Flow Control X-Plane

Final report Group 06

Faculty of Aerospace Engineering

TU Delft Delft University of Technology

AE3200 Design Synthesis Exercise

Challenge the future

Faculty of Aerospace Engineering

Design Synthesis Exercise: Flow Control X-Plane

Final report

by

S.R. *Bal*	4106067
F.G.A. *van de Beek*	1540661
I.R. *Dogan*	4017773
S. *van der Horst*	4015673
S.R. *Janssen*	4077857
A.W. *Kil*	4089537
T.S. *Onnink*	4085280
K.A. *te Riet*	4062221
B. *Roos*	1388576
J.M *Steiner*	4080297

Group number:	6	
Project duration:	April 22, 2013 -- June 21, 2013	
Instructors:	Dr. M. (Marios) Kotsonis	TU Delft, Tutor
	I. (Ilya) Popov, MSc	TU Delft, Coach
	F.G.A. (Francesco) Bertels, MSc	Tu Delft, Coach

Acknowledgements

We would like to expresss our gratitude towards our tutor, Dr. M. Kotsonis and our two coaches, I.B. Popov MSc and F.G.A. Bertels MSc for their guidance, their encouragement and useful points of criticism given throughout the project. We would also like to thank the staff of the Fellowship building and the staff of the OSSA-LR for offering resources to help us complete the project. Finally we would also like to thank our family and friends for their patience and positive attitude.

Preface

To conclude the Bachelor programme of Aerospace Engineering at the TU Delft, students are expected to participate in the course AE3200 Design Synthesis Exercise (DSE). During this exercise students are required to apply the knowledge they obtained in their first three years of study on an assignment to gain more experience in the design process of engineering. Students are divided in teams of ten and are given eleven weeks to complete their assignment. Starting from the very basic formulation of the problem at hand, the exercise takes students to the end of the initial cycle of a preliminary design phase. The project is concluded after this phase.

This report forms the conclusion of the DSE project regarding the design of a Flow Control X-Plane. It summarises all the work done in the last eleven weeks, starting with the project organisation and going up to the end of the first iteration in the preliminary design cycle. This report provides an all-round vision to the design. It does not only focus on technical details, but also illustrates the initial setups to integrate systems and elaborates upon the way the product should be used by future customers. Doing so, the project has been brought up to a level from which a detailed design team can continue where this group has ended.

The start of the project was more difficult than expected. The individual group members needed to get used to each other and to the setting of the DSE. As the project progressed the working atmosphere improved significantly leading up to this final result. Another issue the group encountered, was the fact that the emphasis of the project deliverables was placed too much on systems engineering methods and procedures, rather than actual project content. This issue was accompanied by the problem that group members were questioning themselves to what depth the design of the aircraft should go. This problem was faced at multiple instances during the project. Our tutor and coaches here have provided valuable assistance in providing course corrections whenever these were necessary.

As a conclusion we would like to remark that even though there have been some hiccups along the way and the deadline was moved forward, we have obtained a final result exceeding our expectations. We found this exercise a great opportunity to apply the knowledge gained during lectures in practice.

Summary

Since the aerospace industry continuously focuses on efficiency improvement, new technologies are constantly under development. Innovative Active Flow Control (AFC) systems have the potential to increase the performance of aircraft in terms of lift enhancement, drag reduction, and noise reduction. It would be desired to perform tests of AFC systems by experimental aircraft in real life conditions. However, the development costs of such aircraft is a barrier which is almost insurmountable. To this end, this project was initiated with the following project objective statement: "Develop an experimental cost-efficient platform, aimed at testing current and future flow control technologies in real life conditions, by ten students in eleven weeks".

In order to tackle this objective as time-efficiently as possible, the project has been divided into four phases. The first phase of the project was concerned with the organisation and planning. The second phases comprised the analysis of the mission in more detail. The high-level requirements as given by the client were studied. These were translated to lower level system- and functional requirements for the platform in general, the Active Flow Control (AFC) systems and test systems. In the third and conceptual design phase multiple concepts were generated, based on previously formulated design options. By means of trade-off processes, a final concept was generated. This final concept served as input for the preliminary design and was characterised by a cost-efficient, 'plug and play'-concept. The fourth and final phase was concerned with the preliminary design. This phase consisted of the first preliminary calculations with respect to the wing, pylon and interior (re)design.

The resulting design consists of a modified Boeing 737-500 with a fuselage mounted test section. Most top level requirements set up during the project have been met, with exception of the budget requirement. Production and further development of the test platform will exceed the budget by 7.8 million euros. The main wings are modified such that the outboard sections contain modular skin panels, wingtips, flaps, and slats. These modular skin panels are made non-load carrying by lowering the load carrying wing skin by about 5cm. The fuel located in the wing tank at the testing locations is reallocated to both the wing root and additional fuel tanks in the fuselage. Measurement equipment and AFC systems can be installed in a 'plug 'n play'-concept, enabling testing of multiple AFC systems simultaneously. This means that provisions are made to connect any actuator or measurement systems in terms of power and space allocation. An additional test section is mounted on a pylon truss-structure on top of the fuselage. This test section is mounted to the inner airframe, preventing excessive stress concentrations in the fuselage skin. The pylon structure is designed to enable testing of swept wings, straight wings, or unconventional configurations, such as morphing structures. Test measurement equipment, such as PIV and load cells, are integrated into the pylon structure. The pylon can pivot around an axis to enable testing of various angle of attack. The original interior will be stripped completely. It has been redesigned to accommodate the power, data handling and pressure pump subsystems for both the AFC actuators and measurement equipment.

Recommendations for improvement of the design are mainly focused on calculation methods. Estimation techniques concerning the weight and cost should be updated with more accurate methods. During the preliminary design phase the majority of analyses was performed using statistical relations. Additionally, structural analysis can be improved usingFinite Element Method (FEM) to find the solutions to boundary value problems. Also for the aerodynamic characteristics of both the original aircraft and the modified test platform, the results can be improved by using Computational Fluid Dynamics (CFD). As an example, the lift and drag estimation did not take interference effects into account, which could be properly modeled by CFD techniques.

Contents

Glossary

a	Speed of Sound [$\frac{m}{s}$].
AR	Aspect ratio [-].
b	Wing span [m].
C_{D0}	Zero-lift drag coefficient [-].
C_{D0_f}	Zero-lift drag coefficient for the fuselage [-].
C_{D0_w}	Zero-lift drag coefficient for the wing [-].
C_{D_i}	Induced drag coefficient [-].
C_f	Skin friction drag coefficient [-].
C_{f_f}	Skin friction drag coefficient of the fuselage [-].
C_{f_w}	Skin friction drag coefficient of the wing [-].
c_j	Specific fuel consumption [$\frac{kg \cdot hr}{N}$].
C_L	Lift coefficient [-].
c_r	Root chordlength [m].
c_{root}	Root chordlength [m].
c_{tip}	Tip chordlength [m].
D	Drag [N].
d_f	Fuselage diameter [m].
E	Loiter time [s].
e	Oswald's span efficiency factor [-].
f_{ld_e}	Factor of fineness ratio of the engine [-].
f_M	Mach factor [-].
f_{tc_w}	Chordthickness factor [-].
g	Gravity acceleration [$\frac{m}{s^2}$].
I_{nm}	Area moment of inertia across m in n [m^4].
L	Lift [N].
l	Length [m].
$\frac{L}{D}$	Aerodynamic efficiency [-].
M	Bending Moment [Nm.
m	Mass [kg].
M_{cruise}	Cruising Mach Number [-].
n_e	Number of engines [-].
R	Range [m].
Re	Reynolds number [-].
S	Surface area [m^2].
S_{wet_w}	Wetted area [m^2].
T	Torsion [Nm].
$\frac{t}{c}$	Thickness-to-chord ratio [-].
V	Shear Force [N].
V	True airspeed [$\frac{m}{s}$].
V_{cruise}	Cruise speed [$\frac{m}{s}$].
W	Weight [N].
r	Horizontal distance between aerodynamic centres of main wing and horizontal tailplane [m].
x	Horizontal Distance [m].
x_{cg}	Centre of Gravity Location [m].
z	Vertical Distance [m].
α	Angle of Attack [°].
β	Mach correction factor [-].
Γ	Dihedral angle [°].

δ	Displacement [m].
ϵ	Downwash Angle [$^\circ$].
Λ	Sweep angle [$^\circ$].
λ	Taper ratio [-].
μ	Dynamic air viscosity [$\frac{kg}{m \cdot s}$].
ρ	Atmospheric density [$\frac{kg}{m^3}$].

List of Abbreviations

AFC	Active Flow Control.
APU	Auxiliary Power Unit.
ATC	Air Traffic Control.
AWACS	Airborn Warning And Control Systems.
BR	Baseline Report.
CATIA	Computer Aided Three-dimensional Interactive Application.
CCD	Charge Coupled Device.
CFD	Computational Fluid Dynamics.
CPP	Central Power Point.
DH	Data Handling.
EPU	Electrical Power Unit.
FBS	Functional Breakdown Structure.
FBW	Fly-By Wire.
FEM	Finite Element Method.
FFBD	Functional Flow Block Diagram.
FMS	Flight Management System.
FR	Final Review/Report.
HEIE	High Environmental Impact Elements.
HRM	Human Resource Management.
LEIE	Low Environmental Impact Elements.
MAC	Mean Aerodynamic Chord.
MHR	Man Hour.
MNS	Mission Need Statement.
MS	Measurement Systems.
MTR	Mid Term Review/Report.
OBS	Organisation Breakdown Structure.
OEW	Operative Empty Weight.
PCU	Power Control Unit.
PFS	Primary Flight Systems.
PIV	Particle Image Velocimetry.
POS	Project Objective Statement.
PP	Project Plan.
RAMS	Reliability, Availability, Maintainability and Safety analysis.
SDBD	Single Dielectric Barrier Discharger.
TBD	To Be Determined.
USB	Universal Serial Bus.
WBS	Work Breakdown Structure.
WFD	Work Flow Diagram.

List of Figures

List of Tables

1 Introduction

The ever-increasing need for efficiency improvement in aviation has opened the doors for the development of exotic and innovative active flow control concepts. Although systematically tested in laboratories, these ideas are rarely tested in flight. This is due to the inhibitive costs of such an endeavour. This project involves the design of a cost efficient flying test platform for the assessment of wing-based active flow control technologies.

The project was broken down in four different phases to structure the design process. The first phase focused on the organisation and planning of the project. The second phase aimed at analysing the project itself in more detail, and establishing the design requirements. In the third phase, design options were developed and a trade-off was made to select a final concept. The fourth and final phase focused on the preliminary design in which the details of the concept were determined.

This report is the result of the fourth and final project phase which concerns the preliminary design. It aims to bridge the gap between the conceptual and the detailed design. The preliminary design phase concerns the building of a general framework which will be used to carry out the detailed design phase. Actual detailed design and optimization will not be covered in this report. Preliminary calculations will be provided, accompanied by overall system configurations, including schematics, drawings and diagrams. This is the fourth and final report out of a series of four covering the complete project.

The structure of this report is as follows. A description of the project, including the mission need statement and the project objective statement, is given in Chapter 2. Chapter 3 describes the organisation and planning of the whole project. It includes a description of each project phase and the final Gantt chart. The rest of the report has been split up in three major parts.

Part I concerns the project analysis and conceptual design which holds the major results of the second and third project phase. In Chapter 4 the results of the market analysis are presented. Chapter 5 describes the strategy to integrate sustainable development into the design. All the functionalities of the design are listed and analysed in Chapter 6. It includes the functional breakdown structure and the functional flow diagrams. Chapter 7 contains a list of all the design requirements originating from the stakeholders and the analyses presented earlier in the part. The conceptual overview of the final concept is given in Chapter 8. The part is concluded with Chapter 9. This chapter presents a sensitivity analysis, which was used while designing in the next phase of the design.

The results of the preliminary design and the methods that lead to these results are presented in Part II. Each chapter deals with a different aspect of the preliminary design. By means of Class I & II weight estimation techniques, the weight for the different components is estimated. Chapter 10 presents the results of these estimations. Chapter 11 presents the redesign of the main wing and contains a discussion on structural analysis, integration of test-sections and wing box redesign considerations. The test pylon to accommodate additional test wings is described in Chapter 12. The design of the struts and structural reinforcements required to hold the pylon in place are discussed as well. The redesign of the interior is explained in Chapter 13. It gives an electrical and data handling block diagram as well. Chapter 14 provides an analysis of the aerodynamic performance of the design, both with and without the test section mounted. The final chapter of the second part, Chapter 15, holds an overview of the results of the preliminary design.

The third and last part of the report consists of reviews and discussions concerning the design result. It starts with Chapter 16 in which the test plan is presented. It describes how the organisation of a test mission will look like and how the test panels can be integrated on the test platform. The risks on technical performance are mapped and analysed in Chapter 17. The last chapter of this part, Chapter 18 will discuss the work done up to now by checking how many of the requirements are met. It holds a critical view on the design as it stands and provides a discussion on the future development of the project.

Conclusions of the whole project and the methods used are presented in Chapter 19.

2
High-level project description

This chapter will present a description of the assigned project and explains any constraints accordingly. At the start of the project both the Mission Need Statement (MNS) and the Project Objective Statement (POS) need to be determined. These statements give the need as well as the objective to reach for. The MNS and the POS summarise what the stakeholders are expecting from this project.

The MNS and the POS have been defined as follows:

- **MNS:** Provide the aerospace industry with a platform that is able to test current and future wing-based flow control systems in real flight conditions.
- **POS:** Develop an experimental cost-efficient aircraft that is able to test current and future flow control technologies in-flight, by ten students in eleven weeks.

The functions of the system to be designed are greatly influenced by the POS. The platform that will be used to define this system is an aircraft which has as its main purpose to test wing-based flow control systems in real life conditions. The client has defined a set of top level requirements with which the system needs to comply. These constrain the project and specify the major boundaries. The top level requirements are stated as follows:

- **TLR-AD-01:** The aircraft should be based on an existing medium sized aircraft, such as an Embraer 190 or a Boeing 737
- **TLR-AD-02:** The range of the aircraft should be 3000 kilometers
- **TLR-AD-03:** The aircraft should be able to travel at high subsonic and transonic speeds
- **TLR-AD-04:** The aircraft should be able to represent both swept and straight wing configurations
- **TLR-AD-05:** Modifications of the testing equipment should be quick and easy according to the tested technology
- **TLR-AD-06:** The budget for development and modification is 10 million euros
- **TLR-AR-01:** The aircraft should be able to accommodate all major flow control technologies, of which at least suction, synthetic jet, Micro Electro Mechanical Systems (MEMS), piezo and plasma actuators
- **TLR-TS-01:** Real time measurements of flow control performance by means of advanced diagnostic capabilities should be one of the abilities

The explanation regarding the identifier syntax of the requirements will be postponed to Chapter 7.

3 Project organisation and development logic

Project organisation involves planning the complete project. It includes determining management functions, tasks to be performed and the time schedule of the whole project. This chapter holds the discussion of the project organisation. The following results will be presented:

- The division of tasks and responsibilities, summarised in the Organisation Breakdown Structure (OBS).
- The division of the work in cohering parts, summarised in the Work Breakdown Structure (WBS).
- Developing the logical flow of the work, summarised in the Work Flow Diagram (WFD).
- Determining the timeline for the project, summarised in the Gantt chart.

Each of the above will now be discussed in the subsequent sections. Section 3.1 will focus on the OBS, Subsection 3.2.1 will discuss the WBS. Next, Subsection 3.2.2 will look at the WFD and finally Subsection 3.2.3 will provide the timeline for this project by means of the Gantt chart.

3.1 Organisational structuring

At the start of the project, an initial organisation division was made. The main management tasks to be divided fell into two categories, namely the organisation and the technical management. This initial setup was documented in the Project Plan (PP). Throughout the project, new tasks were be introduced and assigned to people.

3.1.1 Organisation management

The organisation management consisted of the following tasks:

- **Chairman:** The chairman was the first point of contact representing the group to externals. This person made sure that communication within the group was possible and that external communications were efficient and effective. The chairman, together with the chief engineers and the planning managers formed the main management of the group.
- **Planning manager:** The planning manager was the main planner of the group and was responsible for the general calender and phase planning. The planning manager defined the time available for a task. Together with the people working on the tasks he discussed if they expected the allocated time to be sufficient. If this was not the case the planning manager would reallocate time. As the project went along it was decided that one person as planning manager was not enough, therefore a group member was added to planning management. The planning managers were also responsible for the Human Resource Management (HRM) together with the resource manager.

- **Secretary:** The secretary was the supervisor for all coordinator positions to be discussed next. Although the person himself was not responsible for the coordinators work, he was expected to keep an eye on the coordinators to determine whether their work was done properly. Doing so he has checked if someone is up for the task; if not, he has discussed with the person in question if it would be better to reallocate that person to a different position. The secretary also made minutes of all the status meetings and relevant group meetings.

- **Resource manager:** The resource manager was responsible for all resources (human resources, project budget, design budget). However his main focus was on project and design budget:

  - **Project budget:** The project budget given was in the form of facilities and some project funds, namely a budget for printing. The resource manager was responsible for arranging the required facilities such as conference rooms and monitoring the printing budget. He was also responsible for the office supplies provided by the DSE staff to perform the exercise.
  - **Design budget:** The design budget was the budget given by the client. All the costs concerning the design needed to stay within this budget. The resource manager tried his best to ensure that the total project costs would not go over the budget limit.

  Human resources will be taken care of by the planning manager, the resource manager only performs an assisting role here. Finally the resource manager will also ensure that all group members fill in the logbook correctly and on time.

