>
<td>$E_{22}$</td>
<td>8.5</td>
<td>GPa</td>
</tr>
<tr>
<td>$G_{12}$</td>
<td>4.2</td>
<td>GPa</td>
</tr>
<tr>
<td>$\nu_{12}$</td>
<td>0.35</td>
<td>$-$</td>
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Failure Properties

<p>| | |</p>
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<tbody>
<tr>
<td>$\epsilon_{max}$</td>
<td>4500</td>
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<tr>
<td>$\gamma_{max}$</td>
<td>7000</td>
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Density

<p>| | |</p>
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<thead>
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<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>$\rho$</td>
<td>1490</td>
</tr>
</tbody>
</table>

Table 5.2: Material properties for the OERM

(a) Wing planform

(b) Wingbox design

Figure 5.1: One Engine Reference Model planform and wingbox design
5.1 Mesh Convergence

A convergence study is performed by checking how the tip deflection changes for various mesh sizes for a tip load applied to the wing. The aerodynamic mesh is investigated by considering the total lift of a stiff wing that shows very little deformation. The result of the mesh convergence study is shown in Figure 5.3.

For each section the number of structural elements in chord direction is defined. A section is part of the skin between the front and mid spar or mid and rear spar. For 4 elements in a section the solution of the tip deflection is already within 0.3% of the highly converged solution with 24 elements. Therefore the 4 elements per section will be used for the OERM model. This results in 2138 grid points and 2160 CQUAD elements.

To perform the aerodynamic mesh convergence the wing has been made very stiff such that the structural deflection has no influence on the convergence study. The amount of panels in chord directions is varied. The aerodynamic panels are square to achieve the best possible result. From the amount of panels in chord direction the amount of panels in span direction is derived. For 4 panels in chord direction the solution is already within 0.60% of the high
density mesh. For the OERM model 12 panels in chord direction will be chosen (0.11% difference compared to the high density mesh) to make sure that if the wing is deformed a converged solution is reached.

Note that towards the tip the chord decreases and as such the mesh size becomes smaller, hence the amount of elements increases. The higher amount of elements is preferred as towards the tip the lift decreases. To make sure the decay in lift is captured a higher mesh density is preferred. Figure 5.4 shows the lift distribution comparison of the Proteus and Nastran code. In the previous section the lift was already verified and as can be seen in Figure 5.4 the lift distribution for the Proteus and Nastran model is identical.

5.2  Aeroelastic Deformation

To verify the full OERM one of the optimisation load cases will be used. A trimmed flight is considered at 2.5g with a dynamic pressure of $1.94 \times 10^4 \, N/m^2$ at Mach 0.78. The aeroelastic deformation curves are shown in Figure 5.5. The numerical tip displacement and rotation, along with the lift and angle of attack are given in Table 5.3.

From Table 5.3 it can be seen that the difference between the low fidelity and mid fidelity model is highest for the tip (structural) deflection angle $\theta_x$. As with the wingbox wing model this difference occurs due to the different implementation of the wingbox as the low fidelity model is a 1D beam model, while the high fidelity model is a full shell representation of the wingbox.
## 5.2 Aeroelastic Deformation

<table>
<thead>
<tr>
<th>Property</th>
<th>Proteus (m)</th>
<th>Nastran (m)</th>
<th>Difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$</td>
<td>2.43</td>
<td>2.57</td>
<td>5.9</td>
</tr>
<tr>
<td>$\theta_x$ (deg)</td>
<td>6.56</td>
<td>6.97</td>
<td>6.2</td>
</tr>
<tr>
<td>$\theta_y$ (deg)</td>
<td>-3.69</td>
<td>-3.84</td>
<td>4.3</td>
</tr>
<tr>
<td>Lift (N)</td>
<td>1.65</td>
<td>1.66</td>
<td>0.19</td>
</tr>
<tr>
<td>$\alpha$ (deg)</td>
<td>3.38</td>
<td>3.45</td>
<td>2.1</td>
</tr>
</tbody>
</table>

Table 5.3: Tip deformation, angle of attack and lift for the OERM at 2.5g trimmed flight conditions with $q \cdot 10^4 N/m^2$ at Mach 0.78

![Deflection $\delta_z$](image1.png)

![Rotation $\theta_x$](image2.png)

![Rotation $\theta_y$](image3.png)

**Figure 5.5:** Structural deformation curves for the initial OERM design
Table 5.4: Tip deflection and rotation comparison for the factor 10 trick to obtain flutter

5.3 Flutter

So far the flutter has not been verified. The swept forward wing treated in the previous section is divergence critical, so no flutter speed could be obtained. Verifying flutter with Nastran and Proteus is not straightforward. With the converged mesh local eigenmodes appeared in the modal analysis. Due to the local modes the flutter speed cannot be determined. Therefore the material properties are changed such that only the global eigenmodes are obtained. The thickness has been multiplied by 10, while the stiffness coefficients and density are divided by 10. For the first load case the tip deflection and rotations are given in Table 5.4 for the two types of analysis.

The global bending shape is dominated by the in-plane laminate stiffness which remains unchanged with the multiplication of the thickness and dividing the stiffness properties with 10. The global eigenmodes appear due to a low out-of-plane bending stiffness. The bending stiffness scales with the power of three to thickness, and as such the local eigenmodes are removed due to this trick. The equations to calculate the in-plane and out-of-plane stiffness is given in Equation 5.1. For laminates the stiffness of the laminate is calculated from the inverse of the ABD matrix. The power of three for thickness for the out-of-plane stiffness is clearly visible in Equation 5.1b.

\[
E_{\text{in-plane}} = \frac{1}{a_{ij}} \quad (5.1a)
\]

\[
E_{\text{out-of-plane}} = \frac{1}{t^3 d_{ij}} \quad (5.1b)
\]

As can be seen in Table 5.4 the response of the original material properties and the modified properties is comparable. The largest difference is 4.3%. Other multiplication factors (2 and 5) also have been applied, however the local buckling modes where not removed. Therefore the factor 10 is applied.

To correctly calculate the flutter speed also the eigenfrequencies of the structure need to match. Table 5.5 displays the structural eigenfrequencies (with the factor 10 thickness properties).

Not every eigenmode in Table 5.5 can be clearly distinguished. Mode 7 for example appears to be a combination of a bending and torsion mode. The largest difference between Proteus and Nastran occurs for the 6th eigenmode. Flutter takes place if the 1st bending mode and 1st torsion mode collide. Therefore these modes should be taken into account for the flutter analysis.

For the swept forward wingbox in Section 4.2 the eigenvalues and eigenmodes of the Proteus
5.3 Flutter

<table>
<thead>
<tr>
<th>Type</th>
<th>Proteus freq. (rad/s)</th>
<th>Nastran freq. (rad/s)</th>
<th>difference (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1st mode 1st bending z</td>
<td>11.4</td>
<td>11.1</td>
<td>2.63</td>
</tr>
<tr>
<td>2nd mode 1st bending x</td>
<td>31.1</td>
<td>30.1</td>
<td>3.22</td>
</tr>
<tr>
<td>3th mode 2nd bending z</td>
<td>37.8</td>
<td>37.7</td>
<td>0.26</td>
</tr>
<tr>
<td>4th mode 3th bending z</td>
<td>74.6</td>
<td>76.8</td>
<td>2.95</td>
</tr>
<tr>
<td>5th mode 2nd bending x</td>
<td>91.4</td>
<td>89.9</td>
<td>1.64</td>
</tr>
<tr>
<td>6th mode 4th bending z</td>
<td>104.8</td>
<td>112.2</td>
<td>7.06</td>
</tr>
<tr>
<td>7th mode 1th bending-z-torsion</td>
<td>147.8</td>
<td>151.3</td>
<td>2.37</td>
</tr>
<tr>
<td>8th mode 3th bending z</td>
<td>181.4</td>
<td>180.3</td>
<td>0.61</td>
</tr>
<tr>
<td>9th mode 1st torsion y</td>
<td>204.1</td>
<td>208.5</td>
<td>2.16</td>
</tr>
<tr>
<td>10th mode 2nd torsion y</td>
<td>235.0</td>
<td>238.2</td>
<td>1.36</td>
</tr>
</tbody>
</table>

Table 5.5: Eigenmode comparison for the OERM

<table>
<thead>
<tr>
<th>Property</th>
<th>M = 0</th>
<th>M = 0.8</th>
</tr>
</thead>
<tbody>
<tr>
<td>$V_f$ (deg)</td>
<td>Proteus</td>
<td>Nastran</td>
</tr>
<tr>
<td>554</td>
<td>569</td>
<td>2.7</td>
</tr>
<tr>
<td>$\omega_f$ (rad/s)</td>
<td>67.0</td>
<td>73.4</td>
</tr>
</tbody>
</table>

Table 5.6: Flutter velocity and frequency (at Nastran flutter velocity $V_f$)

and Nastran models were already compared. The factor 10 multiplication was not performed as the thickness properties of the wingbox where as such that the local eigenmodes did not appear.

Now the eigenfrequencies are compared, finally the flutter velocity can be verified. Figure 5.6 displays the damping value as a function of velocity $v$. The engine and pylon are removed as they are stabilizing for flutter. The flight Mach number is set to 0 for Figure 5.6. If the damping value is positive flutter occurs. The results for the flutter velocity and the associated frequency are given in Table 5.6 for both a zero and non-zero Mach number.