The coordinators are officially not part of the organisation management, but their work is closely intertwined with it. This is why these tasks are also discussed here. Four coordinators were defined, being:

- **Visual coordinator:** The visual coordinator was responsible for all illustrations required for the reports. The lay-out of the illustrations had to be consistent and of high quality. He was also responsible for the quality of the technical drawing work. For this project it was decided to do this work in Computer Aided Three-dimensional Interactive Application (CATIA).

- **Visio coordinator:** Microsoft Visio is a program which is used to create flow diagrams and breakdown structures. Because of the extensive use of Visio during the project it was decided to appoint a Visio coordinator. This person was responsible to ensure that all the Visio illustrations were consistent in lay-out and of high quality. He was also the person to go to considering problems with Visio.

- **Literature coordinator:** The literature coordinator was responsible to ensure that the Dropbox folder made was clear and had a consistent lay-out. He was also responsible for structuring the report in a logical sequence of chapters and sections.

- **LaTeXcoordinator:** The LaTeXcoordinator ensured that the format and lay-out concerning the reports was done correctly and clearly. The LaTeXcoordinator also debugged the LaTeXcode and he helped group members with questions concerning latex. As the project evolved it was decided that one coordinator was not enough, therefore two more people were assigned to assist the LaTeXcoordinator. The extra task to manage the references was also given to the coordinators.

3.1.2 Technical management

The technical management consisted of two chief engineers. The chief engineers were responsible for the technical quality of the project on an interdepartmental level, between the different subgroups. He was also accountable for the technical quality of the final result. This means that the chief engineers discussed the integration of the separate group parts into one coherent concept. Doing so they checked if the end result is conform the VALID criteria (Verifiable, Achievable, Logical, Integral and Definitive) and oversaw the verification and validation process. The chief engineers also fulfilled the role of quality managers. In this sense they checked the quality of the work produced within the respective groups and the research and design direction taken.

The sub-level of technical management consists of the subgroups working on a part of the aircraft. This part of the organisational breakdown was one which was changed constantly throughout the project due to the fact that as the concept matured the work distribution changed accordingly. In the end three subgroups were created:

- **Wing group:** The wing group was concerned with the (re)design of the wings of the aircraft. They have focused on the structural design of the wing, as well as the implementation of actuators and measurement systems.
- **Pylon group:** The pylon group was concerned with the design of the pylon to be mounted on the aircraft. They have worked on the development of a model test section, as well as the structural implementation of the pylon into the fuselage.
- **Interior group:** The interior group was concerned with the (re)design of the fuselage section. As such they have looked at the fuselage lay-out as well as the electrical installation inside the fuselage.

All of the above can be summarised by means of an OBS. This diagram can be found in Figure 3.1

3.2 Work breakdown and development logic

With the responsibilities and tasks defined the work on the actual project could start. In order to grasp the vast amount of work to be done, the work has been broken down into smaller and coherent parts. These smaller parts were easier to put into a logical order. The division of work is documented by the WBS and the development logic of the work is presented by means of an WFD. Both of these will be discussed in the following.

3.2.1 Work breakdown structure

The project was divided in phases in order to plan the tasks to be performed within the time schedule of the project. It was decided to use the reports and formal reviews set as deliverables by the DSE staff as conclusions for the project phases. Doing so, four project phases were formed, being:

- **Phase 1: Project start-up:** The aim of the first phase has been to organise the project such that work could be started effectively. The mission need statement and project objective statement have been formulated and responsibilities were assigned. The project activities have been discerned and planned. Schedule risks were assessed. The result of this phase is presented in the PP. It holds the strategy and planning that will be used to meet the project goals.
- **Phase 2: Mission analysis:** During the second phase, the project mission has been analysed in more detail. A start has been made with the literature study from which the required functions of the product became more clear. From these functions, requirements have been distilled and at the end of this phase, validated requirements were set. With those requirements and functions in mind, multiple differentiated design option trees have been constructed. Risks and resources for the general mission have been assessed in this phase. The Baseline Report (BR), containing the above, is the milestone concluding the second phase.
- **Phase 3: Conceptual design:** The third phase concerned the conceptual design. The relation between the different interfaces has been investigated and possible concepts were created using the design option trees. Trade-offs and sensitivity analyses at different levels made sure that at the end of the phase the most optimal solution has been found in order to proceed to the next phase. To conclude the phase, the findings were reported in the Mid Term Review/Report (MTR) and reviewed by the tutor and coaches.
- **Phase 4: Preliminary design:** The fourth phase involved the preliminary design of and elaboration on the concept chosen in the previous phase. The output of this stage of the project is the iterated, verified and validated preliminary design of the chosen solution. The functional analysis has been redone and applied to the final design. Also the risk and resource allocation were updated to represent the present solution. This gave an overview of how these factors actually contributed rather than the planned values. Together with what was done in the previous phase, the results are described in this report. The results obtained are also presented in the Final Review and during the Symposium. These two milestones complete the phase and also the project as a whole.

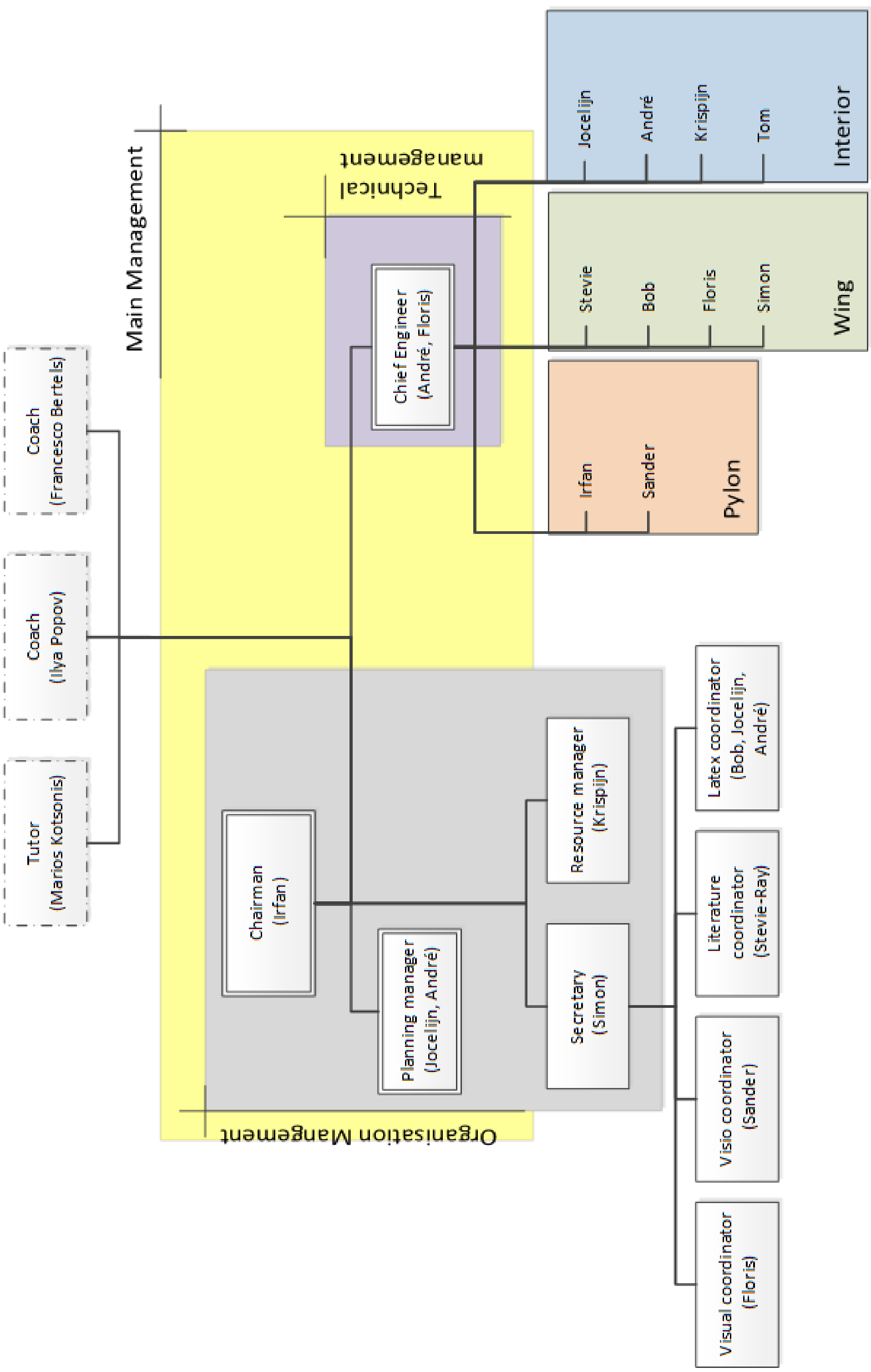


Figure 3.1: The Organisation Breakdown Structure for this project

With the phasing complete the work could be divided per phase. Throughout the process this work breakdown has been updated to represent the most recent situation. The final version of this graph can be found in Figure 3.2. The explanation of the entries is left to Subsection 3.2.2.

3.2.2 Work flow and development logic

With the work divided into smaller parts, these could be put into a logical order, guaranteeing an efficient and logic work flow. Doing so, the bottlenecks in the work were identified and could be mitigated. The final version of the Work Flow Diagrams per phase will be discussed in the following subsubsections.

Phase 1

The project setup phase dealt with both the determination of the fundamental mission objectives and organisational matters. The functional flow is represented in Figure 3.3. The different activities are explained below.

- **Mission Need Statement:** This statement describes the need or opportunity from which the project originates and describes it solution-free.
- **Project Objective Statement:** This provides the answer to the need statement in more concrete terms and is the main goal of the project.
- **Organisation breakdown:** The different roles in the group are determined and assigned to members of the team. It shows the responsibilities and the organisation, so each member of the group knows who to approach with a specific problem.
- **Work structure:** Both a work breakdown structure and work flow diagram are constructed. The former lists the different phases with their respective tasks. The latter presents the logical flow of the assignments in the order they need to be performed.
- **Gantt chart:** The Gantt chart contains the allocated duration of time for the different activities, distinguished in the previous steps, and when they are performed. The start and end date of each task are noted as well. It provides a clear overview of which activities are done concurrently and which need to be finished in order to start with a different one.
- **Schedule risk assessment:** When a Gantt chart is used, the possible risk, induced by starting new tasks ahead of completion of the previous ones, need an assessment. The schedule might be affected when rework is needed and this needs to be accounted for.
- **Procedures:** Rules need to be laid down before starting the actual work. The team procedures constitute general agreements such as the code of conduct and the commitment of working during the appointed hours. The reporting procedures are necessary to accept a common format and language in order to have a uniform report.
- **Sustainable development strategy preparation:** A first approach to the sustainability of the project is already given in an early stage of the project. This is to be fully aware of this important aspect and incorporate it in later design, rather than applying sustainable principles when most of the design is already fixed.

The first phase is concluded by the PP, which is a reference document for all team members during the entire design synthesis exercise.

Phase 2

The second phase of the project was characterised by the analysis of the mission itself. The logical flow of events leading to all necessary information for the conceptual design phase is shown in Figure 3.4.

- **Literature study:** This important task familiarises the group members in-depth with the contemporary practices in experimental techniques of active air flow control and aircraft testing. Furthermore, it is needed to understand all the options that are available today and in the future to make design decisions later on.

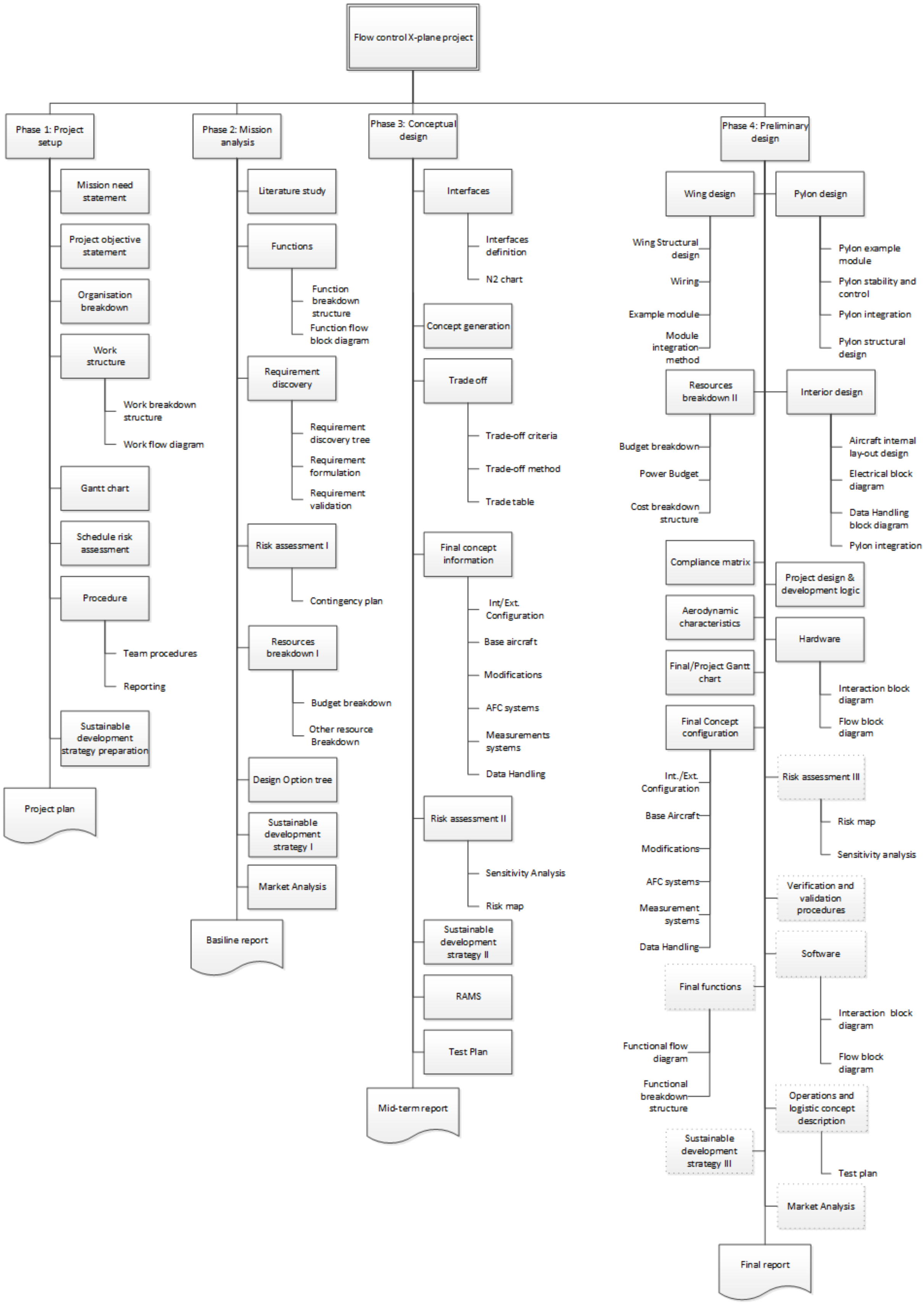


Figure 3.2: The Work Breakdown Structure (WBS) for this project

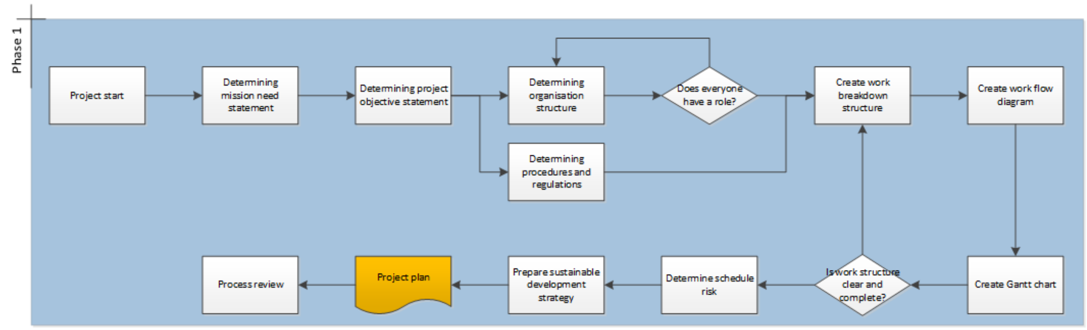


Figure 3.3: The Work Flow Diagram (WFD) of phase 1

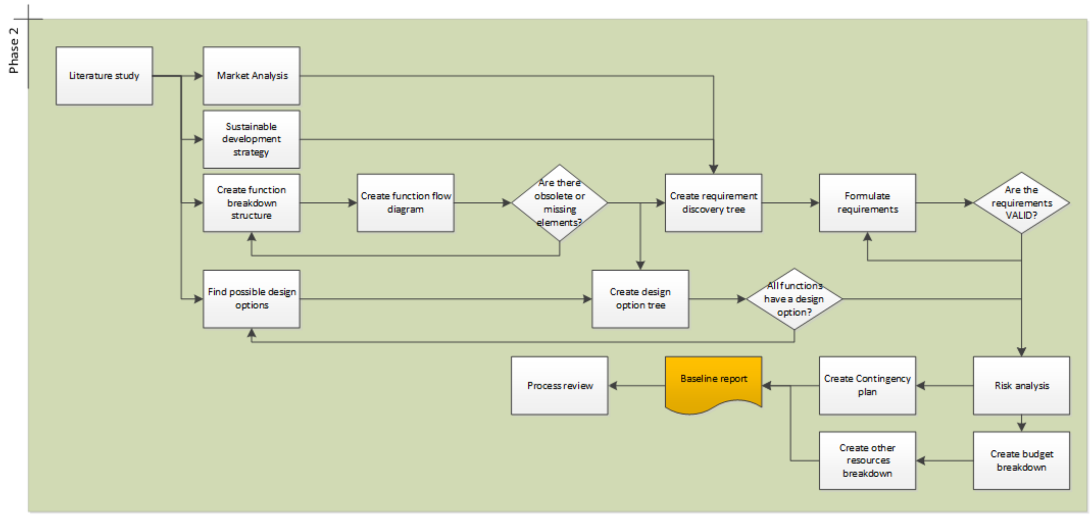


Figure 3.4: The Work Flow Diagram (WFD) of phase 2

- **Market analysis:** The market analysis is the step to determine the impact of the project on the market, but also on what the market expects from this project.
- **Sustainable development strategy:** In the second phase, the sustainable development strategy is further elaborated upon. The basis laid out in the first phase forms the starting point.
- **Functions:** The breakdown structure identifies all functions that the aircraft will need to perform, similar to the work breakdown structure constructed earlier. The flow diagram will then show the order in which the functions need to be happen.
- **Requirement discovery:** Once it is known what the aircraft needs to do (determined during functional analysis), the requirements can be set out to achieve these goals. The discovery tree allows constructive thinking to find requirements not only from the predetermined functions, but also from areas which might have been overlooked. The formulated requirements are then subjected to requirements validation. This means checking whether they are verifiable (e.g. objective, preferably quantitative), achievable (e.g. sufficient resources available), logical (e.g. easy to trace and follow), integral (e.g. complete and all-encompassing) and definitive (e.g. unique and unambiguous).
- **Design option tree:** Concurrent with the functional analysis, a design option tree is already constructed. This allows to focus entirely on the necessary functions to be performed and makes sure to find all possible design options. Later on concepts can be eliminated but this is not the purpose yet.
- **Risk assessment:** This risk assessment pertains the technical performance risks rather than the schedule risk in the first phase. The probability of the occurrence and drivers that cause the risks are identified. The actions to be taken to reduce the probability and effect of the risk events are formulated in a contingency plan.
- **Resource breakdown:** A breakdown of the available resources is made to help estimating which resource should be allocated to a certain task. The workload should be feasible with the available resources. If this is not the case, the scope might be reduced, resources reassigned or different methods might be utilised.
- **Process review:** After the BR is handed in, the clients and stakeholders can input information they are missing.

The second phase was concluded by the BR which included the requirements specification with a requirements discovery tree, functional flow diagram and a functional breakdown of the system.

Phase 3
The third phase of the project concerned the conceptual design. The work flow diagram shown in Figure 3.5 provides a visual overview of how the selection of the final concept was executed.

- **Interfaces:** In this step the relation between the different interfaces is investigated. The influence of the various aspects on each other is analyzed in order to gain more insight in the dependence of interfaces of each of the possible designs. N2 charts help visualise these connections.
- **Possible concepts:** The design option tree, constructed in Phase 2, is evaluated. First non-feasible options are removed. Second the best options are chosen and combined in a concept. All choices are documented so they are traceable later on.
- **Trade-off:** To determine the optimal concept to move forward into the preliminary design phase, a trade-off needs to be made. The criteria are predetermined which will be used to assess the remaining design concepts. The trade-off method is selected and used to construct the trade table which will ultimately lead to the chosen design.
- **Prepare concept details:** The phase is concluded with the preparation of the selected concept details. Verification and validation of the concept is performed and when these are passed, some fundamental aspects of the concept are already provisionally determined. These aspects include the Reliability, Availability, Maintainability and Safety analysis (RAMS), the basic concept lay-out and specific technical risk assessment.

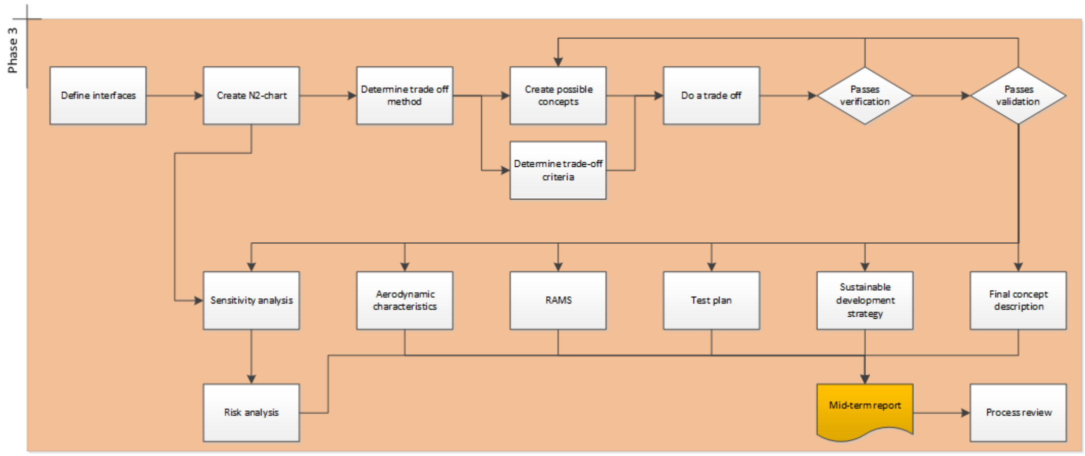


Figure 3.5: The Work Flow Diagram (WFD) of phase 3

- **Sensitivity analysis:** After the concept is designed, a sensitivity analysis will determine the level of influence subsystems have on each other.
- **Risk analysis:** After the sensitivity analysis, the risk of every choice made in the concept needs to be analysed. With it, the bottlenecks can be determined.
- **RAMS:** Next to a risk analysis, the reliability of the system needs to analysed to determine the possible causes of failures during operation and design.
- **Test plan:** The procedures of the design during operational phase will be described in the test plan. After functional analysis, this step will implement procedures and checks to make sure that the test are done correctly and useful data is obtained.
- **Sustainable development strategy:** The second sustainable development strategy will elaborate upon the how the final concept will be designed sustainably.
- **Process review:** After the Mid-term report is handed, in the clients and stakeholders can input information they are missing.

The third phase was concluded by the MTR. It was build on the research done for the BR. The design tree was critically reviewed and the end result of the trade-off process was used in the following phase.

Phase 4

The fourth phase marked the start of calculations to support the design. The functions and requirements established in previous sections were effectively used to investigate the chosen concept in more detail. The further actions taken can be seen in Figure 3.6. The explanation of the tasks is given below.

- **Resource breakdown:** After the concept has been created, the resources available should be redivided to set boundaries in terms of weight and cost.
- **Aerodynamic characteristics:** With the concept known, the first preliminary estimation can be done concerning the aerodynamic characteristics. These include the generation of the lift-drag polars, the moment curves and initial checks on the location of the neutral point for the sake of stability and controllability.
- **Wing design:** With some basic information set up and the final concept known, the first design cycle can be initiated. This first cycle will look at the structural redesign of the wing and give answer to questions such as how the Active Flow Control (AFC) systems and wiring will be integrated in the existing wing structure.
- **Pylon design:** For the final concept, a pylon system has been chosen. The first design cycle for the pylon consisted of tasks such as the management of stability and control requirements as well as the structural sizing and implementation of the pylon.

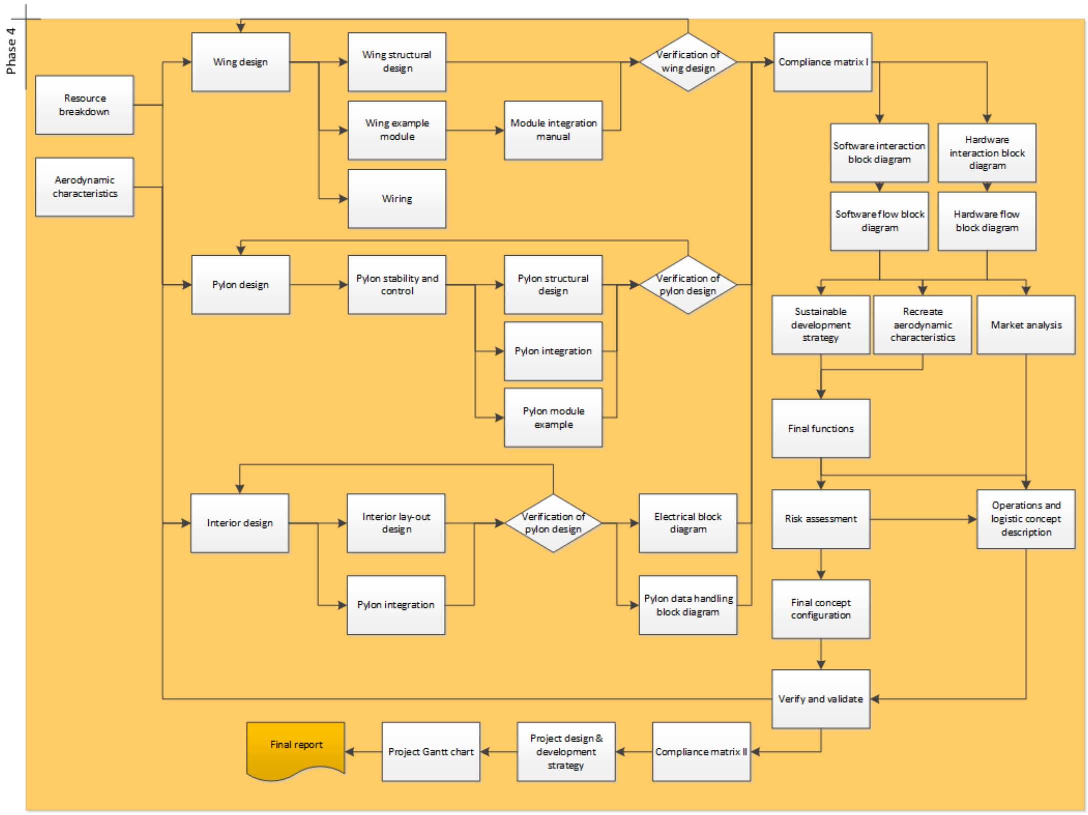


Figure 3.6: The Work Flow Diagram (WFD) of phase 4

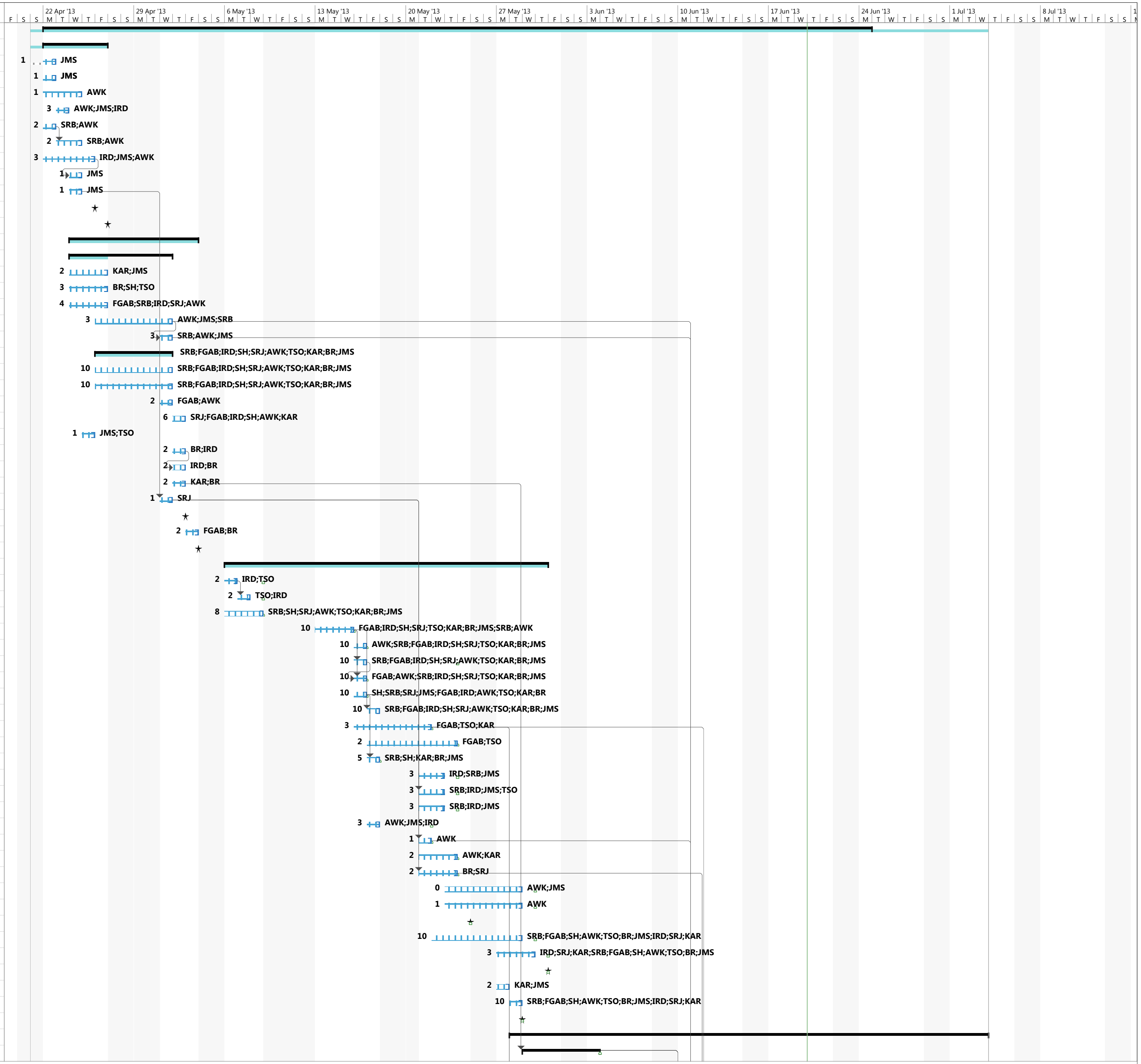
- **Interior design:** The interior redesign consists of the integration of the pylon within the fuselage and the interior lay-out.
- **Electrical block diagram:** The electrical block diagram will illustrate the lay-out of the electrical system installed onboard for both the measurement systems and the AFC systems.
- **Data handling block diagram:** The data handling block diagram will show how the data handling system transfers data throughout the system.
- **Compliance matrix:** A compliance or compliancy matrix is a matrix which illustrates which requirements have been met and which not. The compliance matrix is made twice: immediately after the three groups (interior, wing and pylon) finish their design and also at the very end of the process when all parts have been integrated to form one coherent concept.
- **Sustainable development strategy:** The next step in sustainable development is to document how the system is staying within the already provided strategy and how future steps can be taken to stay within this strategy.
- **Operations and logistics concept description:** The test plan created for the MTR will be complemented with more detailed information on how the aircraft can be used.
- **Functions:** At this point the functions determined for the aircraft will be revised to appropriately fit the final concept chosen.
- **Risk assessment:** After going through the first design cycle the risk assessment prepared for the MTR will be revised to accommodate new risks found in the final project phase.
- **Project design & development strategy:** At the end of the last project phase the project organisation will be complemented by the design and development logic followed during the project.

The fourth phase was concluded with the end of the preliminary design, which was the endpoint for this project. The first design cycle has been performed, verified and validated. The Final Review/Report (FR) presents the result from the fourth phase, together with all topics mentioned for the MTR. The final part to be done is to prepare for the Symposium presentation.

3.2.3 Project timeline development

With the dates for milestones set and the initial work breakdown structure done, the first version of the Gantt chart could be created. This chart showed the estimated timeline for the project and also graphically kept track of the progress made during the project. This document was updated frequently so that it could be used as reference document for all group members. The final Gantt chart, showing the entire project timeline can be found on the next page.