According to Proteus the flutter velocity is 554 m/s, while Nastran calculates 569 m/s (difference of 2.7%). The damping values themselves cannot be directly compared as the computation methods are different. The frequency of the flutter results, the complex part of the solution, however can be compared. Nastran has a frequency of 73.4 rad/s for the flutter velocity of 569 m/s, while Proteus has a frequency of 67.0 rad/s at this velocity (569 m/s). The structural eigenfrequencies already showed a difference of around 3% (Table 5.5) for most nodes and therefore the flutter frequency is not identical. Also the different method causes the values to be different for both calculation methods.

A flutter speed of 569 m/s is unrealistically high. In practice such a speed will not be achieved for the designed wing. The goal is to numerically shown that the flutter velocity is correctly calculated.

A similar analysis is performed by only changing the Mach number from 0 to 0.8. As shown in Table 5.6 the velocity at which flutter starts to occur according to Nastran is 545 m/s with a frequency of 62.5 rad/s. At a similar velocity Proteus has a frequency of 67.8 rad/s which is a difference of 7.9%. The low fidelity code has a flutter speed of 522 m/s. The difference in flutter velocity for M is 0.8 is 4.4%. The larger difference occurs because, as described in

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Section 4.2, adding a Mach number increased the difference in deformation between the two models.

### 5.4 OERM Optimisation

The aircraft is optimized for various conditions in the flight envelope. During the lifetime the aircraft will encounter various circumstances. The safety authorities have defined which conditions the aircraft has to withstand in the Certification Specification. For the OERM the CS-25 manual is used (EASA, 2012). Five load cases are summarized in Table 5.7.

The first load case is the airplane in simple cruise condition. The second refers to a symmetric pull down of the aircraft which produces a negative g load. The third and fourth are two symmetric pull up manoeuvres with a load of 2.5g. The fifth and final load case is related

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to the stability of the wing. Fuel loads are not considered. No flutter or divergence may be experienced by the wing for this flight condition. Also the aeroelastic deflection should be within the specified boundaries.

All load cases are specified at Equivalent Airspeed (EAS). The density decreases as the aircraft flies at an higher altitude. The Equivalent Airspeed, related to the dynamic pressure, is the speed at which the aircraft flies at altitude, at sea-level conditions. For the stability load case (5) the EAS is as such that it relates to a negative altitude as can be seen in Table 5.7.

**Optimisation results**

After the first weight optimisation run the mid fidelity Nastran optimizer obtains a final wing weight of 1586 kg. A reduction in weight is closely related to a minimization of the Root Bending Moment (RBM). To check if a further reduction in weight is possible minimization of the Root Bending Moment has been set as an objective. The thickness is temporarily removed from the optimisation such that only the orientation of the laminate can be changed. The last step is to perform a second weight optimisation with the results of the RBM case to check if any further weight reduction is possible. For the second weight optimisation of the OERM the weight is further reduced with 14% to 1365 kg.

The optimiser in Nastran performs a gradient based optimisation. There is a possibility that the optimizer finds a local optimum. Due to the different objective of the RBM optimisation the sensitivities changes where some laminates have a larger incentive for the stiffness to be tailored. With the second weight optimisation run the initial design point is different and thereby a further reduction is possible. This strategy has been adopted as during the research it turned out that hardly any tailoring for the orientation of the laminate appeared.

The laminate thickness and orientation after the first weight optimisation is displayed in Figure 5.7. For the following tailoring to reduce the Root Bending Moment the laminate orientations are given in Figure 5.8. The thickness and stiffness distribution of the second and final weight optimisation is visible in Figure 5.9. Note that the thickness distribution is plotted on a flat panel, while the actual wingbox structure as an airfoil shape.
Figure 5.7: Optimized tailored wing thickness and stiffness distribution for the OERM after first weight optimization
Figure 5.8: Optimized thickness distribution for the OERM after first weight optimization and root bending moment optimization (t unchanged)
Figure 5.9: Optimized tailored wing thickness and stiffness distribution for the OERM after second weight optimization
Figure 5.7 gives the wingbox laminate design of the OERM. First the thickness distribution is given. The top and bottom skin have an equal skin thickness. The stiffness of the spars is higher and increases towards the tip. The thickness should be considered to be an equivalent thickness as stringers are not included in the optimisation. In Figure 5.7b and 5.7c the stiffness distribution of the top and bottom skin is plotted. The stiffness orientation is given by the red shapes on top of the skin. A circle indicates a quasi-isotropic laminate, while a stretch ellipse indicates a more tailored laminate with the stiffness in the direction of the major axis. Up to the middle both the upper and lower skin laminates is dominated by stiffness along the wing axis. This provides the wing with stiffness to sustain all loads. Towards the tip the laminates remain isotropic with hardly any tailoring. Also for the front spar the laminates are quasi isotropic as well as the rear spar with some tailoring at the root. The stiffness of the mid spar is along the wing axis.

As described earlier, because hardly any tailoring occurred the objective was changed to reduce the root bending moment. The result is given in Figure 5.8. Thickness is not optimized for, so only the orientation of the laminate can be optimized. For the top and bottom skin the first part of the laminates are still tailored along the wing axis to provide stiffness to withstand the aerodynamic loads. Compared to the first weight optimisation at the tip, near 24 m of the span, the top and bottom skin show tailoring to produce less wash-out. The stiffness is perpendicular to the wing axis providing more torsional stiffness. Comparing the stiffness of the laminates in the front and rear spar small changes are visible, however the laminates are still (close to) quasi isotropic.

With the output OERM for weight first and RBM second the design is taken for a second weight optimisation run. The results are given in Figure 5.9. Comparing the thickness distribution of Figure 5.9a and 5.9a a clear difference is visible. The skin thickness starts at 5 mm at the root and reduces to about 2 mm at the tip. The spar thickness is low at the root and increases towards the tip. The front spar consists mainly of quasi isotropic laminates. The third section of the rear spar (at about 6 m), in Figure 5.9e shows a stiffness perpendicular to the wing axis, towards the tip the laminate orientation becomes more isotropic. The mid-spar stiffness has not changed to much with a stiffness along the wing axis.

Comparing the stiffness of the first weight optimisation and RBM optimisation the changes occurred are minor. The main stiffness direction of some laminates was changed. The largest difference occurred for the second weight optimisation run. The reason for the transition is the different initial point from where the gradient based optimisation starts. The largest difference was observed for the thickness distribution.

For the second weight optimisation, the highest laminate stiffness is at the tip of the rear spar. Werter and Breuker (2015) observed that a higher stiffness is required at the tip to make sure the strains are not violated inboard. The stiffness distribution of the entire wing determines how the load distribution is throughout the wing and what the strains are that occur.

The structural deformation in for the wingbox for load case 4 is given in Figure 5.10. The deformation shows a tip down deflection of the wingbox towards the tip.
In Figure 5.11 the wingbox weight is plotted as a function of the iteration cycle. The convergence criterion is as such that if the weight change is below 5 kg (0.01% of the initial weight) the optimizer quits. A higher convergence criterion may lead to a lower weight, however the computation time increases. The criterion in the current optimisation proves to be a good trade-off between accuracy of the solution and computation time. In Figure 5.11b it is clearly visible that a large weight reduction occurs after about 10 iterations. As another initial design is chosen the sensitivities changes and as such the final design is different.

As already described the wingbox is optimized under stability and strain constraints. The normal strain in x- and y-direction is considered. The strains are limited to 4500 µstrain and the shear strain may not exceed 7000 µstrain. Furthermore the tip rotation is limited to 12 deg and the wing should be free of flutter for the conditions of the fifth load case in Table 5.7.

Before the constraints are checked, first the Nastran and Proteus code are compared for the highly tailored wing design. The displacement and rotation of the reference axis of the wing, at 50% of the chord, are displayed in Figure 5.12. The largest difference occurs for $\theta_y$ which is 5.4%. As can be seen in Figure 5.12c the deviation is largest at the tip. Note that a small discrepancy exists for $\theta_z$ around 10 m from the span in Nastran. The constant bending angle is not visible in the $\delta_z$ deformation curve in Figure 5.12a. The error is considered to be a numerical fault in Nastran which occurs at the location where the mid spar ends.

The strain will be the highest for the most critical load case. To identify this critical load case the tip deflection for the five load case are given in Table 5.8. The highest (positive) deflection occurs for load case 4. Also the strains will be checked for load case 2 which has a high negative deflection.
5.4 OERM Optimisation

---

(a) Weight Optimisation 1  
(b) Weight Optimisation 2

**Figure 5.11:** Evaluation of the wingbox weight

<table>
<thead>
<tr>
<th>Load case</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
<th>5</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta_z$ (m)</td>
<td>2.33</td>
<td>-5.04</td>
<td>3.70</td>
<td>3.83</td>
<td>-1.76</td>
</tr>
<tr>
<td>$\theta_y$ (deg)</td>
<td>-4.97</td>
<td>1.43</td>
<td>-8.67</td>
<td>-8.37</td>
<td>-4.01</td>
</tr>
<tr>
<td>$\alpha_{\text{trim}}$ (deg)</td>
<td>3.19</td>
<td>-4.55</td>
<td>5.00</td>
<td>5.16</td>
<td>0.74</td>
</tr>
<tr>
<td>$\alpha_{\text{tip}}$ (deg)</td>
<td>-3.84</td>
<td>-5.18</td>
<td>-5.73</td>
<td>-5.27</td>
<td>-5.33</td>
</tr>
</tbody>
</table>

**Table 5.8**

In the last row of Table 5.8 the angle of attack at the tip of the wing is given. The tip angle is a combination of the static deflection, the trimming angle of attack and the initial twist distribution (Equation 3.13). Due to the tip down deflection of the structure the tip twist is well below the limit of 12 deg.