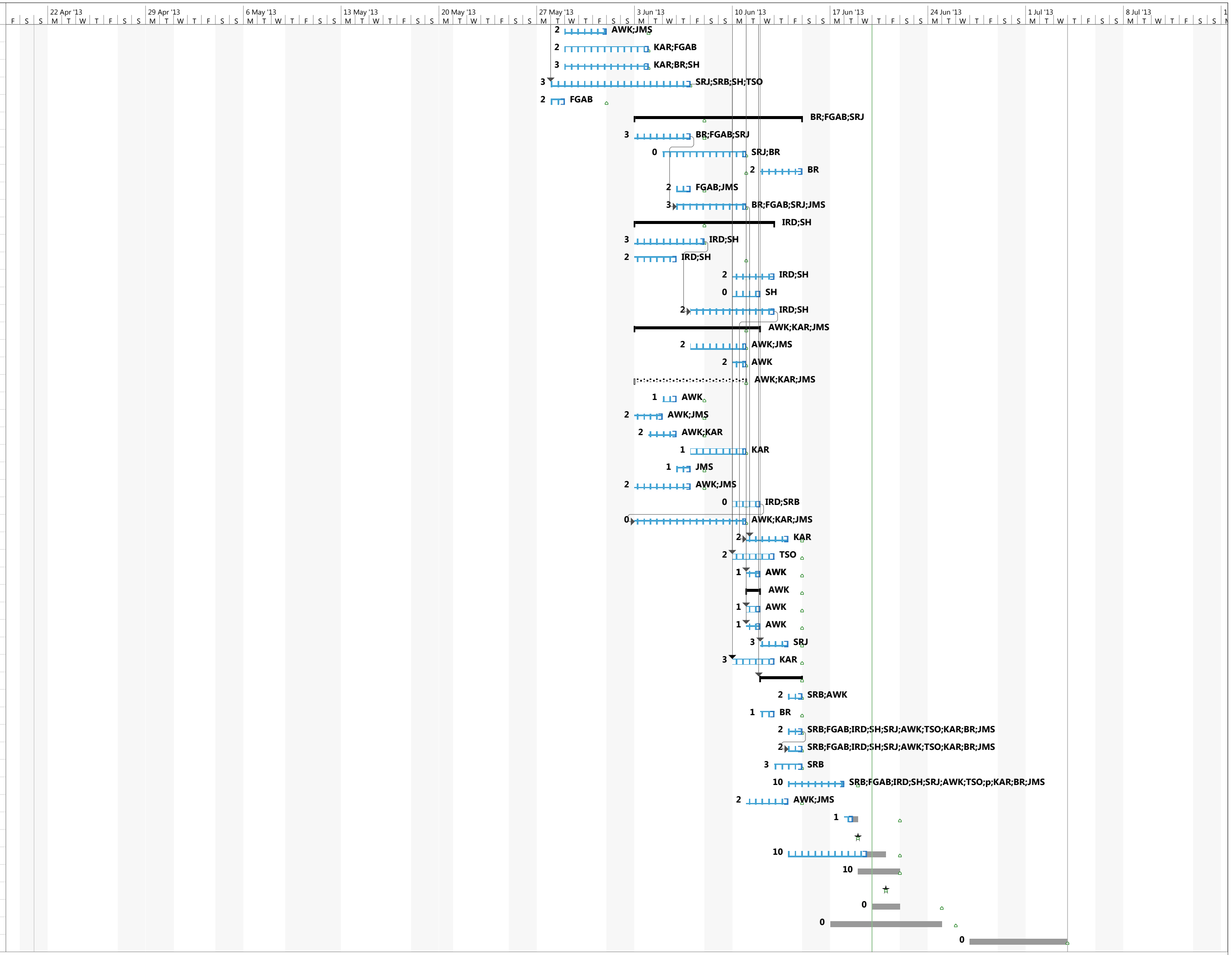
ID	Indicators	#	Task Name	Duration	Start	Finish
1		0	**Design the aircraft**	**41 days**	**Mon 22-4-13**	**Mon 24-6-13**
2	✓	0	**Project Setup**	**5 days**	**Mon 22-4-13**	**Fri 26-4-13**
3	✓	1	Mission Need Statement	1 day	Sun 21-4-13	Mon 22-4-13
4	✓	1	Project Objective Statement	1 day	Mon 22-4-13	Mon 22-4-13
5	✓	1	Determine Organisational structure	3 days	Mon 22-4-13	Wed 24-4-13
6	✓	3	Determine Procedures and Regulations	1 day	Tue 23-4-13	Tue 23-4-13
7	✓	2	Create WBS	1 day	Mon 22-4-13	Mon 22-4-13
8	✓	2	Create WFD	2 days	Tue 23-4-13	Wed 24-4-13
9	✓	3	Prepare Gantt Chart	4 days	Mon 22-4-13	Thu 25-4-13
10	✓	1	Schedule Risk Analysis	1 day	Wed 24-4-13	Wed 24-4-13
11	✓	1	Prepare Sustainable Development Strategy	1 day	Wed 24-4-13	Wed 24-4-13
12	✓	0	Draft Project plan	0 days	Thu 25-4-13	Thu 25-4-13
13	✓	4	Project Plan	0 days	Fri 26-4-13	Fri 26-4-13
14	✓	0	**Mission Analysis**	**6 days**	**Wed 24-4-13**	**Fri 3-5-13**
15	✓	0	**Literature Study**	**4 days**	**Wed 24-4-13**	**Wed 1-5-13**
16	✓	2	Aircraft Design	3 days	Wed 24-4-13	Fri 26-4-13
17	✓	3	Flow Control Systems	3 days	Wed 24-4-13	Fri 26-4-13
18	✓	4	Test Flight Procedures	3 days	Wed 24-4-13	Fri 26-4-13
19	✓	3	Create FBS	2 days	Fri 26-4-13	Wed 1-5-13
20	✓	3	Create FFBD	1 day	Wed 1-5-13	Wed 1-5-13
21	✓	10	**Create Requirements Discovery**	**2 days**	**Fri 26-4-13**	**Wed 1-5-13**
22	✓	10	RDT	2 days	Fri 26-4-13	Wed 1-5-13
23	✓	10	Formulate Requirements	2 days	Fri 26-4-13	Wed 1-5-13
24	✓	2	Validate Requirements	1 day	Wed 1-5-13	Wed 1-5-13
25	✓	6	Create Design Option Tree	1 day	Thu 2-5-13	Thu 2-5-13
26	✓	1	Market analysis	1 day	Thu 25-4-13	Thu 25-4-13
27	✓	2	Risk Assessment	1 day	Thu 2-5-13	Thu 2-5-13
28	✓	2	Create Contingency Plan	1 day	Thu 2-5-13	Thu 2-5-13
29	✓	2	Resources Breakdown	1 day	Thu 2-5-13	Thu 2-5-13
30	✓	1	Sustainable Development Strategy	1 day	Wed 1-5-13	Wed 1-5-13
31	✓	10	Draft Baseline	0 days	Thu 2-5-13	Thu 2-5-13
32	✓	2	Prepare baseline review	1 day	Fri 3-5-13	Fri 3-5-13
33	✓	10	Baseline Report	0 days	Fri 3-5-13	Fri 3-5-13
34	✓	0	**Conceptual Design**	**16 days**	**Mon 6-5-13**	**Thu 30-5-13**
35	✓	2	Define Interfaces (13)	1 day	Mon 6-5-13	Mon 6-5-13
36	✓	2	Create N2 Chart (13)	1 day	Tue 7-5-13	Tue 7-5-13
37	✓	8	Finalize DOT's	3 days	Mon 6-5-13	Wed 8-5-13
38	✓	10	Create Possible Concepts (10, 17)	3 days	Mon 13-5-13	Wed 15-5-13
39	✓	10	Determine Trade-off Criteria (14, 15, 16)	1 day	Thu 16-5-13	Thu 16-5-13
40	✓	10	Perform Trade-off (17)	1 day	Thu 16-5-13	Thu 16-5-13
41	✓	10	Verify Concept (17)	1 day	Thu 16-5-13	Thu 16-5-13
42	✓	10	Validate Concept (17)	1 day	Thu 16-5-13	Thu 16-5-13
43	✓	10	Report concepts	1 day	Fri 17-5-13	Fri 17-5-13
44	✓	3	Reporting final concept	3 days	Thu 16-5-13	Tue 21-5-13
45	✓	2	Configuration int./ext.	4 days	Fri 17-5-13	Thu 23-5-13
46	✓	5	Research stakeholder requests for found concept	1 day	Fri 17-5-13	Fri 17-5-13
47	✓	3	Perform Sensitivity Analysis (27)	2 days	Tue 21-5-13	Wed 22-5-13
48	✓	3	Risk Assessment (9)	2 days	Tue 21-5-13	Wed 22-5-13
49	✓	3	Interface N2 charts	2 days	Tue 21-5-13	Wed 22-5-13
50	✓	3	Method and organization	1 day	Fri 17-5-13	Fri 17-5-13
51	✓	1	Sustainable development Strategy	1 day	Tue 21-5-13	Tue 21-5-13
52	✓	2	RAMS characteristics	3 days	Tue 21-5-13	Thu 23-5-13
53	✓	2	Operations and Logistics Description (18)	3 days	Tue 21-5-13	Thu 23-5-13
54	✓	0	Project organisation revision	4 days	Thu 23-5-13	Tue 28-5-13
55	✓	1	Create new project organisation charts (1,2,3,4)	4 days	Thu 23-5-13	Tue 28-5-13
56	✓	0	Draft midterm review report	0 days	Fri 24-5-13	Fri 24-5-13
57	✓	10	Editing draft midterm review report	5 days	Wed 22-5-13	Tue 28-5-13
58	✓	3	Prepare Midterm Review Presentation	3 days	Mon 27-5-13	Wed 29-5-13
59	✓	10	Midterm Review presentation	0 days	Thu 30-5-13	Thu 30-5-13
60	✓	2	Individual part preperation	1 day	Mon 27-5-13	Mon 27-5-13
61	✓	10	Individual part	1 day	Tue 28-5-13	Tue 28-5-13
62	✓	0	Midterm Review Report	0 days	Tue 28-5-13	Tue 28-5-13
63		0	**Preliminary design**	**27 days**	**Tue 28-5-13**	**Wed 3-7-13**
64	✓	2	**Resource breakdown**	**4 days**	**Wed 29-5-13**	**Mon 3-6-13**



Project: Gantt_Chart

Task	Summary	Rolled Up Task	External Tasks	Inactive Milestone	Duration-only	Start-only	Progress		
Split	Project Summary	Rolled Up Milestone	External Milestone	Inactive Summary	Manual Summary Rollup	Finish-only			
Milestone	Group By Summary	Rolled Up Progress	Inactive Task	Manual Task	Manual Summary	Deadline			

ID	i	Task Mode	#	Task Name	Duration	Start	Finish
65	✓		2	Power budget	3 days	Wed 29-5-13	Fri 31-5-13
66	✓		2	Cost budget	4 days	Wed 29-5-13	Mon 3-6-13
67	✓		3	Weight budget	4 days	Wed 29-5-13	Mon 3-6-13
68	✓		3	Aerodynamic characteristics	8 days	Tue 28-5-13	Thu 6-6-13
69	✓		2	Prepare tools design	1 day	Tue 28-5-13	Tue 28-5-13
70	✓		**3**	**Wing design**	**10 days**	**Mon 3-6-13**	**Fri 14-6-13**
71	✓		3	Structural design	4 days	Mon 3-6-13	Thu 6-6-13
72	✓		0	Module integration	4 days	Wed 5-6-13	Mon 10-6-13
73	✓		2	Example module	3 days	Wed 12-6-13	Fri 14-6-13
74	✓		2	Wiring	1 day	Thu 6-6-13	Thu 6-6-13
75	✓		3	Reporting	3 days	Thu 6-6-13	Mon 10-6-13
76	✓		**2**	**Pylon design**	**8 days**	**Mon 3-6-13**	**Wed 12-6-13**
77	✓		3	Pylon Structural design	5 days	Mon 3-6-13	Fri 7-6-13
78	✓		2	Pylon influence on stability and controllability	3 days	Mon 3-6-13	Wed 5-6-13
79	✓		2	Pylon integration	3 days	Mon 10-6-13	Wed 12-6-13
80	✓		0	Pylon module example	2 days	Mon 10-6-13	Tue 11-6-13
81	✓		2	Reporting	4 days	Fri 7-6-13	Wed 12-6-13
82	✓		**3**	**Interior design**	**7 days**	**Mon 3-6-13**	**Tue 11-6-13**
83	✓		2	Electrical block diagram	2 days	Fri 7-6-13	Mon 10-6-13
84	✓		2	Data Handling block diagram	1 day	Mon 10-6-13	Mon 10-6-13
85	✓		**0**	**Aircraft internal lay-out design**	**6 days**	**Mon 3-6-13**	**Mon 10-6-13**
86	✓		1	Cabling	1 day	Wed 5-6-13	Wed 5-6-13
87	✓		2	Engineering booth/Data verifcation possiblities	2 days	Mon 3-6-13	Tue 4-6-13
88	✓		2	Power	2 days	Tue 4-6-13	Wed 5-6-13
89	✓		1	Pressure	2 days	Fri 7-6-13	Mon 10-6-13
90	✓		1	PIV	1 day	Thu 6-6-13	Thu 6-6-13
91	✓		2	Fuel system	4 days	Mon 3-6-13	Thu 6-6-13
92	✓		0	Pylon integration	2 days	Mon 10-6-13	Tue 11-6-13
93	✓		0	Reporting	6 days	Mon 3-6-13	Mon 10-6-13
94	✓		2	Compliance matrix	3 days	Tue 11-6-13	Thu 13-6-13
95	✓		2	Second aerodynamic characteristics	3 days	Mon 10-6-13	Wed 12-6-13
96	✓		1	Sustainable Development Strategy	1 day	Tue 11-6-13	Tue 11-6-13
97	✓		**1**	**Final functional analysis**	**1 day**	**Tue 11-6-13**	**Tue 11-6-13**
98	✓		1	Function breakdown structure	1 day	Tue 11-6-13	Tue 11-6-13
99	✓		1	Function flow block diagram	1 day	Tue 11-6-13	Tue 11-6-13
100	✓		3	Final concept configuration	2 days	Wed 12-6-13	Thu 13-6-13
101	✓		3	Final resource breakdown	3 days	Mon 10-6-13	Wed 12-6-13
102	✓		**3**	**Operations and logistic concept description**	**3 days**	**Wed 12-6-13**	**Fri 14-6-13**
103	✓		2	Test plan update	1 day	Fri 14-6-13	Fri 14-6-13
104	✓		1	Panel integration manual	1 day	Wed 12-6-13	Wed 12-6-13
105	✓		2	Verification	1 day	Fri 14-6-13	Fri 14-6-13
106	✓		2	Validation	1 day	Fri 14-6-13	Fri 14-6-13
107	✓		3	Risk assessment	2 days	Thu 13-6-13	Fri 14-6-13
108	✓		10	Design summary	2 days	Fri 14-6-13	Mon 17-6-13
109	✓		2	Project & development strategy	3 days	Tue 11-6-13	Thu 13-6-13
110			1	Final gantt chart	1 day	Tue 18-6-13	Tue 18-6-13
111			0	Draft final report	0 days	Tue 18-6-13	Tue 18-6-13
112			10	Editing	5 days	Fri 14-6-13	Thu 20-6-13
113			10	Final review preparation	3 days	Wed 19-6-13	Fri 21-6-13
114			0	Final report	0 days	Thu 20-6-13	Thu 20-6-13
115			0	Jury Summary	2 days	Thu 20-6-13	Fri 21-6-13
116			0	Symposium poster	6 days	Mon 17-6-13	Mon 24-6-13
117			0	Executive Summary	5 days	Thu 27-6-13	Wed 3-7-13



Project: Gantt_Chart

Task	Summary	Rolled Up Task	External Tasks	Inactive Milestone	Duration-only	Start-only	Progress
Split	Project Summary	Rolled Up Milestone	External Milestone	Inactive Summary	Manual Summary Rollup	Finish-only	
Milestone	Group By Summary	Rolled Up Progress	Inactive Task	Manual Task	Manual Summary	Deadline	

I

Project analysis and conceptual design

4
Market analysis

This chapter is devoted to describing the components of a market analysis as well as a step by step analysis of the components. This project will introduce a completely new product. The platform will be able to serve the scientific community that is concerned with the development of AFC systems. The development and exploitation of new aircraft usually involves a large amount of money. Therefore, it is important to analyse what the market for this new aircraft is, and how the behaviour and interests of this market will influence the design decisions to be taken.

4.1 Market analysis composition

The project aims at designing a new test platform able to test flow control techniques. The resulting product will be unique in its sort. Therefore allocating this product within existing markets is challenging. The success of the product will eventually depend on how usable it will be for its market. The target market is identified as the active flow control testing market, since the aim of this project is to design a product that is able to fulfil their needs. The second thing which needs to be defined is who is considered the client for this project. Even though the assignment was set by the principal tutor and coaches, serving as stakeholders, they will not be using the aircraft itself. The POS stated that a product will be designed to serve the AFC developers. Therefore the scientific community is one of the clients. However, large aircraft manufacturers such as Boeing and Airbus might be so impressed with the results obtained during this project that they would like to use the test platform as well. Therefore the second client(s) identified are the aircraft manufacturers. With the market and clients defined, the analysis was continued with the step by step analysis according to [1]:

- Market Size
- Market Growth Rate
- Market Profitability
- Market Trends
- Industry Cost Structure
- Key Success Factors

4.2 Market size

Requirements for efficiency improvement in aviation are ever increasing. This makes the development and use of active flow control systems an increasingly attractive option. In-flight testing will be crucial for successful implementation of these systems in future aircraft. Current test aircraft involving flow control systems emphasise on the use of one specific flow control system, rather than multiple.

Development of AFC technology is primarily done by the scientific community. In the current market a versatile test platform, such as the one designed here, does not exist. As a consequence a large need is expected for such a platform which is also affordable for these AFC developers.

4.3 Market growth rate

There is very little data available which describes the amount of test aircraft produced for a given time period. Therefore, it is difficult to determine the growth rate of this market. In general it can be said that the design and exploitation of test aircraft is very expensive. Since such aircraft are only rarely created it is very unlikely that the market should be considered as a growing market.

4.4 Market profitability

The design and development of test aircraft is a very costly process. Test aircraft do not enter service as regular civil aircraft do and therefore do not generate revenue at the start of their service. The test aircraft will not be designed to generate profit. Rather, this design will enable AFC developers to perform tests in real-life conditions. These tests enable the client to sell their own AFC products. The barriers to enter the test-aircraft market are relatively low and there is little rivalry between individual companies due to the fact that different test aircraft often test completely different functions. Considering that the product under development right now offers new testing features, there are two methods by which this product can create income. Either selling the product to an interested party, or by leasing the test platform to clients so they can mount and test their own AFC systems.

4.5 Market trends

Up till now it has been customary in the flow control test aircraft market to (re)design an aircraft for one specific system. Therefore the aircraft which is under design in this project has the opportunity to fill a gap which currently exists in the market. This product will attempt to demonstrate that multiple flow control systems can be mounted and tested on a single test platform.

Testing new AFC systems and flight testing in general is for obvious reasons mostly kept secret. Large airframe manufacturers do not publish results of such experiments. Therefore information on these studies proves hard to find. However, one institute, the NLR, sells itself as a flight test operations platform. It is possible to hire the services of the NLR to provide data of desired experiments. This test facility is unique and most companies do not provide such a testbed. Airbus, Boeing and other large companies will most likely design their own aircraft, specific for the desired tests.