The strain distribution for load case 2 is given in 5.13. For load case 4 the strains are plotted in Figure 5.14. The maximum strains for load case 2 and 4 are summarized in Table 5.9.
Figure 5.12: Structural deformation curves for the optimized OERM design for load case 4
5.4 OERM Optimisation

Figure 5.13: Nastran strain results for load case 2

(a) Strain x top skin and front spar
(b) Strain x bottom skin and rear spar

(c) Strain y top skin and front spar
(d) Strain y bottom skin and rear spar

(e) Strain xy top skin and front spar
(f) Strain xy bottom skin and rear spar
Figure 5.14: Nastran strain results for load case 4
5.4 OERM Optimisation

<table>
<thead>
<tr>
<th>Strain</th>
<th>load case 2 (µstrain)</th>
<th>load case 4 (µstrain)</th>
</tr>
</thead>
<tbody>
<tr>
<td>(\epsilon_{x_{max}})</td>
<td>4190</td>
<td>2970</td>
</tr>
<tr>
<td>(\epsilon_{x_{min}})</td>
<td>-4110</td>
<td>-3330</td>
</tr>
<tr>
<td>(\epsilon_{y_{max}})</td>
<td>4780</td>
<td>4345</td>
</tr>
<tr>
<td>(\epsilon_{y_{min}})</td>
<td>-4700</td>
<td>-4265</td>
</tr>
<tr>
<td>(\epsilon_{xy_{max}})</td>
<td>2800</td>
<td>4050</td>
</tr>
<tr>
<td>(\epsilon_{xy_{min}})</td>
<td>-2750</td>
<td>-5040</td>
</tr>
</tbody>
</table>

Table 5.9: Maximum strain values of load cases 2 and 4 for the optimized OERM wingbox design

The normal strain in x and y direction in the middle (centre) of the laminate is constraint. In the optimisation environment only the strains at the top and bottom of the laminate can be optimized for (strain at FIBER location). Therefore the individual strains at the top and bottom of the laminate have been combined to calculate the strain in the middle of the composite plate (Equation 3.14). The results in Figures 5.13 and 5.14 have been obtained by analysing a single load case and requesting the strains in the middle of the laminate (CENTER location in Nastran). Due to the different computation environment (optimisation versus single aeroelastic analysis) and an averaging a small offset or violation of the results can be expected and is not necessarily considered to be bad as long as the error is not too large.

As can be seen in Table 5.9 for load case 4 all strains are below the critical value of 4500 µstrain for \(\epsilon_x\) and \(\epsilon_y\) and 7000 µstrain for \(\epsilon_{xy}\). A violation occurs for load case 2 for \(\epsilon_y\). The error occurs at the laminates near the root of the wing as can be seen in Figure 5.13c and 5.13d. The combination of different computation environment and strain request is considered to be the cause of the error as explained earlier.

From Figure 5.13 the mid spar is visible. A reduction of the strain occurs near the edges of the laminate where the mid spar is located. For both load case 2 (Figure 5.13) and load case 4 (Figure 5.13) the strains reduce to zero towards the tip region. In the tip region the aerodynamic loads are the lowest and thereby the strains are lower as well.

Various options exist for the type of strain which is considered in the optimisation. The normal and shear strain in x- and y-direction is chosen, other options are the principal strains, von Mises strains or maximum shear. The principal strains are a better constraint as a combination of strains in x and y direction is used to obtain the principal directions. The principal strains are higher compared to the normal strains depending on the loading condition. As a result if the principal strain values are optimized for the critical strain values are higher. To make sure the strain constraint is not violated the optimizer will find a higher laminate thickness, resulting in a higher wingbox weight.

The optimized OERM design appears to meet all constraints except for a small violation of the strain. For the remainder of the report it will be assumed that the final design the optimizer gives does not violate any constraints.
Chapter 6

Engine & Sweep Variation

The created interface between Proteus and Nastran has now been verified. The next step is to apply the interface to various wing designs. In Section 6.1 the engine location is varied with respect to the span of the One-Engine Reference Model (OERM). Note that as the engine location is shifted, also the pylon will move along. If the engine location is mentioned, actually the engine-pylon combination is shifted. The quarter chord line is swept in Section 6.2.

6.1 Engine Location

The influence of nine different engine locations on the weight of the OERM wing will be investigated. Starting at 10% of the span, the engine and pylon (together) location will be increased to 90% of the span. The position of the engine and pylon with respect to the chord is constant. In Figure 6.1 the engine and pylon locations studied are shown. Engine thrust will not be considered.

The wing properties of the OERM are given in Table 5.1. As with the OERM optimisation described in Chapter 5 various load cases will be considered. For completeness these load cases are highlighted in Table 6.1. Fuel loads are not taken into account.

<table>
<thead>
<tr>
<th>ID</th>
<th>Description</th>
<th>EAS (m/s)</th>
<th>Altitude</th>
<th>Mach</th>
<th>g</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Cruise</td>
<td>117.5</td>
<td>11900</td>
<td>0.78</td>
<td>1.0</td>
</tr>
<tr>
<td>2</td>
<td>Symm. push down</td>
<td>162.4</td>
<td>11900</td>
<td>0.78</td>
<td>-1.0</td>
</tr>
<tr>
<td>3</td>
<td>Symm. pull up</td>
<td>117.5</td>
<td>7620</td>
<td>0.80</td>
<td>2.5</td>
</tr>
<tr>
<td>4</td>
<td>Symm. pull up</td>
<td>180.8</td>
<td>6096</td>
<td>0.69</td>
<td>2.5</td>
</tr>
<tr>
<td>5</td>
<td>Stability</td>
<td>222.0</td>
<td>-2300</td>
<td>0.57</td>
<td>1.0</td>
</tr>
<tr>
<td>(6)</td>
<td>Landing</td>
<td>-</td>
<td>-</td>
<td>-</td>
<td>2.0</td>
</tr>
</tbody>
</table>

Table 6.1: Load cases for the One-Engine Reference Model
Figure 6.1: Investigated Engine locations for the OERM
6.1 Engine Location

As shown in Table 6.1 a landing load case has been added. For all variations in the engine location an optimisation will be performed with and without this additional sixth load case. For the landing load case a static analysis is performed in Nastran. The 2g load represents a hard landing. During the landing the engine may not hit the runway. The clearance of the engine for the OERM is 0.87 m which is taken from a similar type of aircraft (Airbus A330) (S.A.S., 2015). This distance is obtained from the aircraft characteristics manual as shown in Figure 6.2 (R). The 2g load is obtained from the airworthiness manual. According to the CS-25 airworthiness requirements an aircraft impacts the ground at a prescribed (vertical) velocity of 3 m/s during landing (Wright and Cooper, 2007). If the aircraft is decelerated in a fifth of a second a deceleration of 15 m/s² occurs which is rounded to a 2g acceleration load.

The deflection and 2g load is taken as an initial input to investigate the influence of a landing load case on the optimisation results. The dihedral of the wing is not taken into account and as such the engine clearance is not increased for engine locations further down the span of the wing.

The following optimisation is performed:

1. Weight optimisation run 1; stopping criterion: 0.01% of total weight
2. Root Bending Moment Optimisation without thickness; stopping criterion: 0.01% of total RBM
3. Weight optimisation run 2; stopping criterion: 0.01% of total weight

First an optimisation is performed with minimizing weight as an objective. In Chapter 5 a further weight reduction was achieved by minimizing the Root Bending Moment and subse-
Figure 6.3: Tailored wing weight for the OERM with varying engine location

sequently performing a second weight optimisation run. By changing the objective the search
direction of the gradient based optimisation changes as the sensitivities are different. Also
the initial design is different potentially allowing for a further weight reduction in the second
run.
The result for the engine optimisation are shown in Figure 6.3. The optimizer aborts its
optimisation if the weight change compared to the previous iteration is below 0.01% of total
weight which is about 5 kg.

In Figure 6.3 the result of the OERM wingbox optimisation with a varying engine location is
given. The green dot represents the original OERM as discussed in Chapter 5. The blue line
is the wingbox weight without load case 6, the landing load case, while for the red line this
load case is considered. As with the OERM first an initial weight optimisation is performed,
highlighted with the dashed lines, subsequently the Root Bending Moment (RBM) is reduced
(without thickness as a design variable) and a second weight optimisation is performed (solid
lines). Each engine location examined has a separate label which is connected by a line to
indicate similar load case combinations.
The baseline OERM has a wingbox mass of 1365 kg. By placing the engine at 50%, taking
into account the landing load case, a weight reduction of 250 kg is achieved as the wingbox
mass has been reduced to 1115 kg. This is an 18% reduction. Placing the engine further
towards the tip can therefore be very beneficial for the final wingbox design. The reason for
this reduction is explained later on. The optimal wingbox design without landing load case
has a mass of 994.7 kg which is an even more significant weight reduction of 27%. 
As the engine is placed further down the span the wingbox weight goes down. At around 50% of the span the landing load case becomes critical and a higher wingbox stiffness, achieved by an increase in laminate thickness, is necessary such that the engine displacement is below the critical 0.87 m.

In the region in which the engine location is between 4 and 15 m the trend of the optimisation is similar. The landing load case is not critical yet and both optimisation cases go to a similar weight. Small differences occur however. Although a similar stopping criterion is used, a different optimum is observed. The optimisation procedure in Nastran is gradient based, so a local optimum may be found. Further research is required to study the influence of the stopping criterion or to investigate the use of a different optimizer.

A gradient based optimizer calculates sensitivities to find the optimum design. As the objective is changed from minimum weight to the minimum Root Bending Moment (RBM) the sensitivities changed. As with the original OERM a second weight optimisation is performed to see if the wingbox weight has changed. For the optimisation without the landing load case the potential weight reduction is much larger compared to the optimisation with load case. A possible cause could be the used stopping criterion in combination with the gradient based optimisation due to which a local optimum is found. The reduction in weight for the optimisation without the landing load is given in Table 6.2.