Other than providing a missing product this design opens up possibilities for collaboration between industry and universities. Such collaborations have already been performed for a wide range of purposes. For example MIT, the Aurora Flight Sciences and Pratt and Whitney are redesigning a Boeing $737 - 800$ to obtain the D8 aircraft, realising large reductions in noise and drag [2]. Another example is a project regarding research into flow control by Boeing and the Notre-Dame University [3]. The goal here is to perform more research into aero-optics and plasma flow control. As a final note it has been found that PhD students regularly do research for their promotion work cooperating with aircraft manufacturers (for example, the students use the manufacturers facilities to perform certain tests).

This project will take cooperation between university and the industry to the next level. The strength of this universal platform lies in the fact that the possible range of customers is substantial. The platform can be used within AFC developers, by companies such as NLR and DLR, but can also be of interest to aircraft manufacturers such as Airbus and Boeing. In this way university can directly contribute to the progress and new design processes occurring in industry at the moment.

4.6 Industry cost structure

Test aircraft do not generate revenue in the same way as regular civil aircraft do. The industry cost structure is usually assessed by inspecting the value chain of the industry. This chain shows where value is added to products. The costs required for the design of flow control test aircraft are substantial. This is something to keep in mind when determining the selling and leasing prices of the aircraft.

4.7 Key success factors

One of the key success factors for this project will be the adaptability of the design. Most existing test aircraft are able to test only one technology. The design will have the advantage that multiple technologies can be tested (simultaneously). Another key success factor is the fact that the design should have the lowest costs possible. Existing aircraft usually are very expensive. Thus, to keep an advantage with respect to already existing competitors the low costs will also ensure an advantage leading to greater success of the design. The third key success factor is that the design focuses on modularity. Flow control actuators should be mounted onto the aircraft with ease. This rapid mounting ensures efficient testing programs to be executed. The final key success factor is that the design accounts for future technological progress, such that future AFC actuators can also be mounted onto the platform.

5 Sustainable development strategy

This chapter explains how the design will influence the sustainability and how sustainability is integrated within the design.

5.1 Impact on aerospace industry

Active flow control technology holds the possibility to increase the efficiency of future commercial aircraft. For example, replacing complex high-lift devices with AFC actuators is very appealing, since it would decrease the weight of the aircraft, enhance the lifting capabilities, decrease drag, and reduce noise. It could decrease total fuel consumption with only a few percent, but would lead to a decrease of direct operating costs of millions of euros a day. The fuel savings would directly lead to a decrease of the environmental impact. Though systematically tested in laboratories, the AFC technology did not yet find its way to the industry. This design could be the missing link in the development of wing-based AFC applications. The design will allow accommodation of AFC actuators in development, such that they can be developed for the industry. The impact on sustainable development of the industry is therefore large. If the design is able to prove its modularity, efficient testing, and low cost, it will allow for a breakthrough in AFC development.

5.2 Sustainability within the design

The environmental impact of this concept is limited from the start. The amount of aircraft produced from this project will be minimal. In addition, the amount of flights will be much less than regular civil aircraft. This means that reducing the exhausts during flight has a minimal influence on the environmental footprint of civil aircraft. It is aimed to take future developments of AFC actuators and measurement equipment into account to improve the sustainability of the design.

With respect to using High Environmental Impact Elements (HEIE) in the structure no choices have been made in the concept. Since a base aircraft is chosen, a lot of HEIE are already present on the aircraft. It should be determined however, whether the HEIE can be replaced by low environmental impact elements Low Environmental Impact Elements (LEIE) that are now available. Regarding new systems, the type of materials usage will be taken into account from the start.

To keep track of all the impacts and wastes, a list should be created that shows the amount of materials used and wasted. At the same time an indicator should be given to the HEIE. In this way it can easily be determined if too much waste is created or if too much HEIE are used.

The combination of variety and integration of AFC systems demands a proper test system to collect the data of good quality. This data can the be used to show the aerospace industry the effects of AFC systems on large scale in real flight conditions. If the test are successful, the the aerospace industry can try to implement these system in future aircraft. Therefore, this concept aircraft can help reducing the footprint of future aircraft, and create a more sustainable future.

6 Functional analysis

This chapter focuses on the functional analysis of the mission that is performed. The mission is divided into different main phases and subdivided by the various functions which need to be performed during each specific phase. The subsequent section explains the flow of these events for every phase in more detail. The complete functional flow block diagram is presented at the end of the chapter to serve as an overview. From this functional analysis, requirements could be deducted.

6.1 Mission phase division

The first thing that is done is a reflection based on the MNS, which was given in Chapter 2. This is done to reduce the risk of missing functions. Every step of the mission is analysed and documented. The system is subdivided in operating phases and each smaller part of the complete problem is investigated. Step by step, the different functions are pinpointed and linked to each other and the phase of interest.

Eight discrete phases are discerned to complete the mission. Each contains activities which concern the main phase objective only. These are the preparation of flight, take-off, flight to test area, flight during testing, measurements to be performed, the return from test area, landing and refurbishing. These would cover the complete set of actions which are needed to satisfy the MNS. The general Functional Breakdown Structure (FBS) is given in Figure 6.1. The analysis of the mission phases and the functions that need to be performed at each step, as seen on the FBS, are explained in the following section.

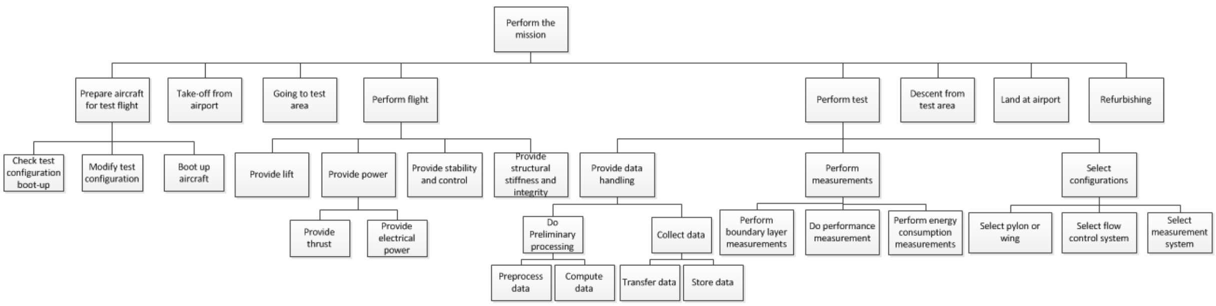


Figure 6.1: Functional Breakdown Structure (FBS) of the mission to be performed

6.2 Mission phase analysis

As part of the functional analysis the mission to be performed has been divided in several phases. This section will discuss the phases both on a high level basis as well as focus on each of the individual sub-phases. This discussion serves the purpose of clarifying the FBS and the Functional Flow Block Diagram (FFBD). The general high-level FFBD with the continuity between the individual phases can be found in Figure 6.2.

The FFBD uses a certain syntax to define the phases of the mission as well as their corresponding sub-phases (and sub-sub-phases). The syntax used is presented in Figure 6.3.

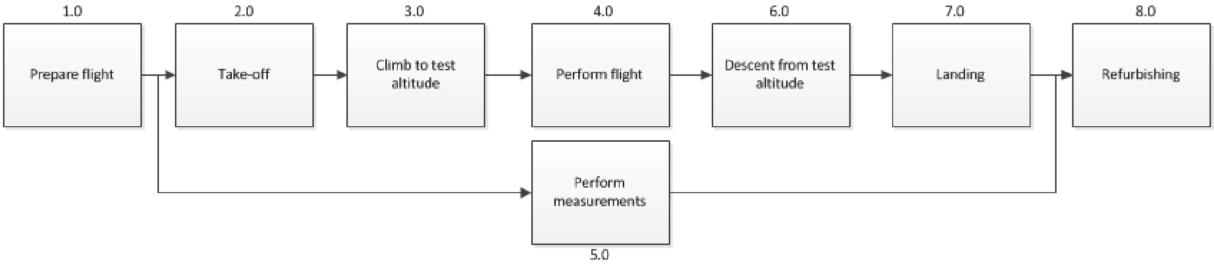


Figure 6.2: Overview of the flow of high level mission phases

Figure 6.3: Legend corresponding to FBS and FFBD

6.2.1 Mission phase 1.0, prepare flight

The first phase of the mission consists of the preparation of the aircraft for the test flight. The aircraft is assumed to be parked 'cold' in a hangar. This phase consists of three sub-phases which are the following:

1. **Modify test configuration:** At first the configuration which needs to be tested should be mounted on the aircraft. This sub-phase consists of only the technical integration of the test configuration. The steps in this sub-phase are highly dependent on the concept and will be elaborated upon after conceptual design. All the other choices of what exactly needs to be integrated has already been decided by the client and is thus not included in the mission phase.
2. **Boot-up aircraft:** Once the correct testing equipment has been mounted the aircraft systems need to be initialised. All systems required to perform the flight and the measurements are switched on and are prepared for use. Such systems include the Flight Management System (FMS) and the pressurisation systems.
3. **Check test configuration boot-up:** With the aircraft booted up it should be checked if all on-board systems have initialised correctly and if all testing equipment required is present. If this is not the case one should go back and either remount the required testing equipment or inspect why the on-board systems failed to boot, fix the problems found and then attempt another boot.

This phase of the FFBD can be found in Figure 6.4.

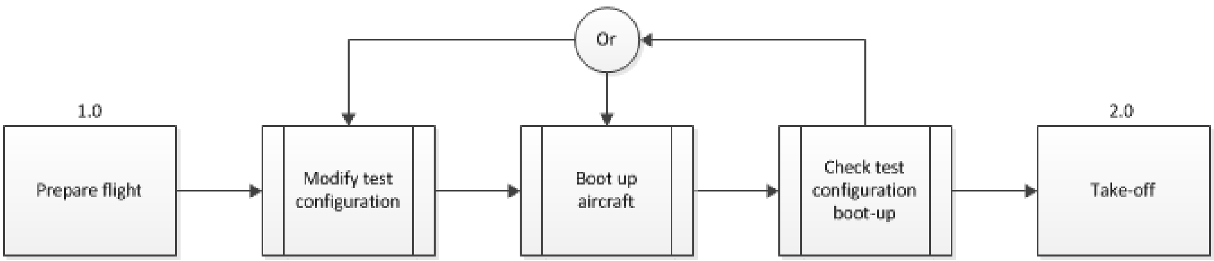


Figure 6.4: Functional Flow Block Diagram (FFBD) of the first mission phase

6.2.2 Mission phase 2.0, take-off

The second phase of the mission consists of the take-off procedure. Here it has been chosen to not elaborate further on the operations required during take-off in order to not focus on details which are of less interest to the mission. Take-off in this sense should be interpreted in a broad sense, also including communication with Air Traffic Control, taxiing to the runway and actually performing the take-off roll.

6.2.3 Mission phase 3.0, going to test area

Following the take-off, the aircraft has to go to the area (location and altitude) where the tests are performed. Here again it is difficult at this stage of the project to completely determine which processes will belong to this phase. For example, it might very well be possible that the aircraft needs to fly to a certain airspace before tests are allowed to be performed (such as military airspace). Functions such as these are also part of this phase. Different testing areas might be used during one flight. It is very possible that this mission phase is re-occurring during a testing mission.

6.2.4 Mission phase 4.0, perform flight

The fourth phase of the mission consists of performing the actual flight. For the present discussion only the functions of the aircraft itself are considered. These are the actions needed to be taken during the test flight, but not strictly related to the measurement of data. Four basic sub-phases are distinguished to perform flight, which are:

1. **Provide lift:** This may seem trivial to mention, but the aircraft needs to produce the required amount of lift to sustain flight for the complete flight envelope.
2. **Provide power:** This sub-phase concerns the provision of power for both the subsystems and the propulsion of the aircraft. Therefore this sub-phase is further divided into the generation of thrust and electrical power.
   (a) **Provide thrust:** The thrust by the engines is used to propel the aircraft.
   (b) **Provide electrical power:** This sub-phase covers the need for electrical power of the complete range of avionics, but also for the active flow control devices and the test measuring equipment.
3. **Provide stability and control:** The aircraft must be both stable and controllable. In order to accurately perform the measurements during the testing stage of the mission, the aircraft needs to be able to fly stable to retain or replicate attitude, altitude and velocity during the experiment. Of course the aircraft should also be controllable for the pilot within the bounds of manoeuvrability.
4. **Provide structural stiffness and integrity:** A main function of the airframe is that it provides the structural stiffness and integrity needed during flight. The aircraft should withstand the forces and moments encountered in all operating conditions.

The flow block diagram of the fourth mission is given in Figure 6.5 and presents the logical flow of functions visually.

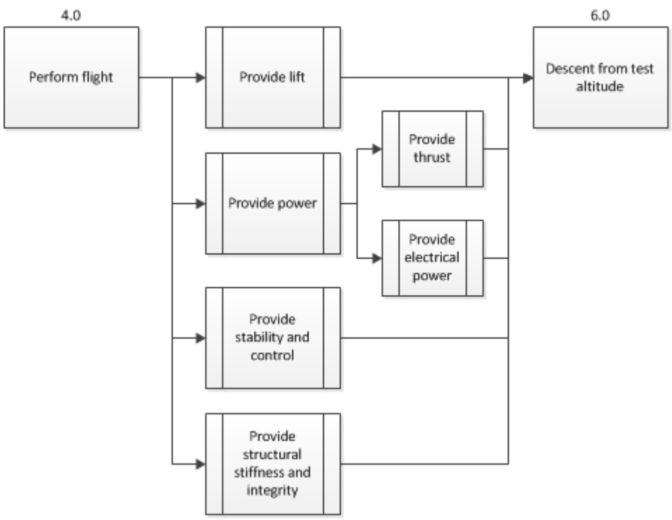


Figure 6.5: Functional Flow Block Diagram (FFBD) of the fourth mission phase

6.2.5 Mission phase 5.0, perform measurements

The fourth phase described in the previous section and the fifth phase described in the following run simultaneously during the mission. The fifth phase focuses on the performing of the measurements. This phase has been divided into the following sub-phases:

1. **Select configurations:** Before the measurements can be performed it should be decided which configurations will be used for the test at hand. The planning for this phase has already been before take-off. These sub-phases again consist of two further sub-sub-phases, which are:
   (a) **Select flow control systems:** The flow control systems which will be switched on for the test should be selected here. It might for example be possible that only one of the installed systems will be tested to make sure there can be no interference between systems, or that multiple systems will be tested concurrently. Tests can also run parallel or separate depending on the preferences of the test crew. This is therefore the first step in selecting the correct configuration.
   (b) **Select pylon or wing:** During flight it is possible to switch between the pylon and the wing for testing. The pylon and the wing cannot be operated at the same time, so a choice has to be made which of the two elements is going to be used for testing. In case the pylon is not present on the aircraft this sub-phase can be ignored.
2. **Regulate flow control systems:** With the proper flow control systems selected, these systems need controlled operation throughout the test. This sub-phase focuses on the regulation of the flow control systems due to changing test conditions.
3. **Perform measurements:** This sub-phase comprises the actual performing of the measurements. The three measurement types to be performed are:
   (a) **Do performance measurements:** In this function, the general performance characteristics will be measured, such as the fuel flow, angle of attack, attitude etc.
   (b) **Perform boundary layer measurements:** Here the boundary layer properties of the aircraft will be measured, such as the pressure distribution. Also later on from these measurements other parameters, like when transition occurs, can be computed.
   (c) **Perform energy consumption measurements:** Energy consumption measurement are the information on what and how much a actuator requires. So for example measuring the current and voltage received by an actuator.
4. **Provide data handling:** During the measurements data is produced. This sub-phase is concerned with the handling of this data as generated during the measurements. This subsystem also consists of a different number of subsidiary functions, being the following:
   (a) **Collect data:** This sub-phase is concerned with the general process of data collection. This sub-phase is again further divided into two tasks, being the following:
      i. **Transfer data:** The data generated during the test needs to be transferred to the storage medium. This is done during this stage of the mission.
      ii. **Store data:** The measured data needs to be stored for detailed analysis after testing. All the data measured will be stored, but also additions made by possible test engineers.
   (b) **Do preliminary processing:** This sub-phase consists of two tasks, both of which should happen in real time:
      i. **Preprocess data:** The preprocessing of the data concerns the transformation of the data into a suitable format which is required for further analysis. The result of this action flows into the next.
      ii. **Computation of data:** The last step of the data enhancement is the actual computation of certain parameters of interest such as the lift and drag coefficient, making use of the preprocessed data. This allows for making changes to the inputs during the test flight. (This is not the real processing, but a initial check to determine if the test has been done correctly and there are no errors).

A graphical representation of this phase can be found in Figure 6.6

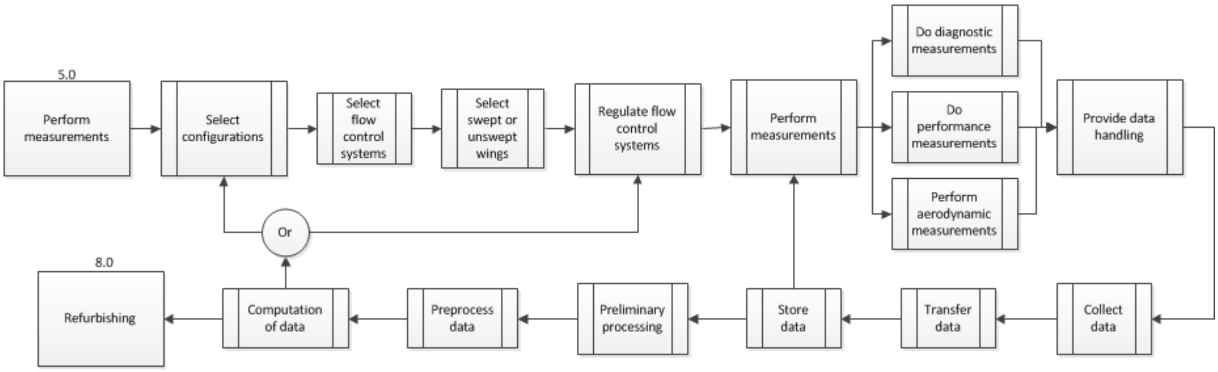


Figure 6.6: Functional Flow Block Diagram (FFBD) of the fifth mission phase

6.2.6 Mission phase 6.0, returning from test area

The sixth phase of the mission consists of returning from the test area to the landing site which comprises the final descent. The precise functions which should be included in this phase (e.g. prepare control surfaces or select thrust setting) are deemed too detailed for this general breakdown and are not further elaborated upon.