Table 6.2 shows the necessity to be very careful with the chosen convergence criterion and objective. A reduction up to 14% is achieved (engine at 30% of the span) by changing the objective to a RBM minimization and a second weight optimisation. The weight decrease with the changing objective is less if the engine is placed closer to the tip.

Going back to the weight result plot in Figure 6.3 first the weight results without considering the landing load is discussed (blue line). As the engine is located further outward the engine has a larger relief effect. The engine prevents the wing from bending up and thereby achieving to high strains. This is also visible in Figure 6.4 where the tip displacement is plotted for each load case. The tip displacement is a measure to examine which load case is critical. Each line represents the displacement for the given load case at the analyzed engine locations without the landing load case (blue lines in Figure 6.3).

As long as the engine is at 50% of the chord or less load case 4 is critical for an upward tip deflection, while load case 2 is critical for the downward deflection. In Figure 6.4 the relief effect is also visible, as the engine is positioned further to the tip load case 4 and 2 become less critical and other load case are more critical.

The stabilizing effect of the engine on the wingbox is only applicable for engine locations up to 70% (20.3 m) for the OERM. If the engine location is placed any further to the tip at 23 or 26 m the wingbox weight increases as load cases 2 and 5 become critical as shown in Figure 6.4.

Table 6.2: Engine Location optimization results

<table>
<thead>
<tr>
<th>Engine (% of span)</th>
<th>10</th>
<th>20</th>
<th>30</th>
<th>40</th>
<th>50</th>
<th>60</th>
<th>70</th>
<th>80</th>
<th>90</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight Run 1 (kg)</td>
<td>1499</td>
<td>1499</td>
<td>1587</td>
<td>1410</td>
<td>1216</td>
<td>1161</td>
<td>1072</td>
<td>1100</td>
<td>1252</td>
</tr>
<tr>
<td>Weight Run 2 (kg)</td>
<td>1444</td>
<td>1358</td>
<td>1360</td>
<td>1266</td>
<td>1178</td>
<td>1005</td>
<td>994.7</td>
<td>1101</td>
<td>1254</td>
</tr>
<tr>
<td>Weight gain (kg)</td>
<td>55.3</td>
<td>141</td>
<td>227</td>
<td>144</td>
<td>38.4</td>
<td>159</td>
<td>77.7</td>
<td>-0.41</td>
<td>2.13</td>
</tr>
<tr>
<td>% Weight gain (%)</td>
<td>3.7</td>
<td>9.4</td>
<td>14.3</td>
<td>10.1</td>
<td>3.2</td>
<td>13.4</td>
<td>7.2</td>
<td>0.0</td>
<td>0.17</td>
</tr>
</tbody>
</table>
Figure 6.4: Tip deflection $\delta_z$ for each load case for each OERM optimized wingbox design where the engine location is varied

The red line in Figure 6.3 represents the wingbox weight including the landing load. As the engine is placed further to the tip more stiffness is needed in the wingbox to make sure the engine does not hit the ground with a 2g landing load. The higher stiffness, without violating the strain constraint, is achieved by increasing the thickness of the wingbox up to the engine location. Therefore the weight of the wingbox increases. For the OERM the landing load case is critical if the engine is placed beyond 50% of the span. Note that the results for 70 and 80% are 2688 and 4556 kg respectively. With the engine located at 90% the optimizer could not find a feasible design within the design space.

The thickness and stiffness distribution of the lightest wingbox design, with and without considering the landing load case, is discussed in the next sections. In Appendix C the results of all optimisations are presented.

### 6.1.1 Best Design Without Landing Load

The best design without landing load is with the engine at 70% of the chord at 20.3 m. The thickness and stiffness distribution is plotted in Figure 6.5.
6.1 Engine Location

The thickness of the skin increases from 2 mm at the root to about 5 mm near the engine location. Towards the tip it decreases back to 2 mm. Similarly to the OERM (Figure 5.9) the front spar thickness increases to 7 mm, while the rear spar increases to 10 mm. As explained the higher stiffness at the tip ensures the strains are below the threshold at the inboard segments. The stiffness of the top and bottom skin can be divided into two regions.
The first half, up to about 15 m of the span, of the skin shows a high stiffness along the wing axis. Thereafter the orientation gradually changes from ± 45 degrees to a stiffness perpendicular to the beam axis. A wing wash-out occurs as the stiffness is perpendicular to the wing axis creating a bending-twist coupling. At the tip itself, also for the spars, the layup is quasi-isotropic as a change of stiffness has little influence on the overall wing design. The spars show a large variation in the orientation. In the front spar up to the engine location (indicated with the red dot) the stiffness is mainly along the beam axis. After the engine the stiffness is perpendicular to the beam axis creating a washout. The rear spar shows a larger variation with an extreme 0/90 stiffness direction at the engine location. The mid spar stiffness gently changes from along the beam axis to perpendicular to it.

As already described the landing load case is critical if the engine is placed further down the span. For the above wing design the landing load case is applied. The engine displacement with a static 2g load is 2.63 m. The displacement constraint for the engine is 0.87 m, so this is a clear violation of the constraint.

6.1.2 Best Design With Landing Load

If the landing load is considered the thickness distribution is as shown in Figure 6.6.
Compared with the no landing load considered, Figure 6.5, the thickness at the root for the top and bottom skin has increased. The thickness increases to 4.5 mm near the engine position. Towards the tip the skin thickness decreases to 1.5 mm. The spar thickness distribution is unchanged with a high thickness towards the trailing edge tip. The top and bottom skin stiffness show a similar stiffness distribution. In the root region up to the engine the stiffness
is along the beam axis. Towards the tip the skin is dominated by ±45 laminates. The laminates in the trailing edge spar between 5 and 10 m show a high stiffness perpendicular to the wing axis. This causes a wash-out effect as the torsional stiffness is increased. This also appears in the front and rear spar between 20 and 25 m. As with the no landing optimized design the spar stiffness goes from along the beam axis to perpendicular to the beam axis.

### 6.2 Sweep

Instead of the engine location a changing quarter chord sweep is examined. The OERM is taken as a reference and the quarter chord sweep is varied in steps of 10 degrees from -20 to 40 degrees. The aerodynamic taper and initial twist distribution have remained unchanged. In Figure 6.7 a top view is given in which all optimisation cases are given. The same load cases as for the engine variation optimisation are considered (Table 6.1) including the landing constraint. As the engine is at 30% of the chord this load case is not considered to be driving the design. The optimisation strategy is similar:

1. Weight optimisation run 1; stopping criterion: 0.01% of total weight
2. Root Bending Moment Optimisation without thickness; stopping criterion: 0.01% of total RBM
3. Weight optimisation run 2; stopping criterion: 0.01% of total weight
Figure 6.8: Tailored wing weight for the OERM with varying quarter chord sweep
The optimized wingbox weight as a function of sweep is plotted in Figure 6.8. The green dot is the original OERM treated in Chapter 5. The blue line is the wingbox weight after the first weight optimisation (Step 1). The red line represents the weight after a Root Bending Moment (RBM) reduction (without thickness) and subsequent second weight optimisation (Step 3).

The lightest wingbox design is for a quarter chord sweep of 50 deg. The weight is 1306 kg, which is a reduction of only 4.3% compared to the baseline. The location of the engine-pylon combination has a much larger weight reducing influence on the wingbox design as a weight reduction of 18% was observed. Therefore the engine has a larger potential to be used in reducing the wingbox weight.

The benefit of the RBM and second weight optimisation is for sweep not as significant as compared with the engine variation. Only for the 40 and 50 deg sweep a weight reduction is present.

In Nastran the divergence constraint can only be added to the optimisation if the wing is divergence critical. If divergence is not present for the wing, the optimizer will not function and the divergence constraint has to be removed. For the -20, 10 and 0 deg swept wing, divergence is a critical constraint which has to be analyzed. As discussed in Section 3.2.2 a forward sweep of the wing has a negative effect on the divergence pressure.

The main reason to sweep a wing is to increase the critical Mach number. By sweeping the wing, the aircraft drag divergence is delayed to higher Mach numbers. Increasing the sweep nonetheless reduces the lift. Specifically the lift over drag ratio L/D is decreased. The main aerodynamic advantage of swept forward wing is that flow separation occurs at the root first. The aircraft remains controllable as the ailerons still have a good flow conditions (Anderson, 2005). Note that for backward swept wings the separation is controlled by varying the tip distribution with a lower twist at the tip of the wing as shown in Figure 5.2.

The optimum wingbox weight is achieved for a sweep angle of 50 degrees. The quality of the lift distribution is not considered in the analysis. As described the sweep reduces the produced lift therefore the trimmed angle of attack $\alpha_{trim}$ has to be higher as displayed in Figure 6.9. The solid blue line is the first load case representing a steady and symmetric flight. With the higher angle of attack the drag is also higher. Because of the influence on the distribution and the larger trimming angle, higher sweep angles are not considered.

The tip deflection per load case and wingbox design is given in Figure 6.10. The sweep angle is on the horizontal axis and the tip deflection is on the vertical axis. For load case 6, dotted black line, the engine displacement (constraint for this load case) is plotted.
Figure 6.9: Trimmed angle of attack $\alpha_{trim}$ for the various load cases for each optimized wingbox design
Figure 6.10: Tip displacement $\delta_z$ for each load case for each OERM optimized wingbox design where the sweep angle is changed. For load case 6 the engine displacement is drawn.
As can be seen in Figure 6.10 it is safe to assume that the landing load case (dotted black line) is not limiting. For all sweep angles the engine displacement is around 0.25 m which is below the critical 0.87 m.