6.2.7 Mission phase 7.0, landing

The seventh phase of the mission is named 'Landing', but this procedure concerns more than only the landing roll itself. Apart from every function to perform the landing (e.g. the thrust setting), it also includes functions such as the taxiing on the runway to the apron or storage place, communication with the Air Traffic Control and the shutdown of every system.

6.2.8 Mission phase 8.0, refurnishing

The final phase of the mission is setting the aircraft back to its initial condition before phase one. For example clients could want some of their inserted equipment back, or it is faster for the next set-up to be done. An other option is that phase eight is the same as phase one if two clients are planned after each other.

The FFBD with all sub-phases and activities integrated in the general phase overview as given in Figure 6.2 can be found in Figure 6.7. It gives an overview of the functions the mission consists of which were described in this chapter.

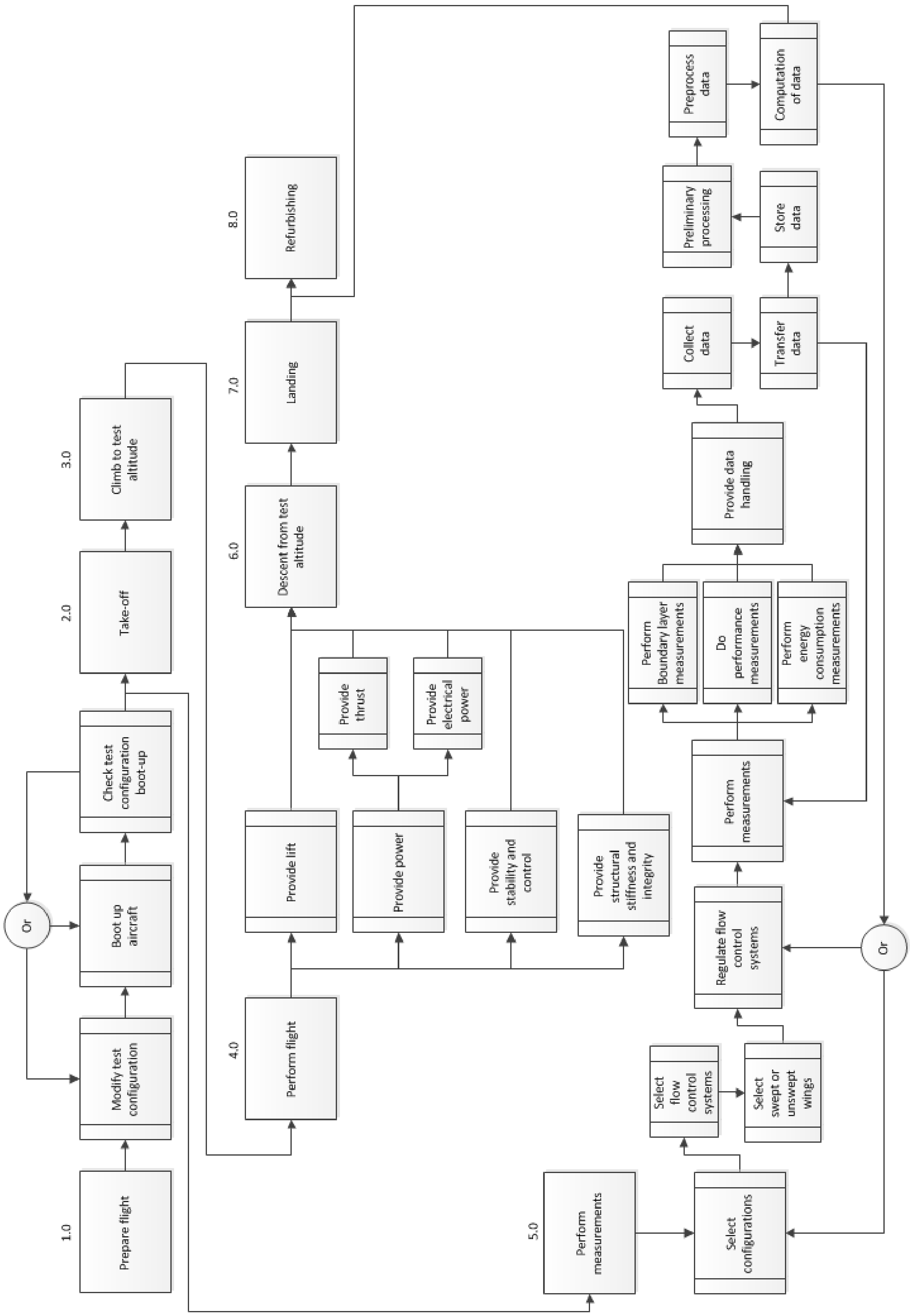


Figure 6.7: Complete Functional Flow Block Diagram (FFBD) with all sub-phases integrated

7 Requirement analysis

This chapter will elaborate upon the constraints placed on the design. Following from the market analysis (Chapter 4), sustainable development strategy (Chapter 5) and functional analysis (Chapter 6), a set of constraints for the overall system has been obtained. Together with the requisites of the stakeholders, these will lead to the boundaries of the system defined in the requirements presented in this chapter.

7.1 Identifier syntax

The requirements must be usable and allow for quick referencing without having to mention it entirely. This is done through an identifier which is unique for a requirement. One of the extra functions of the identifier is to show the flow-down structure of the setup. By using the identifier, the corresponding higher level requirement and the logic of the design can be determined.

The identifiers for the systems consist of three elements: requirement type, requirement affiliation and requirement number.

- The requirement type consists of three possibilities: Top Level Requirement (TLR), Functional Requirement (FR) and System Requirement (SR). Depending on its origin, the type of this requirement is determined.
- The affiliation has to do with either the top level requirement which has influenced the requirement or to the department (Aircraft Design (AD), Test System (TS) or Aerodynamics (AR)) which it influences.
  - In case there is a top level requirement that leads to a system or functional requirement, a department abbreviation is given plus a number (eg. AR.1). The department abbreviation of the functional/system requirement is then the same as that of the top level requirement. The number is equal to the number of the top level requirement belonging to that department (eg, top level requirement for AR.1 is TLR-AR-01).
  - In case there is no top level requirement that influences the functional/system requirement, the affiliation refers to the system the requirement influences the most. The top level requirements only use this type, since no higher requirements are present.
- The requirement number is the part that makes every identifier unique. If two requirements have the same type and affiliation, the number is the only unique part of the requirement.

For example, the identifier FR-TS.1-02 is a functional requirement which is mainly influenced by the top level requirement with the identifier TLR-TS-01 and is the second requirement with the same type and affiliation. Another example is SR-AD-04. This identifier is a system requirement with no top level requirement, but influences mainly the aircraft design and is the fourth requirement with the same type and affiliation.

* The TBD requirements are beyond the scope of the Design Synthesis Exercise.

7.2 Top level requirements

In the following section, a complete overview of all requirements determined up to the end of phase 2 have been shown.

- **TLR-AD-01:** The aircraft should be based on an existing medium sized aircraft, such as an Embraer 190 or a Boeing 737
- **TLR-AD-02:** The range of the aircraft should be 3000 kilometers
- **TLR-AD-03:** The aircraft should be able to travel at high subsonic and transonic speeds
- **TLR-AD-04:** The aircraft should be able to represent both swept and straight wing configurations
- **TLR-AD-05:** Modifications of the testing equipment should be quick and easy according to the tested technology
- **TLR-AD-06:** The budget for development and modification is 10 million euros
- **TLR-AR-01:** The aircraft should be able to accommodate all major flow control technologies, of which at least suction, synthetic jet, Micro Electro Mechanical Systems (MEMS), piezo and plasma actuators
- **TLR-TS-01:** Real time measurements of flow control performance by means of advanced diagnostic capabilities should be one of the abilities

7.3 Platform design requirements

The testing platform is the main system which shall contain all AFC, testing and data handling systems. To be able to accommodate all these systems and maintain overview, the requirements of the system need to be set up in categories. Four categories were discerned, namely airframe, propulsion, electrical power and stability and control.

7.3.1 Airframe

The airframe is the part which shall accommodate all the systems. One of the main driving elements, originating from the top level requirements, is the accessibility and adaptability to the subsystems which the aircraft has to provide.

- **FR-TS.1-01:** The platform shall be able to facilitate the measurement equipment
- **FR-TS.1-02:** The platform shall be able to accommodate a data handling system
- **FR-TS.1-03:** The platform shall be able to accommodate a data measurement system
- **FR-AR.1-03:** The platform shall be able to accommodate two different AFC systems at the same time
- **SR-AD.5-01:** The platform shall provide access to the AFC systems for maintenance
- **SR-AD.5-02:** The platform shall provide access to the AFC systems for replacement by another AFC systems
- **SR-AR.1-01:** The platform shall provide space to place active flow control actuator systems in any desired positions over the whole wing span
- **SR-AR.1-02:** The platform shall have the ability to be changed to facilitate different AFC systems in two to four weeks
- **SR-TS.1-01:** The platform shall have the ability to be changed to facilitate different measurement equipment two to four weeks
- **SR-TS-07:** The redesign platforms MTOW shall not be more than MTOW of the original aircraft

* The TBD requirements are beyond the scope of the Design Synthesis Exercise.

- **SR-AD.2-05:** The platform system costs shall not exceed 2 million euros
- **SR-AD-10:** The main wing needs to be able to sustain its structural integrity when one module is not present
- **SR-AD-11:** The main wing needs to be able to generate enough lift to sustain flight when one module is not present
- **SR-AD-13:** The test section connection needs to have at least a double redundancy
- **SR-AD-20:** The fuselage bottom should not deteriorate completely when landing without the landing gear down at 1.2 times the stall speed with high lift devices in landing configuration.

7.3.2 Propulsion

Considering the platform design requirements, the fuel system was also deemed to be part of the propulsion system. Only one main requirement has been formulated for this system.

- **SR-AD.2-01:** The aircraft should be able to hold at least 18700 kg fuel

7.3.3 Electrical power

The electrical power system has to be able to provide energy to all the systems in the platform. This includes the AFC which are modular and thus require varying power depending on which type and amount of actuators are installed. This puts a lot of requirements on the electrical system, due to the all the different power requirements of all the different AFC systems.

- **FR-AD-01:** The electrical power system shall provide a 120 % more than net amount of kW required by the subsystems
- **FR-TS.1-04:** The electrical power system shall be able to provide 15 kW for the measurement system
- **FR-TS.1-05:** The electrical power system shall be able to provide 5 kW for all on-board aircraft segments of the data handling system
- **FR-AR.1-02:** The electrical power system shall be able to provide 10 kW for the AFC systems.
- **SR-AD-01:** The electrical power system shall be able to provide To Be Determined (TBD)* kW peak power for TBD* seconds
- **SR-AD-02:** The electrical power system shall provide electrical power with a voltage variable between 0 V and 15 kV for the AFC system
- **SR-AR.1-04:** The electrical power system shall provide variable power to the AFC systems up to 10 kW
- **SR-TS-08:** The electrical power system shall weigh no more than TBD* kg
- **SR-TS-09:** The electrical power system shall perform with a 99.9 % reliability
- **SR-AD.2-06:** The electrical power system costs shall not exceed TBD* euros
- **SR-AD-14:** The electrical power system shall be able to accommodate a waveform transformer
- **SR-AD-16:** The electrical power system shall be able to accommodate a frequency transformer
- **SR-AD-17:** The electrical cable system shall be able to handle at least voltages up to 15 kVAC
- **SR-AD-18:** The electrical cable system shall be able to handle at least frequencies up to 15 kHz
- **SR-AD-19:** The electrical cable system shall be able to handle sine, square, triangular, pulse and sawtooth waveforms
- **SR-AD-21:** The electrical cable system of the AFC shall be able to handle at least 428 VDC

* The TBD requirements are beyond the scope of the Design Synthesis Exercise.

7.3.4 Stability and control

Stability and control of the platform is obviously critical throughout the complete mission. In-flight testing is a key driver in this design and the stability may be influenced by activating the AFC systems. Therefore requirements on the aircraft's stability are a necessity.

- **SR-AD-06:** The aircraft must be initially statically stable without the AFC systems active
- **SR-AD-07:** The aircraft must be initially dynamically stable without AFC systems active
- **SR-AD-08:** The aircraft must be able to manipulate stability in-flight
- **SR-AD-09:** The aircraft must be controllable according to CS-25 regulations

7.4 Active flow control requirements

A large part of the mission is the in-flight testing of active flow control. The requirements mentioned below map the most important constraints the AFC system has to deal with.

- **SR-AR-01:** The AFC system shall weigh no more than TBD* kg
- **SR-AR.1-05:** The platform shall provide 1 bar differential pressure for a pressure driven actuator
- **SR-AR.1-03:** The platform shall provide 200 kg of fuel for combustion type actuators
- **SR-AR-03:** The AFC systems will be placed symmetrical over the X-axis in the stability frame

7.5 Test system requirements

To analyse the effect of the flow control actuators, a test system needs to be incorporated. This test system should be able to both measure the effects and save the obtained data. A subdivision between these two categories is made.

7.5.1 Data handling

During the in-flight tests, all the data should be collected, stored and analysed in a short period of time to be able to have the highest efficiency in testing. Taking that into account, a set of requirements is determined to improve the set-up and make sure that all the data can be analysed extensively.

- **FR-TS-01:** The data handling system must allow for transfer of all obtained test data to the ground station
- **FR-TS-02:** The data handling system must be able to store all obtained test data during one flight mission
- **FR-TS-04:** The data handling shall be able to control the AFC systems in any configuration
- **SR-TS.1-02:** The data handling system must allow for real time processing of the flow control performance measurements
- **SR-TS-01:** The data handling system shall weigh no more than a TBD* kg
- **SR-TS-02:** The data handling system shall perform with a 99.9 % reliability
- **SR-AD.2-03:** The data handling system shall cost no more than a TBD* euros
- **SR-TS-10:** The data handling system shall have a back-up storing system available during flight
- **SR-TS-11:** The data handling system shall have a back-up processing unit available during flight
- **SR-TS-12:** The data handling system shall be able to support data coming from redundant measurement systems

* The TBD requirements are beyond the scope of the Design Synthesis Exercise.

7.5.2 Measurement equipment

The measurement system is required to capture all the required information to analyse the effect of the flow control systems. With that in mind, a set of requirements is created to be able to capture these effects with the best quality available.

- **FR-TS-05:** The measurement system must be able to measure the effect of active flow control actuators on the performance of the aircraft with a TBD* accuracy (%)
- **FR-TS-06:** The measurement system must be able to measure the effect of active flow control actuators on the flow over the whole wing span with a TBD* accuracy (%)
- **FR-TS-07:** The measurement system must be able to measure the feedback performance of the active flow control actuators with a TBD* accuracy (%)
- **FR-TS-08:** The measurement system must be able to measure the magnitude of control surface actuator forces in N with a TBD* accuracy (%)
- **SR-TS-04:** The measurement system must be able to measure 'standard' flight parameters (by standard, as defined in Table 2.1 of [4] is meant) with a TBD* accuracy (%)
- **SR-TS-05:** The measurement system shall weigh no more than TBD* kg
- **SR-TS-06:** The measurement system shall perform with a 99.9 % reliability
- **SR-AD.2-04:** The measurement system costs shall not exceed TBD* euro

* The TBD requirements are beyond the scope of the Design Synthesis Exercise.

8 Conceptual design

This chapter presents the results of the conceptual design. The information found through literature studies and the analyses presented earlier in this part lead to the generation of possible concepts. The scope of this chapter will be limited to the characteristics and features of the final concept. A reader interested in the trade-off procedure that lead to the final concept, is referred to the midterm review.

In the first section the layout of the overall design is presented. It includes drawings that show the main features of the design. The second section provides a discussion on which AFC technologies can be integrated and tested. The final sections holds a discussion of the measurement systems to be implemented.

8.1 Overall concept

This section describes the chosen base aircraft. It will present the original state of the aircraft, what modifications the aircraft will undergo, and the general layout of the aircraft.

8.1.1 Base aircraft

The chosen base aircraft is a 1991 Boeing 737-500, previously owned by Macedonian Airlines. The aircraft costs US$ 1.746.000,00, and underwent 22636 total airframe cycles and 28360 total airframe hours since new [5]. The maximum number of allowed flight cycles, defined by the FAA, is 60000 for a B737. The Boeing 737-500 is a twin-engine, medium-range passenger aircraft. It has a conventional planform in the sense that both its engines are located in pods under the wing and that the empennage has no exotic configuration, Figure 8.1. Table 8.1 presents some parameters describing the B737-500.