Due to the divergence (stability) constraint the swept forward wings are heavier (Figure 6.8) as they have a higher skin thickness as shown in Figure 6.11. The higher thickness results in a higher stiffness which causes the tip deflection to be lower for negative sweep angles as visible in 6.10.

For the engine location load cases 2 and 6 were critical (Figure 6.4). The highest tip deflection is also present for load case 2 (negative) and load case 6 (positive) for the varying sweep angle. The thickness and stiffness distribution for the optimum design with a quarter chord sweep of 50 deg is shown in Figure 6.12. The thickness and stiffness distribution of all sweep variations of the OERM are given in Appendix D.
Figure 6.12: Optimized tailored OERM wing thickness and stiffness distribution with a quarter chord sweep of 50 deg

The thickness for both the top and bottom skin decreases towards the tip of the wing as shown in Figure 6.12a. The trailing edge spar near the end of the wing has the highest thickness which was observed earlier. For the top and bottom skin multiple regions can be distinguished. The centre box region shows a high stiffness along the beam axis and perpendicular to it. Up to about 10 m the stiffness is roughly aligned with the beam axis. Looking from the back,
the front spar shows a stiffness of about 30 deg while the rear spar has a stiffness opposite of it. The mid spar has a similar distribution as the trailing edge spar.

Comparing the optimal sweep design of Figure 6.12 with the optimal engine location wingbox of Figure 6.6 the stiffness distribution of the top and bottom skin, as well as the spars is very different. The wingbox of Figure 6.6 shows a much higher stiffness along the beam axis compared to Figure 6.12. This is also visible in the tip deflection under an aerodynamic load. The tip deflection for load case 4 for the optimal sweep design is 5.6 m, while the for the optimal engine design this is 3.0 m. The optimal sweep design is far more flexible as the engine optimal design. Further research is required to investigate the optimal design if the principal strains are used instead of the normal strains. For the sweep variation the difference between the normal strains and principal strains may be very large.
Chapter 7

Conclusions and Recommendations

Conclusion

The goal of this thesis is twofold. In the first part of the thesis the 1D Proteus model (Werter and Breuker, 2015) has been verified. A 2D Nastran model serves as the verification model. Secondly the influence of engine location and sweep has been investigated using the One Engine Reference Model (OERM) as baseline.

To verify the 1D beam model first a rectangular beam is created. Under a variety of forces the deformation of the 1D model and 2D model compare very well with elementary mechanics. The error between the two models is below 1%. With the rectangular beam both the correct implementation of the beam in Nastran and the results of the 1D beam model have been verified. In the second phase of the verification Nastran is a benchmark for the aeroelastic effects in the Proteus model.

Due to the different assumptions of the models the maximal error increased to around 5%. The aeroelastic deformation, flutter and divergence analysis have similar results for different conditions including a non-zero flight Mach number, trimming the aircraft and gravitational loads. Even for the more complex OERM the error is around 5%. Therefore it can be concluded that the Proteus 1D is a helpful and verified tool to perform aeroelastic tailoring.

In the second part of the research the effect of engine location and quarter chord sweep is studied. The engine location has a large influence on the wingbox weight. The engine position can be used to further reduce the weight of the wingbox. Compared to the baseline model a weight reduction of 18% is achieved by placing the engine at 50% of the span. The further the engine is placed towards the tip, the larger the relief effect the engine has on the wingbox. A landing load case should be considered as an upper boundary for the spanwise engine location. The further the engine is placed towards the tip, the higher the wingbox weight to ensure the engine does not hit the runway during landing. When performing aeroelastic tailoring the engine location is very important to the overall wingbox design.

The sweep angle also have a large influence on the wingbox weight. For swept forward wings
Conclusions and Recommendations

The wingbox is divergence constraint. Compared to the baseline OERM, the optimal sweep angle design of 50 deg only showed a 4.3% weight reduction. For this high sweep angle a high trimming angle of attack is required for steady and symmetric flight. A constraint should be considered to limit this angle, to make sure the design is feasible.

**Recommendation**

The verification performed so far can be expanded. Although the Proteus model proves to be a verified tool to perform aeroelastic tailoring it has its restrictions. As it is a 1D beam model it does not model the ribs. The Nastran model has the possibility to include ribs and even perform optimisation for the rib laminates. The verification can be extended by comparing the buckling of the wingbox structure in both models. Buckling can be added as an additional constraint in the medium fidelity Nastran optimizer. The same holds for the gust loads as Nastran has a possibility to consider these dynamic loads.

In the current Nastran optimisation the normal strain in x, y and shear (xy) has been constrained. Instead of the normal strains the principal strain can be used. On the plane of the principal strain the shear strain is zero, while the direct strain is either a minimum or a maximum. Perpendicular to this strain the maximum (principal) shear strain is found. These principal strain is a combination of the normal strains x, y and xy. Depending on the strain condition the principal strains are higher compared to the normal strains. If the optimisation is performed with principal strain the obtained skin thickness will therefore be higher and a more conservative wingbox is designed.

During the research the Nastran optimizer turned out to be very hard to understand. The constraints are not easily monitored during the optimisation. The optimisation is a black box and the design cannot be monitored during the optimisation. Therefore an external optimizer is advised to have more control over the optimisation.

For the influence of engine location and sweep not all possibilities are investigated. So far the effect of engine location and sweep has been studied separately. A test matrix can be created to examine the combined effect of engine position and sweep. Also the relative location of the engine with respect to the chord has remained unchanged. Swept forward wings are prone to divergence. Placing an engine outboard in front of the chord has a stabilizing effect for divergence as it creates a tip down torsional moment on the wingbox. More optimisations have to be performed to investigate this effect.

Engine thrust is not considered in the current research. If the engine is located underneath the wingbox, as in the OERM, a tip up moment will be created due to the thrust. Especially for the swept forward wing thrust will have a disadvantageous effect which needs to be included.
References


S.W. Hertgers

MSc Thesis
REFERENCES


During the research multiple problems were encountered to develop an aeroelastic wingbox optimization using a 2D shell model in Nastran. This section contains guidelines based on the author’s experience. The guidelines are applicable for flutter calculation, aeroelastic analysis, optimization and trimmed flight analysis.

A.1 General Remarks

Although the next section might contain some trivial remarks, they are easily forgotten and therefore briefly mentioned.

The mesh seed of the skins, spars and ribs should match. The ribs for example cannot have a higher mesh density compared to the skin.

In default configuration Nastran produces a .dball file containing all information on the analysed problem. For example stiffness matrices and the deflections of all grid points are saved. In the optimization process, with much iterations, the file size can exceed 32 GB after which Nastran will quit the optimization. By using the 'buffsize=32769' command, prior to Nastran execution, the allowed size is increased to 64 GB which was sufficient for the performed optimization problems. The available memory can be further increased if necessary. Note that the .dball file does not contain any result. Another option is to use the command 'scratch=yes old=no' command. As the .dball file is a scratch command it will be deleted after execution of the bdf file. The 'old=no' command deletes the old results files such as .f04, .f06 and .xbd if they exist.

A.2 Optimization with Flutter

To calculate the flutter characteristics the eigenmodes of the structure, which can be obtained by a Nastran Model analysis (103), are required. If the original thickness is used local eigenmodes, in which a local panel eigenmode is found, occur. For a correct flutter calculation
the global modes are required. The Nastran flutter calculation method requires a number of eigenmodes to be included. Although Nastran can take any number of eigenmodes into account, the more eigenmodes are calculated, the longer the computation time. To have the global eigenmodes as the first to be calculated the thickness is multiplied by 10, while the stiffness and density is reduced by a factor 10 to keep the response similar. It has been verified that similar aeroelastic deformations occur. With the multiplied thickness the local modes are removed and a correct flutter speed is determined.

Nastran requires a list with reduced frequencies for which the aerodynamic loads are calculated. Via interpolation the aerodynamic properties for intermediate frequencies are obtained. Flutter normally occurs for a reduced frequency in the range of 0.2-0.4. At lower speeds, where flutter is unlikely to occur, the reduced frequency is higher (1.0-5.0). If the higher reduced frequencies are not defined in the flutter optimization analysis the flutter constraint is violated. Instead of interpolation the results for higher frequencies are extrapolated. The properties are as such that the optimizer violates the flutter constraints, while with the included higher reduced frequencies the flutter constraint is not violated. An early violation of the flutter constraint for higher reduced frequencies leads to an aborted design which can be further optimized. Therefore the higher reduced frequencies should be included in the flutter analysis. Flutter is a dynamic aeroelastic phenomena where two eigenmodes, usually torsion and bending, combine. In the Nastran flutter analysis only a restricted number of eigenmodes are taken into account. As already described above, the more modes are considered, the longer the calculation takes. To find realistic flutter properties however the bending and torsion modes should be within the amount of eigenmodes defined.

A.3 Nastran Environments

Every solution method has its own environment. So an aeroelastic calculation (144 analysis) is different from an aeroelastic calculation in the optimization environment. The final solution is within 1% for both methods however similar answers should not be expected. As the environments are different, so are the keywords used. Problems may occur between similar bdf files for aeroelastic analysis and aeroelastic optimization. In the 144 environment grid points and concentrated mass elements may have a similar identification number. In the optimization (200) environment the identification number should be different. The author has also observed that the optimization environment can only take 25 concentrated mass elements into account, although further research is required to confirm this observation.