Table 8.1: B737-500 characteristics

Parameter	Unit	Value	Parameter	Unit	Value
Thrust	[kN]	90	M_{cruise}	[-]	0.82
Range	[km]	4444	Wing surface	[m^2]	105.40
MTOW	[kg]	60550	Service ceiling	[m]	11278
OEW	[kg]	31983	Height	[m]	11.10
Length	[m]	31.00	Width$_{fus}$	[m]	3.76
Span	[m]	28.90	Height$_{fus}$	[m]	4.11
Sweep	[deg]	25			

The B737-500 is part of the B737-classics, and has the same wing and tail as the larger B737-400. The fuselage has the same structure as the B737-400 as well, but is shortened. As a result, the B737-500 is structurally overdesigned. This will leave margins for modifications. Also, this aircraft is a nice representation of commonly used aircraft in the commercial aircraft industry. This would make test results more valuable, since the platform will be able to prove that the actuators work on a frequently used aircraft.

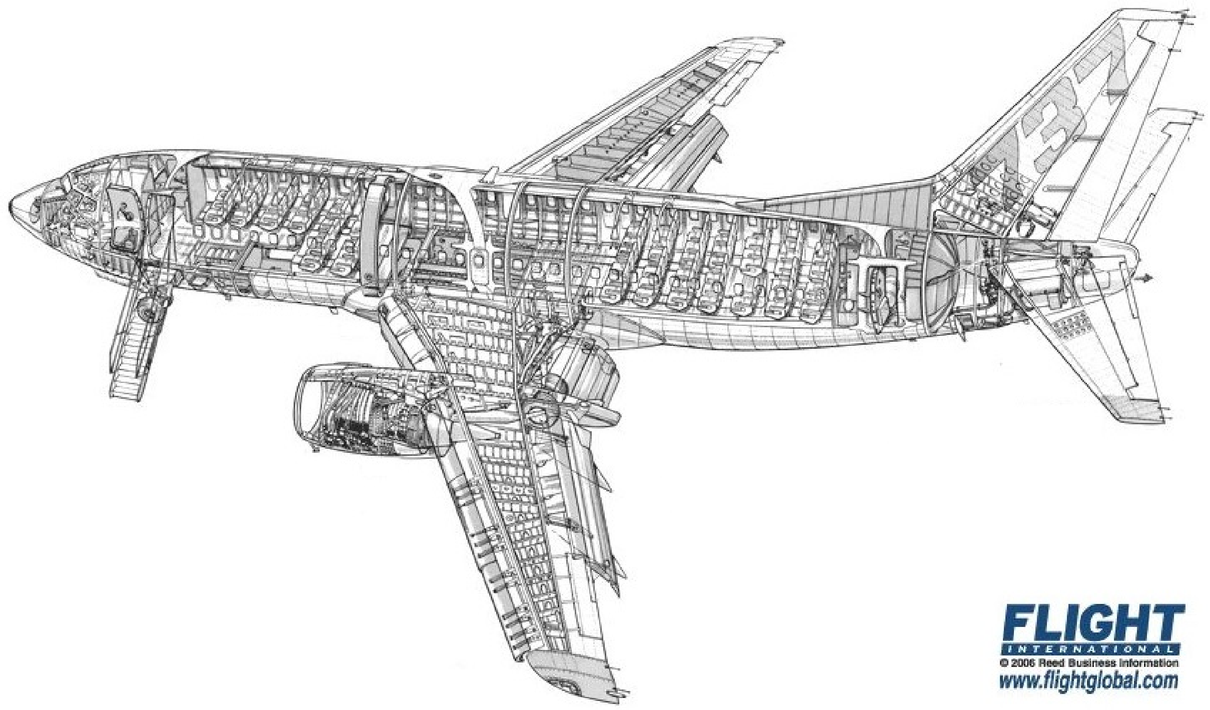


Figure 8.1: Open view of the B737-500 showing the configuration and layout of the original model

8.1.2 Modifications

The base aircraft needs to be modified, before it can act as a test platform for AFC systems. These modifications have been categorised in three major parts:

- **Wing Redesign:**. The wing needs to be redesigned such that it is able to accommodate ACF actuators, and measurement equipment. To create space for these systems, the fuel tank will be allocated to the fuselage. Structural reinforcements will be made where necessary.
- **Pylon Design:**. A pylon needs to be designed that is able to hold AFC actuators, and measurement equipment. The pylon should designed such that stability & control of the whole aircraft remains guaranteed. The mounting for the pylon should be designed, and the structure of the fuselage should be reinforced such that it is able to carry the pylon.
- **Interior Redesign:**. The interior should be stripped to accommodate the supply systems for the AFC actuators, and the measurement equipment. The data handling, and test engineer booth should be placed inside the fuselage as well. Finally an additional fuel tank needs to be designed.

These modifications will influence the performance of the original aircraft. The effects on the aerodynamic properties, C_L, C_D, should be monitored. It should also be analysed how the modifications influence the weight and the stability of the aircraft.

8.1.3 General layout

The configuration of the final concept will be presented in this section. First, an overview of the exterior of the aircraft will be given. Secondly, a lay-out of the fuselage interior will be presented.

Figure 8.2 shows the exterior of the design. The test sections that will be able to integrate AFC actuators, and measurement equipment are marked. The parts that not have been marked will mainly remain untouched. More details on the AFC actuators, and measurement equipment that can be integrated will be discussed in the upcoming sections. The pylon will be placed such that impact on the stability & controllability will be minimal.

The existing interior of the aircraft will be stripped and redesigned to accommodate all the testing support systems. The cockpit will largely remain the same. The cabin environmental control system of

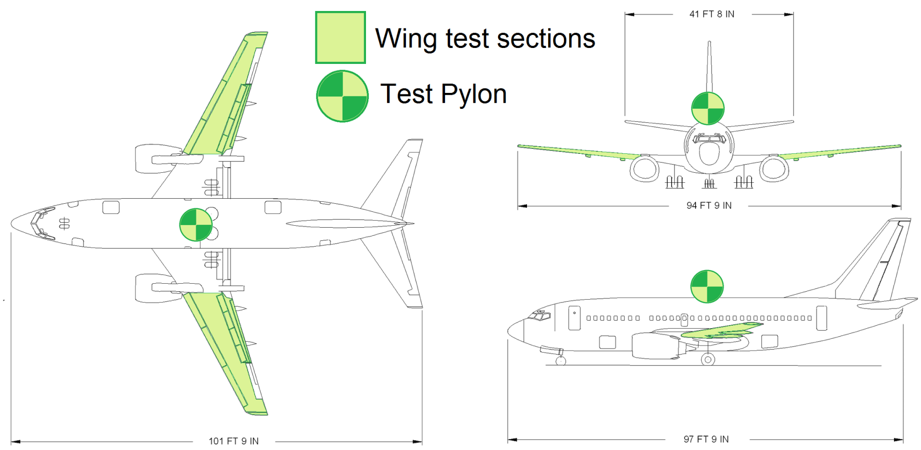


Figure 8.2: Three view of the B737-500 with the test sections which will accommodate the AFC and test systems, marked in colour. The test pylon can be seen on the top of the fuselage.

the aircraft will be left intact and unchanged. The interior will be divided in different parts as can be seen in Figure 8.3, and Figure 8.4. A distinction between the following parts will be made:

- **Engineer booth:**. The engineer stations will be placed in the most forward part of the fuselage. It is aimed to accommodate approximate six test engineers. The engineers will have access to computers on which preprocessing of the obtained data will be done. They will be able to monitor the testing real time, and make testing adjustments if necessary. The engineers will also control the AFC actuators, and the measurement equipment.
- **Data handling:**. The data handling and storage system will consist of the processors and data storage devices. They will be able to store all the obtained data.
- **Reinforcement allocation:**. The fuselage structure underneath the pylon will be reinforced such that it is able to carry the loads and moments that will be induced by the pylon. This sections will also be used to accommodate PIV cameras that will measure PIV laser sheets on the pylon or one of the wings, Figure 8.4. The PIV laser will be placed in the fuselage. Laser beams will be transported to the test sections to generate a laser sheet at that location.
- **Power allocation:**. Power plants will be placed in the most rearward section of the fuselage. These power plants will consist of electricity driven pumps to generate differential pressure.
- **Additional fuel tanks:**. The fuel tank from the wingbox will be partially removed. The shortage of fuel will be stored in additional tanks placed at the luggage storing compartments, Figure 8.4.

Redesigning of interior will also focus on all the cables required to connect the subsystems. Fuel lines will run from the fuselage fuel tanks to the engines. Cables for power, electronics (control and data), and differential pressure will run between the fuselage and the wings as well. Similar cables will run to the test pylon.

8.2 AFC systems

Two locations are available for AFC systems: the main wing and the pylon wing. The pylon wing will be able to hold any AFC actuator available, as it can be shaped specifically to fit an actuator. It will be mainly used for AFC actuators inappropriate for installation on the main wing (e.g. morphing wing). On the main wing, a certain span is available for the installation of the AFC systems. Currently only the root is excluded, due to the wing-fuselage interaction which causes inverted flows.

The wing tip will be made modular, such that it can be taken off and replaced in a convenient way. This way, different vortex-counteracting actuators and wing tip configurations can be tested.It was

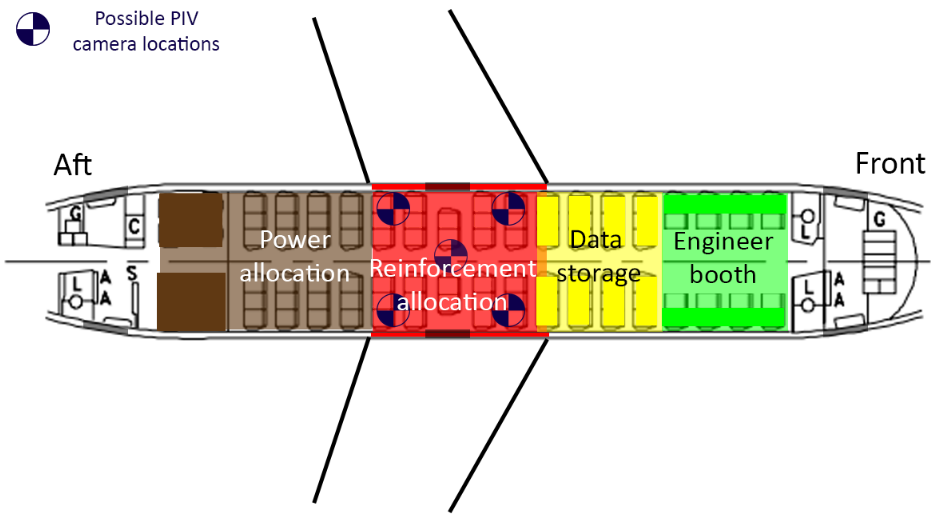


Figure 8.3: Conceptual redesign interior of the B737-500.

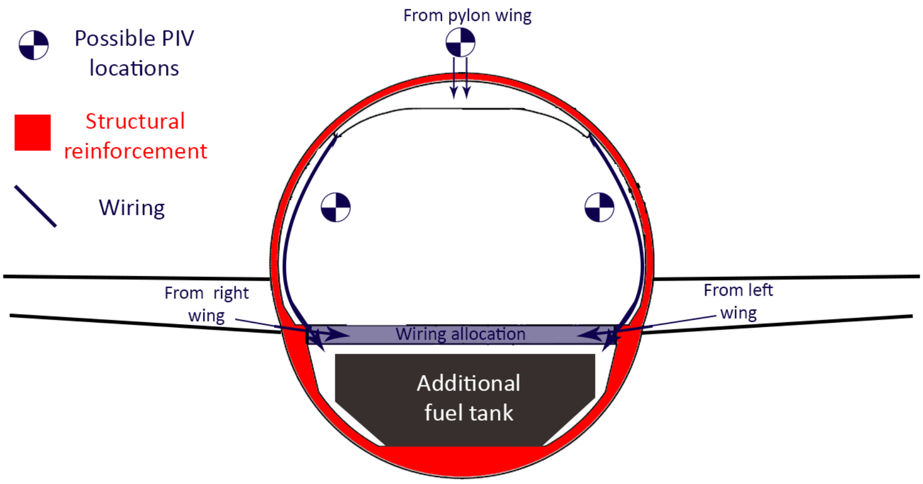


Figure 8.4: Cross sectional view of the conceptual fuselage interior seen from the front with possible locations of PIV cameras and the location of the additional fuel tank.

chosen to completely replace the wing tips, since this part of the wing is relatively small in both size and structural impact compared to the rest of the wing. It is simpler to replace the complete tip instead of creating a modular panel system inside. Modification of the main wing will happen on both wings to assure symmetry of the wings and aircraft.

All AFC actuators have been categorised with respect to aerodynamic effect and impact on structure. An actuator can either have transition or separation delaying effects and have a low or high impact on the structure. Transition delaying actuators are most effective near the leading edge of a wing and thus their modules will be located right behind the slats. Separation delaying actuators are most effective when located over a large part of the chord. Hence, their modules will be located between the former modules and the high lift devices or control surfaces. Low impact commonly means it can be integrated in a panel. High on the other hand means whole parts of the wing have to be replaced in order to accommodate an actuator. High impact actuators will therefore be located on the pylon wing. Besides the four mentioned categories, some actuators are also interesting for vortex-counteracting. These actuators can be integrated in a customised wing tip. The following actuators will be compatible with the AFC system on the main wing:

- Located near leading edge (transition delay)
  - Vibrating ribbon devices
  - Piezoelectric vibrating parts
  - Pulsed jet
  - Power resonance tube
  - Zero net mass flux
    - Piezoelectric
    - Electrodynamic
  - Fluidic oscillator.
- Located downwind of leading edge (separation delay)
  - Active dimples
  - Suction actuators
  - Single dielectric barrier discharger
- Located at the wing tip (vortex-counteracting)
  - Pulsed jet
  - Single dielectric barrier discharger
  - Suction and blowing actuators
  - Rotating surfaces.

All of the above actuators can be located in the skin of the wing, since these actuators will have a low impact on the structure. This makes modularity easy to implement. Also, by using this classification of actuators, testing of combinations of actuators is possible. By placing a transition and a separation delaying actuator in series, their combined effect can be measured. The remaining actuators, will have to be located on the pylon wing, if these are to be tested. These include:

- Rotating surfaces
- Plasma heating
- Morphing structure

Note that plasma heating has been assigned a location on the pylon wing not for structural reasons, but for electrical reasons. This actuator might disturb other electrical systems due to the high voltages it works with. No reference reports have been found which confirm this suspicion and global estimations result in a high unlikelihood of interference with other electrical systems. However this decision has been made to decrease any risk as much as possible. The other two AFC systems are either not compatible or demand too much space to be placed on the main wing.

The main philosophy for the design of the AFC systems is a convenient 'plug and play' system. Power and pressure plugs will be installed in multiple modules located in the main wing where actuators can be removed and installed easily. This will be done by use of a panel system. This panel system will incorporate the skin of the wing which will be split up in a certain amount of panels which are easily removable. In and under each panel one or more actuators can be installed. This way the wing can be modified easily without reducing structural performance. Also, this makes the wing compatible with future AFC systems which can be integrated in a panel. If not, they can be integrated in the pylon wing. Overall, the concept strives to be able to accommodate all possible types of actuators and, in result, act as a flexible test platform as much as possible.

8.3 Measurement systems

To register the performance of the AFC systems, measurements will be performed on three levels. The highest level concerns measuring the performance of the aircraft as whole. This level is used to measure the effects of the AFC actuators on the performance of the aircraft. The second level concerns measuring of the boundary layers and the local flow around the AFC actuator. It is used to analyse how the AFC actuators influence the flow field. The third and final measurement level, is the measurement level that is concerned with the power consumption of the AFC actuators. It is required to gain knowledge of the usability of the specific actuator.

8.3.1 Aircraft performance level

First, the aircraft level measurement equipment will be discussed. It will include measurements of the amount of fuel that is being used up by the aircraft's propulsive system. The fuel consumption is used to determine the total weight and thrust of the aircraft during flight. An on-ground weight & balance scale will be used to determine the initial weight of the aircraft. Furthermore, measurement equipment is required that is able to determine the flight velocity, attitude, and acceleration of the aircraft. These are present in the base aircraft. However, some systems may require an upgrade to improve measurement accuracy. The following instruments will be required to measure the previous mentioned parameters:

- Pitot-static ports
- Pitot tubes
- GPS receivers on the nose, tail and both wing tips
- Accelerometers and gyroscopes
- Flow-vanes
- On-ground weight & balance scale

The pitot systems are necessary when determining the outside air pressure, temperature and air speed. The GPS receivers can be used to have additional determination of attitude, (ground) speed and altitude. The flow-vanes are required to determine an accurate angle of attack, and angle of sideslip.

8.3.2 Boundary layer level

On an aerodynamic level, a PIV system will provide detailed information about the flow field around both the main wing and the test section. This system can be calibrated for different test sections. This will enable measurement of the velocity field at different positions. The implementation of this system requires a so called carrier fluid containing particles. It has been decided that environmental particles, water drops stored in clouds, will be used. The implementation of smoke generators to generate particles, proved to be too complex. The cloud particles will be used as the carrier fluid. The main downside is, that accuracy of PIV measurements will be depend on the weather and the type of clouds available.

There will also be a force-moment balance present to measure the forces and moments acting on the pylon test section. The force-moment balance will be integrated into the suspension of the pylon. In addition to this the accelerometers in the main aircraft fuselage should give the input towards the system that processes the data from the pylon such that disturbances can be filtered out.

Finally, there will be the possibility to integrate pressure, and temperature taps in panels on the wing and in the pylon. This will give the opportunity to create a specific measurement configuration for a specific AFC actuator. The can be connected in a similar fashion as the AFC actuators via a 'plug and play'- environment. These taps will be required to determine the flow characteristics in greater detail. It will also be able to verify the data of the PIV measurements.