A.4 Optimization

Using the DRESP1 keyword the structural design responses used in the optimization, either as constraint or as an objective, are defined. In this research the CQUAD4 structural elements are used. According to table 6-3 of the Nastran Optimization manual (Software, 2012a) only the strains at the top and bottom of the laminate can be used for the optimization. To optimize for the mid-strains in the laminate the strains on top and bottom are averaged.
During the optimization the author observed that the statement to print the strains to the .f06 results file influences the optimization results. Different optimization results were obtained by the statement to print the strains in the mid-plane, STRAIN(CENTER), or on the top and bottom of the laminate, STRAIN(FIBER), to the .f06 results file. It is advised to use the FIBER print command in the optimization. For a single aeroelastic analysis the user can decide on which strain to print depending on the needed strain.

To optimize for composite laminates the laminates parameters, as used by (Gurdal et al., 1999), define the laminate properties. Those parameters are defined from -1 to 1. Section 3.3.1 explains the relation between the lamination parameters and the stiffness properties. The Nastran optimizer showed problems in tailoring the stiffness of the laminates. After research a small scale optimization problem, the author decided to shift the design space to the positive domain. Instead of a design space ranging from -1 to 1, the parameters may vary between 0 and 2. The whole range is positive now. The equations to calculate the properties are changed accordingly. As a results the optimizer showed more tailoring. Therefore the author advises to only use a positive design space.

A.5 Optimization Debugging

In the optimization results file the analysis value and design value for the optimized properties are given. The analysis value displays the input as given in the bdf file. The design value is the result using the optimization parameters. If the input in the file matches the entered design values the user can check the correct implementation of the model. The analysis value and design value should be similar.

A.6 Divergence

For forward swept wings divergence is a constraint. The DIVERG1 DRESP1 card is used to obtain the divergence response. Only for divergence critical wings this constraint should be considered. If the wing is divergence free the optimizer cannot optimize for divergence as it is not active. Note that the option to optimize for the divergence constraint (DIVERG1 response) has not been documented in the manuals yet.

A.7 Trimmed Flight Analysis

The aeroelastic analysis is performed for a trimmed wing. To trim the wingbox is free to rotate. The angle of attack and movement in the z direction are free (ANGLEA and URDD3). In the trim card the movement in z direction is put to zero. Nastran trims the aircraft using the angle of attack (ANGLEA). The gravity load is modelled using the gravity parameter in combination with the load card. Another option could be to put the URDD3 statement to 1 and set the gravity load in the PARAM AUNITS definition.
A.8 Resultant Force

In the Nastran results file different kinds of resultant forces (and moments) can be requested. Using the GRDPNT command the masses of the wing are given as well as the resultants with respect to the entered reference point. The base reference point, to which the wingbox is clamped to restrict rigid body motions, is the used reference point. To clamp the wingbox a single point constraint is used in Nastran called SPC.

Instead of the GRDPNT command another function exists to directly print the resultant forces in the constraint base. The SPCFORCE command prints the resultants with respect to the constraint point. Only if a SPC consists in the point the resultant forces can be printed. The author has experienced that both resultant force options produce different results. The best results are delivered by the SPCFORCE command.

A.9 Root Bending Moment Optimisation

For the optimization in Nastran the DRESP1 card defines the structural response. The card can be used as either constraint or objective. To obtain the Root Bending Moment (RBM) as a response two response types can be chosen: SPCFORCE and GPFORCE. With the correct input both types should deliver similar results. As described in Section A.8 already a difference exists in the results to obtain the resultant force itself with the GRDPNT and SPCFORCE command. Using the GPFORCE command results in an error message to contact MSc Client Support. The SPCFORCE response however is available as an objective. Note that the SPCFORCE resultant force should be printed in the results file, if the response is used as an objective.

A.10 Card Description

Nastran works with so-called cards to define the FEM properties. The GRID card for example defines the location of a point. Depending on the task the user can use a combination of cards to retrieve the result. In this section the specific cards for aeroelastic and flutter calculation are explained as well as the optimization cards in Section A.10.3 (Software, 2012b). The section only contains a brief discussion with the most important notes. For a more in-depth discussion the Nastran Quick Reference Guide can be used or any specific manual on aeroelastic calculations or optimization.

Often the case control section will be mentioned. Before the bulk data (like GRID and CQUAD cards e.g.) is defined the case control section defines the type of analysis, the constraints, the loads and calculation method among other user defined options.

A.10.1 Aeroelastic Calculation

The cards required for aeroelastic calculation in Nastran are:

- CAERO1: The CAERO1 card is very common to use if an aeroelastic analysis is performed. It defines the panels for the aerodynamic analysis using the Doublet-Lattice
theory. Among its position parameters and property definitions it is important that it refers to a PAERO1 card.

- DIVERG: With this card the Mach Number for which a static aeroelastic divergence analysis is performed is defined. This card only becomes active if defined in the case control section at the start of the bdf file with the unique identifier in the first entry of the card. The card describes the amount of lowest divergence pressures to be printed and the Mach numbers at which those divergence pressures are computed. The method uses an eigenvalue analysis to obtain the divergence speed. Therefore a method for the complex eigenvalue analysis should be defined using the EIGC card.

- EIGC: This card is only required if a divergence analysis is requested as it defines the method to perform the complex eigenvalue analysis. As with the DIVERG card the first entry is an identifier also to be used in the case control section. Required entries are the method used and the desired number of eigenvalues.

- TRIM: Using the CAERO1 cards the wing planform itself is described. The flight properties at which the wing aerodynamics are analysed is described with the TRIM card. From the case control section a TRIM card is chosen. The Mach number and dynamic pressure should be defined. Depending on the used labels the type of flight is chosen. The labels should also be defined with the AESTAT CARD. The flight types are:

  1. Given angle of attack $\alpha$: only the ANGLEA AESTAT card with the value in radians

  2. Trimmed flight: Define the AESTAT cards ANGLEA and URDD3. In the TRIM card only the URDD3 label is fixed. If a value of 0.0 is entered the acceleration in $z$ direction is 0. The gravitational loads are defined in the LOAD card. Another option could be to set the value to 1.0 and use the PARAM ANUNITS card to define the gravitational forces.

- AEROS: If a static aeroelastic analysis is performed this card is used to define the basic parameters like reference chord, reference span and reference area. The card also defines the symmetry properties of the model. If, in the case control commands, the symmetry properties are already described these properties are used (so the case control commands are used).

- PAERO1: For the Doublet-Lattice method associated bodies need to be defined. Also if there are no bodies the entry is required. The property of the cards is referred to from the CAERO1 which define the (wing)panels.

### A.10.2 Flutter Calculation

- AERO: Similar card to the AEROS card described for aeroelastic analysis. The AERO card is for unsteady aerodynamics (e.g. flutter). Basic parameters like reference chord and reference density are defined. As with the AEROS card if, in the case control commands, the symmetry properties are already described these properties are used (so the case control commands are used).
• EIGR: For the flutter analysis the real eigenvalues are obtained. To calculate these eigenvalues the method is defined with the EIGR card. As with the EIGC card the first entry is the identification number to be set in the case control section. Thereafter a method should be defined as well as the number of desired roots.

• FLUTTER: The first entry of the FLUTTER card is the identification number which also should be defined in the case control commands. Next the flutter method is defined. The most common in flutter analysis, like K and PK, are available. The density, Mach number and reduced frequency are all defined in FLFACT cards. An important parameter is the amount of eigenmodes to include in the analysis. These modes should be similar to the number of desired roots in the EIGR card and the amount of modes stated in the PARAM LMODES card. Make sure that the first torsion and bending mode is included in the amounts of defined modes as flutter is usually a collision of these two modes.

• FLFACT: To calculate the flutter of the wing three FLFACT cards are required. The first defines the density, the second the Mach number and the third the velocity at which a flutter analysis is performed. Note that the current interface only calculates the flutter characteristics for a single velocity. If the velocity at which the wing starts to flutter needs to be determined this cards needs to be expanded with all velocities at which a flutter analysis needs to be performed.

• MKAERO: This card gives the combination of reduced frequencies and Mach numbers at which the aerodynamic matrices are calculated. Interpolation and extrapolation is used for reduced frequencies which are not defined. For a correct flutter analysis it is important to also denote higher reduced frequencies (for which flutter is unlikely to occur) as, due to the extrapolation, wrong aerodynamic properties are obtained.

### A.10.3 Optimisation

For optimisation in Nastran a combination of cards is used. The Figure A.1 highlights the relation between the various cards.

The FEM model is constructed with the design variables. Using the DVMREL and DVPREL cards the design variables are related to the FEM model. The response are retrieved with the DRESPi card. The response card is required to define the objective (DOPTPRM card) and the constraints (DCONSTR). Details on the specific cards are given below.

• DCONSTR: The design constraints are represented with the DCONSTR card. The first entry is the unique identification number to be referred to in the case control section at the start of the bdf file. If multiple constraint types are required they can be combined with the DCONADD card. Note that the strain and flutter constraint cannot be combined (with the DCONADD card). Each DCONSTR card links to a DRESPi card which contains a (structural) response. At last the upper and lower boundary (of the constraint) are defined. It is advised to normalize the constraints such that all constraints have a similar interval.
DEQATN: For some optimisation procedures user defined equations are required. Using the DEQATN card the user can supply equations to Nastran. For the performed optimisation DEQATN cards are used for the lamination parameter constraints, the material relations and the strain averaging.

DESVAR: To design variables are defined with the DESVAR card. The card is straightforward to use with a name of the design variable, a identification number, an initial value and the boundaries. The parameters entered in the property cards should be similar to the initial value entered in the design variable cards.

DOPTPRM: The optimisation procedure in Nastran has default parameters which define the optimisation process. Examples are the maximum amount of iterations and the stopping criterion. Using the DOPTPRM card these default values are overwritten. The most used parameters are CONV1 which contains the relative criterion to detect convergence and DESMAX to define the maximum amount of iterations. With this card also the constraints can be requested to be printed to the results file.

DRESPi: The response of the structure need to be connected to the optimizer. This is performed with the DRESPi card which connects the structural response to either the constraints or objective. With the DRESP1 card a direct relation is established between the optimizer and the response. With the DRESP2 card the relation is established using a DEQATN card. The identification numbers of DRESPi cards should be unique. The specific response attributes are different for all elements. For details the quick reference guide can be consulted. The two most important used as a design objective are given.
below. DRESP1 with ID 1 is the weight optimisation objective. Evident is the requested weight response. The attributes (ATTA and ATTB) refer to specific response, in this case the weight location in the Nastran weight matrix. DRESP1 with ID 2 is the Root Bending Moment (RBM) optimisation objective. It is a SPCFORCE (Single Point Constraint). The RBM is found in the fourth position of the residual force matrix ($F_x$, $F_y$, $F_z$, $M_x$, $M_y$, $M_z$).

- **DVMRELi**: The DRESPi card defines a relation between the response and the optimizer. A relationship between the design variables and the structure still need to be established. The DVMREL card provides a relation between a design variable and a material property relation. As with the response card the type, 1 or 2, determines if a user supplied equation is used.

- **DVPRELi**: Some design variables are related to element properties like shell thickness. To establish a relation between the design variable and a property the DVPRELi card is used. As with the previous cards the user can decide to use an DEQATN card.

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The properties DRESPi, DVMRELi and DVPRELi have multiple cards indicated with a number. If 1 is used (e.g. DRESP1) a default option in Nastran is used. If option 2 is used a user supplied Equation is used to obtained the response.
Appendix B

Interface Manual

The Mid Fidelity Nastran model is an extension to the Proteus code. The low fidelity input is required before the Nastran model is constructed. Additional information should be added to the low fidelity model such that the Nastran model can be created. This manual describes the structure to create the Nastran bdf file and details on the additional parameters required. The manual also contains recommendations on how the interface can be further improved. The manual is best understood with the Matlab input file.

B.1 Code Structure

The mid fidelity Nastran model is developed to be an extension to the developed Proteus code. Three functions are added to the main Matlab file as shown in Figure B.1.

The Mid Fidelity Function creates the bdf file required for Nastran. Additional parameters, described in Section B.2, are necessary.
In the Mid Fidelity Function first some calculations are done to create the Nastran FEM model. Thereafter, using separate Mid Fidelity Functions as described in Chapter 3.1, the ModGen input file is generated as is shown in Figure B.1.
With the input file ModGen creates separate text files containing the Nastran FEM model. The correct files are combined into a single bdf file. If required the text files are manipulated to take multiple skin and mid spars into account. Additional info is added depending on the Nastran input parameters. For example the type of output file produced by Nastran or parameters that describe the optimization process. The combination of cards required to perform the optimization is written by the author. ModGen is not used to do so.
After the bdf file has been created Nastran is run and subsequently the results are analysed. Both processes are performed in the main directory or script has displayed in Figure B.1.

In the Proteus code the reference axis of the wing is described using xyz coordinates. Around this reference axis the Nastran model is created.
Between the xyz coordinates entered segments are created. Each segment has 7 laminates...
Figure B.1: Mid Fidelity code structure

Main Directory

- Proteus Input
- Medium Fidelity Function

Medium Fidelity Directory

- Initial Calculations
- ModGen Input File
- ModGen Run
- Input manipulation
- Build BDF
- Run Nastran
- Analyze Results

- bdf file
- .inp file
- Medium Fidelity Functions
- Multiple Skin
- Mid Spars
- Combine files
- Add info

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consisting of a mid spar, top and bottom skin (divided by the mid spar) and a front and rear spar. The mid-spar, present on default, divides the skin in two. Only if a mid-spar laminate is defined it will be constructed in the Nastran model, else the mid-spar CQUAD elements are removed from the analysis.

If the external force is described as a structural mass, the force is not written to the bdf file. The lumped mass definition is used to describe masses in the Nastran model.

The wingbox cross-section definition has a special lay-out. A general wingbox layout has been sketched in Figure B.2.

First the location of the front, mid and rear spar are entered. The location is entered as percentage of the chord. Next the laminates in the segment are defined. Depending on the required mesh elements in chord direction the stringer location is determined (see FEM_LOC in the next section). The stringer position determines the location of the mesh elements. Only if stringer elements are added the stringer will be physically present. At last the data structure is defined to manipulate the input. In this data structure the presence of the stringer as well as a multiple skin layup for the segment in chord direction can be expressed. The lay-out of this data structure is given in Table B.1.

The first row of the data description in Table B.1 describes the location in the segment of the stringer as a fraction. The second row contains an integer to define if the stringer is physically present (1=yes, 0=no). In the example of Table B.1 in the middle of the section a stringer
Table B.1: Cross-section definition in the input

is present. The Lam ID in the third row indicates the laminate on the top skin to the left of the point. The last point is assumed to have a similar lam ID compared to the left-hand side of the point. The fourth and final row is for the bottom skin. In Table B.1 each chord section has a different laminate ID numbered from 1 to 12. Hence also 12 laminates should be described in the input.

The user can decide to optimize for weight or Root-Bending Moment (RBM). For the RBM optimization the thickness can be excluded from the optimization. If the thickness is excluded the optimizer can only change the orientation of the laminate to reduce the RBM. At last a possibility exists to print sensitivities after which the optimization is quit.

B.2 Parameter Description

The following additional parameters (next to the existing Proteus parameters) are required to create a successful bdf Nastran file.

1. Load Case (LC): Defines the load case with Mach number, dynamic pressure, load factor, velocity and the type of analysis to be performed for the load cases. The available types are aeroelastic (1), flutter (2), divergence (3) and landing (4). Flutter is not treated as an aeroelastic optimization load case. If strains are limiting for the flutter load case a separate load case has to be created for aeroelastic optimization of this flutter load case.

2. lc: load case if a single analysis is performed.

3. SOL: Nastran solution procedure
   (a) SOL 101: Static analysis
   (b) SOL 103: Modal analysis
   (c) SOL 144: Aeroelastic analysis including divergence pressure
   (d) SOL 145: Flutter analysis
   (e) SOL 200: Optimization for all the defined load cases

4. DMI type: For the aerodynamic analysis the Doublet-Lattice Method (DLM) is used. DLM assumes the wing to be a flat plate. To take curvature and twist into account each panel can be rotated using the DMI correction method. This parameter defines which DMI matrix is taken.
   (a) 0: no correction (uncambered airfoil)
   (b) 1: camber only (ModGen)
(c) 2: camber and twist (ModGen)
(d) 3: twist only (ModGen)
(e) 4: based on low fidelity input (calculation as described in Section 3.2.1)

5. FEM_LOC: the amount of mesh elements in the skin segments. The amount of elements in spars and ribs is derived from this definition. Only CQUAD4 elements are used.

6. Wingbox structure: Define the wingbox geometry as described in Section B.1.

7. Airfoil data: For each segment Nastran requires airfoil text files. The first column of this text file contains the x coordinate and the second column is the y coordinate. Via linear interpolation the PrepAirfoilData function creates airfoil data for each y (span) location of the wingbox structure. At the first and last y location an airfoil file should be defined.
   (a) name: name of the airfoil data files (.txt files)
   (b) airfoilxyz: y position were the above airfoils are defined

8. Name: name of the bdf file

9. OW_file: Gives a path to the folder where the Nastran bdf file will be executed. If the path is empty a new folder will be created using the name and time at which the job is submitted.

10. String_on: Flag to mesh at the stringer position (1) or at a regular distance based on the wingbox geometry(calculated by ModGen) (0)

11. post file type: 0 creates a xbd file, -1 an op2 file.

12. stype: Position in the laminate where the strain will be calculated either an top and bottom, FIBER, or in the centre (CENTER). In the optimization process FIBER should be used as the optimizer uses the top and bottom strains.

13. Rib laminate ID: ID of the rib laminate. The laminate is separately created, as it is not a laminate which is optimized.

14. Riblam: Laminate properties for the rib

15. stringlam: The stringer properties are also created as a laminate. The laminate however should be isotropic as the current implementation in Nastran uses MAT1 cards to describe the material properties of the stringer.

16. full: flag to create a bdf model of the full wing, including leading and trailing edge, in Nastran. Not functioning yet.

17. yribs: location of the ribs in span direction. Currently only equal to the segment definition in span direction.

18. lamtype: Flag to vary between the laminate definition. Currently only option 2 should be used.
19. ftype: Forces can be introduced using three dimensional RBE3 elements or using RBE2 elements. In the latter case the force is introduced to the rib, while in the first option the force is applied to a segment. The first option is preferred.

20. fmodes: Amount of eigenmodes to include in the flutter analysis. In case of a modal analysis it determines the amount of eigenmodes which will be calculated.

21. fk: list of reduced eigenfrequencies for flutter aerodynamic analysis. Note that higher reduced frequencies, for which flutter does not occur, are required as Nastran uses extrapolation for reduced frequencies outside the range. Due to the extrapolation a wrong aerodynamic matrix is used leading to (possible) wrong results.

22. skin_opt, spar_opt and rib_opt: flag to optimize skin and spars, rib flag is not active.

23. author: author name which will be displayed in the bdf file to trace back the creator of the bdf file.

24. li: load introduction type. The aeroelastic loads can be introduced in various ways into the structure. Option 1 introduces the lift load on top of the wingbox structure, while option 2 uses the reference axis to introduce the aeroloads.

25. tlim: Boundary on the thickness used in the (thickness) design variable definition.

26. opttype: Type of optimization, either weight (1) or Root-Bending Moment (2)

27. incl_t: flag to included or exclude thickness from the optimization. Helpful for optimization of the RBM.

28. sens: flag to only calculate the sensitivities. No optimization is performed.

29. edisp: allowed engine displacement for the landing load case

In the current implementation a (unit) force is always required.
The flutter load case only optimizes for flutter. Nastran is unable to perform a flutter and (static) aeroelastic analysis. Therefore the flutter load case should be repeated and applied as both aeroelastic and flutter load case.

B.3 Interface Constraints

The current interface between ModGen and Proteus to create a Nastran FEM model has some restrictions. The optimizer designs symmetric laminates. The interface can be extended to asymmetric laminates by defining additional design variables and equations to calculate the properties of the B matrix of the ABD matrix. The wingbox geometry is given, its shape is not optimized (no topology optimization). Gust loads and buckling are not constraint yet, however Nastran has possibilities to do so.
The above mentioned constraints are with respect to the general implementation of the interface. However the available options for the wingbox are also limited. The wingbox FEM models is restricted to:
• Ribs only perpendicular to the spars at the xyz segment definition (y) location.

• A mid spar definition has to be present. If no mid-spar is required it can be removed throughout the entire wing.

• Tip rotation (related to angle of attack and linearity of aerodynamic solver) if limited to 12 degrees.

• Fuel is not included. If a mass is added the mass is present for all load cases defined. It is not possible to have fuel masses for certain load cases only. If optimization has to be performed for load cases with and without fuel loads separate bdf optimization files have to be created.

• Stringer pitch in each segment is similar (constant stringer pitch in the wing). This ensures mesh continuity.

• A multiple skin layup can only be defined for skin sections (not for ribs or spars) in chord direction. In each segment the laminate in span direction is constant.

• The amount of mesh elements in each section is constant.

• For verification between the low and high fidelity code the mesh has to match to analyse a similar structure.

• The mesh of the spar skins and ribs has to match. A continuous mesh is created between the segments.

• If a segment has no mid-spar, any further section in the wing cannot have a mid-spar.

• Separate coordinate systems are created for the top- and bottom-skin and for the front and rear spar in each segment. The coordinate system main axis is in the direction of the reference axis. The mid spar has a similar coordinate system as the front spar.

• If stringers are defined they are always straight up or down.

• Stringer laminate ID should be different from skin laminate ID.

B.4 Recommendations

Currently ModGen is used to create the GRID points and CQUAD elements. The stringer is defined between GRID points using a CBAR element. The material properties of the stringer should be described using a MAT1 card which is a homogeneous material definition. A manipulation function can be written to create stringers in a similar manner as in the low fidelity code. Extra GRID points should be created below the stringer GRID points. Secondly CQUAD elements can be created. By doing so the stringer is no longer represented by CBAR elements, but by CQUAD elements. As in the Proteus code the stringer can be a laminate, instead of a homogeneous material currently used.

In the mid-fidelity Nastran model ribs can be modelled. Between ribs skin and spar segments are created. Each skin and spar laminate can be optimized for. In the Protesus model a new laminate is created for each xyz input location. A mismatch exists in the amount of
laminates created between Proteus and Nastran if more intermediate ribs are defined besides the xyz location used in the Proteus model. To allow for the analysis and comparison of the Proteus code with Nastran for a wingbox including ribs the mid-fidelity Nastran code should be adapted.

Currently the full potential of ModGen is not utilized. ModGen has much more options which are not used. Examples are the possibility to add ribs in flow direction, include skin in front of leading and behind trailing edges or to include fuel loads. These functions still have to be implemented.
Appendix C

Engine Location Optimization Results

The wingbox of the One-Engine is optimized for various engine locations. The engine location is varied in chord directions from 10% of the span to 90% of the span. In Section C.1 the results are shown for the optimization without the landing load case. In Section C.2 the landing load case is included. The engine location is highlighted with a red dot. Each page shows 6 different figures. In the first, top-left, plot the thickness distribution is shown. To the right the stiffness distribution in the top skin is plotted. The middle left and right plot give the stiffness distribution of the bottom skin and front spar respectively. Finally at the bottom the stiffness in the rear and mid spar are shown. Figure C.1 gives an overview of all engine locations for which the wingbox has been optimized.

C.1 Without Landing Load Case
Figure C.1: Investigated Engine locations for the OERM
Figure C.2: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 10% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1499 kg.
Figure C.3: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 10% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1444 kg.
Figure C.4: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 20% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1498 kg.
Figure C.5: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 20% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1358 kg.
C.1 Without Landing Load Case

Figure C.6: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 30% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1587 kg.
Figure C.7: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 30% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1360 kg.
Figure C.8: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 40% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1410 kg.
Figure C.9: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 40% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1266 kg.
Figure C.10: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 50% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1216 kg.
Figure C.11: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 50% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1178 kg.
Figure C.12: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 60% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1161 kg.
Figure C.13: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 60\% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1005 kg.
Figure C.14: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 70% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1072 kg.
Figure C.15: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 70% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 994.7 kg.
Figure C.16: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 80\% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1100 kg.
Figure C.17: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 80% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1101 kg.
C.2 With Landing Load Case

Figure C.20: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 10% of the chord initial weight optimization with landing load. Optimized wingbox weight: 1534 kg.
Engine Location Optimization Results

Figure C.18: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 90% of the chord initial weight optimization without landing load. Optimized wingbox weight: 1252 kg.
C.2 With Landing Load Case

Figure C.19: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 90% of the chord optimized weight after RBM optimization without landing load. Optimized wingbox weight: 1254 kg.
Figure C.21: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 10% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 1528 kg.
C.2 With Landing Load Case

Figure C.22: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 20% of the chord initial weight optimization with landing load. Optimized wingbox weight: 1402 kg.
Figure C.23: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 20% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 1358 kg.
C.2 With Landing Load Case

Figure C.24: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 30% of the chord initial weight optimization with landing load. Optimized wingbox weight: 1380 kg.
Figure C.25: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 30% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 1357 kg.
Figure C.26: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 40% of the chord initial weight optimization with landing load. Optimized wingbox weight: 1261 kg.
Figure C.27: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 40% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 1236 kg.
Figure C.28: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 50% of the chord initial weight optimization with landing load. Optimized wingbox weight: 1268 kg.
Figure C.29: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 50% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 1115 kg.
Figure C.30: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 60% of the chord initial weight optimization with landing load. Optimized wingbox weight: 1541 kg.
Figure C.31: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 60% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 1501 kg.
Figure C.32: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 70% of the chord initial weight optimization with landing load. Optimized wingbox weight: 2771 kg.
(a) Optimized thickness distribution  
(b) Top skin, top view  
(c) Bottom skin, top view  
(d) Leading edge spar, rear view  
(e) Trailing edge spar, rear view  
(f) Mid spar, rear view

Figure C.33: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 70% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 2688 kg.
Figure C.34: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 80% of the chord initial weight optimization with landing load. Optimized wingbox weight: 4556 kg.
Figure C.35: Optimized tailored wing thickness and stiffness distribution with the engine and pylon at 80% of the chord weight optimization after RBM optimization with landing load. Optimized wingbox weight: 4556 kg.
Appendix D

Sweep Variation Optimization Results

The quarter chord line sweep of the One-Engine Reference Model (OERM) has been varied from -20, in steps of 10 degrees, to 40 degrees as shown in Figure D.1.

The results highlighted are the thickness distribution, main stiffness direction for the top- and bottom skin, front-, mid and rear spar. The thickness distribution is plotted on the 0-sweep wing to be able to compare all configurations.
Figure D.1: Researched quarter chord sweep OERM configurations
Figure D.2: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of $-20 \text{ deg}$ after 1 weight optimization. Optimized wingbox weight: 3034 $kg$. 
Figure D.3: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of -20 deg weight run 2 after RBM optimization. Optimized wingbox weight: 3023 kg.
Figure D.4: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of -10 deg after 1 weight optimization. Optimized wingbox weight: 2344 kg.
Figure D.5: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of -10 deg weight run 2 after RBM optimization. Optimized wingbox weight: 2249 kg.
Figure D.6: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 0 deg after 1 weight optimization. Optimized wingbox weight: 2058 kg.
Figure D.7: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 0 deg weight run 2 after RBM optimization. Optimized wingbox weight: 1916 kg.
Figure D.8: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 10 deg after 1 weight optimization. Optimized wingbox weight: 1675 kg.
(a) Optimized thickness distribution

(b) Top skin, top view

(c) Bottom skin, top view

(d) Leading edge spar, rear view

(e) Trailing edge spar, rear view

(f) Mid spar, rear view

**Figure D.9:** Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 10° weight run 2 after RBM optimization. Optimized wingbox weight: 1660 kg.
Figure D.10: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 20 deg after 1 weight optimization. Optimized wingbox weight: 1485 kg.
Figure D.11: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 20 deg weight run 2 after RBM optimization. Optimized wingbox weight: 1439 kg.
Figure D.12: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 30 deg after 1 weight optimization. Optimized wingbox weight: 1383 kg.
Figure D.13: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 30 deg weight run 2 after RBM optimization. Optimized wingbox weight: 1387 kg.
Figure D.14: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 40 deg after 1 weight optimization. Optimized wingbox weight: 1550 kg.
Figure D.15: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 40 deg weight run 2 after RBM optimization. Optimized wingbox weight: 1380 kg.
Figure D.16: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 50 deg after 1 weight optimization. Optimized wingbox weight: 1423 kg.
Figure D.17: Optimized tailored wing thickness and stiffness distribution with a quarter chord sweep of 50 deg weight run 2 after RBM optimization. Optimized wingbox weight: 1306 kg.