8.3.3 Energy consumption level

Finally there will be systems measuring the power, and pressure usage by the AFC systems. These will consist of simple power meters attached to the available power and pressure sockets. These will be present in both the pylon and the main wing.

9 Sensitivity analysis

The purpose of this chapter is to analyse the sensitivity between the interfaces of the aircraft. This is needed to facilitate the preliminary design, which will be explained in the next part. The first step is to identify the different interfaces of the aircraft, including their functions and their interrelations. This provides a general overview of the system and allows the designer to check whether the entire system fulfils the required purpose. This will be done in Section 9.1. This chapter will be continued with a sensitivity analysis on the interrelations between all subsystems, quantifying the amount of complexity involved per interface. This allows the designer to see how much the design of a subsystem affects another subsystem. The results of this analysis will be presented in Section 9.2.

9.1 Interface definition

The N^2-chart is used in systems engineering to identify the interfaces and their functional interrelations. In this section, the interfaces of the aircraft's conceptual design will be identified and briefly elaborated upon. The function descriptions are based on descriptions from [6] and completed with additional functions followed from the concept design. The N^2-chart is shown in Figure 9.1.

Wing group: The primary function of the wing is to provide lift. In addition the wing provides space for fuel storage, transfers loads to the fuselage, provides space for subsystems and measurement systems and provides aerodynamic control.

Fuselage group: The main function of the fuselage group is to provide space for subsystems and payload. A large contribution of the payload comes from pylon wing on top of the fuselage. The pylon wing functions as a test section for AFC systems. In addition, the fuselage is designed to maintain pressure and provide space for the crew.

Empennage group: The key function of the empennage group is to provide aerodynamic control. This includes both aerodynamic stability, provided by the horizontal and vertical stabiliser, as well as control, provided by the control surfaces (e.g. elevator, rudder). Additional attention should be paid to the empennage group due to the wake caused by the test section.

AFC systems: The functions of AFC systems are to increase flight efficiency in terms of lift enhancement, drag reduction and noise control. However, in this particular design the AFC systems will be investigated and tested for their application outside laboratory conditions. A supporting system should be able to provide measurement data.

Fuel system: The fuel system provides fuel to the engines, the electrical system and the Auxiliary Power Unit (APU). In addition, the fuel system comprises fuel storage, fire control, active stability in terms of fuel flow control and fuel flow measurement.

Electrical system: The electrical system's function is to provide AC and DC power, required by the sub-systems. Necessary adjunct functions include providing back-up power and distributing the electrical load. During flight the electrical power is generated by the engines and stored in batteries if necessary. Due to the AFC systems this particular design requires more electrical power compared to other civil aircraft. Systems demanding a substantial amount of electrical power are the engine starting, avionics, AFC systems, including measurement systems, data handling system and the Power Control Unit (PCU).

Propulsion system: The propulsion system includes the power plant, thrust management and propulsion monitoring. The nacelles provide support and space for subsystems of the engine. The power plant's primary function is to provide thrust. In addition it provides bleed air to the pneumatic system, certain AFC systems and the environmental control system. The function of the thrust management subsystem is to regulate the required forces for each phase of flight and landing roll-out.

Mechanics: The mechanical group consists of the landing gear and the hydraulic systems used to power the control surfaces and the high lift devices. The main function of the landing gear is to withstand landing loads, and enable the aircraft to move on ground. The hydraulic systems need to have sufficient power to deflect the control surfaces to keep the aircraft controllable. The movability of the test section is also provided for by this system.

Environmental control: The primary function of the environmental control system is to regulate the temperature, pressure and oxygen levels in the aircraft. Another function of the environmental control system is to provide ice and rain protection. This consists of an anti-icing, de-icing, and the rain protection subsystem.

Avionics and fly-by-wire system: The Fly-By Wire (FBW) system translates the pilot inputs in the cockpit into control surface deflections. The avionics system provides the aircraft with an automatic pilot and navigation. The avionics, including the FBW system, have a close collaboration with the data handling system and the communication system.

Test measurement system: The test measurement system will provide measurement data on three levels. The first level of measurement comprises the performance of the complete aircraft. The second level measures the boundary layer effects of the AFC, and the third level measures the power consumption of the specific AFC systems.

Data handling system: The primary functions of the data handling system is to process and store data. Data comprises both primary flight data(e.g. velocity, altitude, attitude etc.) and test measurement data. The system will provide the pilots and engineers with critical information.

Communication System: The main function of the communication system is to provide communication between the aircraft, ground station and Air Traffic Control (ATC). The communication system also receives signals for the GPS system, and manages internal communication, such as indication of errors or audio communication.

Interior: The interior provides accommodations for the crew to perform the test flight, as well as life support and evacuation. The interior's main components are the data handling system, processing unit, and test measurement equipment. In addition the interior includes the water and waste management, lavatories and plumbing.

9.2 Sensitivity analysis

The previous section defined the subsystems for the aircraft, as well as the existing interfaces between them. This section will continue with a sensitivity analysis. This is a process in which each interface is examined and graded from one to five in terms of complexity. The criteria for the grading are as follows:

Grade 1: The interface does not provide any design complexity and does not need to be taken into account actively.

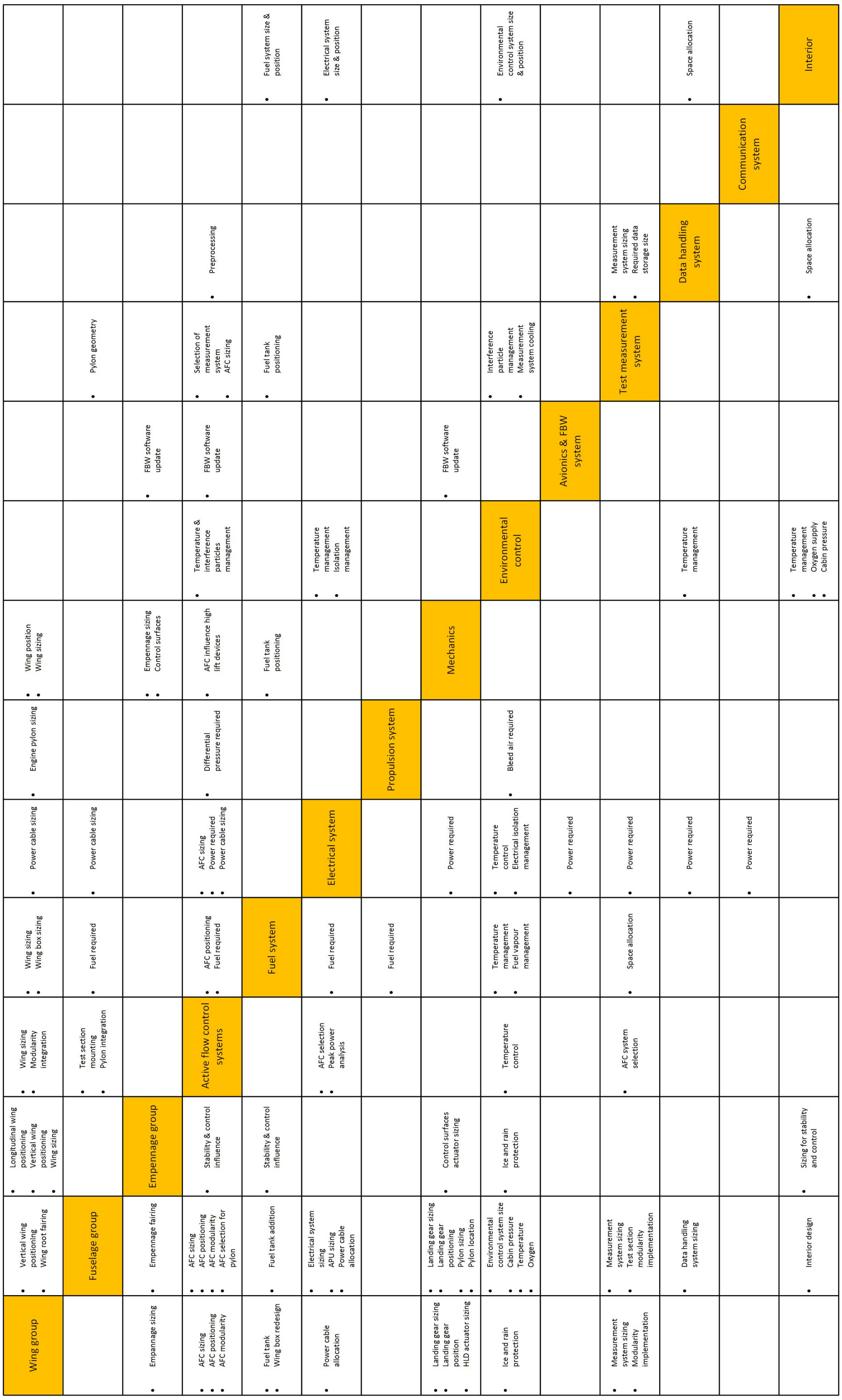


Figure 9.1: Aircraft conceptual design N^2-chart. The main diagonal contains the interfaces. The output of an interface is shown in a horizontal connections with the specific interaction, the input is shown vertical.

Grade 2: The interface provides some design complexity, but this will only be monitored passively.

Grade 3: The interface provides a level of complexity which needs to be monitored actively, but significant design issues are not expected.

Grade 4: The interface provides such a complexity that active monitoring will not prevent the introduction of design issues.

Grade 5: The interface is so complex that constant active monitoring is required. Significant design issues are expected to arise.

In total 182 interfaces exist. In order to keep the length of this discussion manageable, only the interfaces which require active monitoring (that is, have a grade of 3 or higher) will be discussed. The other interfaces will not be discussed explicitly. This section will follow the main diagonal of the N2 chart presented in Figure 9.1. The format in which this section should be read, is as follows. The paragraph title is the subsystem to which changes are made. The subsystems within each paragraph show what the effect on them is by changing the subsystem.

The Table 9.1 provides an overview of the grades given to the interface interrelations.

Wing Group

- **Fuselage group (3):** The introduction of a modular wing skin will influence the weight of the wing. This means that the wing root fairing needs an update to cope with the change in loads. Initially it will be tried to not alter the wing position with respect to the fuselage, thereby keeping complexity limited. Another consideration to be taken into account is the sizing and location of cabling to both the AFC actuators and their accompanying measurement systems.
- **AFC systems (4):** The AFC systems need to be integrated into the wing. Therefore the design of the wing will have major influence on both the selection as well as the implementation of these systems.
- **Fuel system (3):** The introduction of the AFC systems in the wing will cause some of the fuel tanks to be moved to the fuselage. Even though the basic planform of the wing (e.g. sweep, wing surface area, airfoil) will remain the same, the internal wing structure including the wing skin will need to be redesigned. During this redesign, it should also be investigated how the bending relief of the wing is altered when replacing fuel weight by AFC systems weight.

Fuselage group

- **AFC systems (4):** The fuselage group will accommodate the test section for straight wings, which in turn houses some AFC systems. The redesign of the fuselage puts limitations on the exact structural implementation of the pylon. The main expected problem is that the fuselage should be reinforced to be able of carrying the loads introduced by both the pylon and the test section.
- **Fuel system (3):** The presence of the pylon and the test section will introduce additional drag, making the aircraft less efficient. It should be investigated if this drag increase is such that extra fuel should be taken along. This extra fuel will require extra storage room. Also the test section might be fitted with fuel driven actuators which again increase the amount of required fuel.
- **Test measurement system (3):** The design of the pylon also needs to account for the fact that measurement systems should be implemented on the pylon. This will introduce some structural complexity.
- **Interior (3):** It is expected that the fuselage does not need to be redesigned to fit the interior, however the following modification are required. The existing passenger seats will be taken out and replaced by other systems (e.g data handling systems). Since these systems are most likely placed in the front of the fuselage, it should be monitored if the weight of these systems does not effect the stability too much.

Table 9.1: Grading sheet for sensitivity analysis. The interactions between interfaces are graded according according their impact. The left column lists the interfaces that have an impact, the top row shows the interfaces that are affected. For example, the fuselage group impacts the wing group with 2, while the wing group impacts the fuselage group with 3.

	Wing group	**Fuselage group**	**Empennage group**	**AFC system**	**Fuel system**	**Electrical system**	**Propulsion**	**Mechanics**	**Environmental control**	**Avionics**	**Test measurement system**	**Data handling**	**Communication**	**Interior**
Wing group	0	3	2	4	3	2	1	2	1	1	2	1	1	2
Fuselage group	2	0	1	4	3	2	1	2	1	1	3	1	1	3
Empennage group	2	2	0	1	2	1	1	2	1	1	1	1	1	1
AFC system	5	5	3	0	4	4	2	2	3	3	4	2	1	1
Fuel system	3	3	1	3	0	1	1	2	2	1	2	2	1	1
Electrical system	3	3	1	4	3	0	1	1	1	2	2	1	1	3
Propulsion	1	1	1	1	1	1	0	1	1	1	1	1	1	1
Mechanics	2	1	2	1	1	2	1	0	1	2	1	1	1	1
Environmental control	3	3	2	1	1	1	1	1	0	1	3	1	1	3
Avionics	1	1	1	1	1	1	1	1	1	0	1	1	1	1
Test measurement system	5	5	1	4	3	3	1	1	4	1	0	2	1	3
Data handling	1	2	1	1	1	3	1	1	2	1	2	0	1	3
Communication	1	1	1	1	1	2	1	1	1	2	1	1	0	1
Interior	3	2	1	1	2	2	1	1	2	1	3	3	1	0

AFC systems

- **Wing group (5):** The implementation of modularity will introduce structural complexity. For this design it has been chosen to use removable skin panels to accommodate the flow control systems. The exact implementation of this system will require a complete redesign of both the wingbox and its surrounding structure.
- **Fuselage group (5):** The presence of a modular test section on the fuselage will introduce some structural complexity. In addition, the test section will be equipped with AFC and measurement systems. These systems have to be integrated into a modular pylon and test section.
- **Empennage group (3):** The empennage group will not be actively redesigned and will only change due to modification in other interfaces. It should however be actively monitored, because the empennage plays a fundamental role in both the stability and controllability of the aircraft.
- **Fuel system (4):** This interfaces has two main points of attention. First the selection of certain AFC systems might introduce the need to remove fuel from the wing. Second, some of the AFC systems are fuel driven. This requires extra fuel, which might result in larger fuel tanks.
- **Electrical system (4):** AFC systems such as plasma actuators need a substantial amount of power. This power needs to be transported from the power generator to the actuator. This requires extra attention for the power cable sizing .
- **Environmental control (3):** Some of the AFC equipment might only work in certain weather or temperature conditions. Monitoring these conditions during the test will present additional complexity for the environmental control system.
- **Avionics and fly by wire system (3):** The use of AFC systems could alter the flow over the main wing in such a way that the control surfaces work less efficiently. This means that the avionics require a software update to cope with such changes in the flow over the wing.
- **Test measurement system (4):** The selection of certain AFC systems can require or exclude the use of measurement systems. Because the AFC systems are fitted in a modular fashion, the test measurement systems have to be adaptable as well.

Fuel system

- **Wing group (3):** The required fuel tanks influence the sizing of the wing. A possible reduction in fuel storage in the wing could have as a result that the wing box needs to be redesigned
- **Fuselage group (3):** When fuel is moved from inside the wing to a different location, additional fuel tanks will be placed inside the fuselage. The fuselage needs to accommodate this structurally. Additionally, extra fuel might be required due to the drag induced by the mountings and pylon on top of the fuselage.
- **AFC systems (3):** The allocation of modular panels in the wing will reduce the amount of available space. As a result, the fuel system will need to be resized. Combustion-driven actuators, in both the pylon-mounted test section and the main wing, needs extra piping from the fuel tank.

Electrical system

- **Wing group (3):** To prove sockets for electricity, used by the AFC and measurement systems, cut-outs have to be made in the wing structure. Additionally, the length of extra cabling will have to be taken into account.
- **Fuselage group (3):** The fuselage needs to be reinforced to be able to carry the weight of the Electrical Power Unit (EPU). The system needs to be placed meticulously in order to minimise the amount of required cabling.
- **AFC systems (4):** The electrical system has a large influence on the selection of the possible AFC systems. Actuators which draw a very large amount of current might not be feasible to be placed on the platform. Also the peak power consumption of the systems can be a limiting factor.

- **Fuel system (3):** The EPU could need supplementary fuel. As such, the fuel system might need modification.
- **Interior (3):** The location of the generator and suitable cooling devices will take up space from the interior and reduces the options to furnish the aircraft.

Environmental control

- **Wing group (3):** Since a redesign of the wing is inevitable, the environmental control system to protect the wing from ice and dust will have to be reconsidered as well. The modular panels should still ensure that the wing is an integral part and that no leakage occurs.
- **Fuselage group (3):** A main concern of the environmental control system is to provide the appropriate conditions in the fuselage. These concern the temperature, cabin pressure and oxygen levels.
- **Test measurement system (3):** Making sure the test measurement system can operate in optimal conditions is also part of the environmental control. Surface tappings need to be kept clean of interfering particles, while other systems need active cooling.
- **Interior (3):** The environmental control system needs to be allocated in the given interior space. Temperature control in the presence of data handling systems and certain measurement systems is important.

Measurement system

- **Wing group (5):** The wing needs needs to accommodate measuring devices. The necessary cabling and piping for the different need to be integrated into the wing and might require alternations of the wing structure.
- **Fuselage group (5):** The complexity of implementing a PIV system on the pylon mounted test section demands large modifications. Also the modularity of systems of the pylon will affect the structure of the fuselage.
- **AFC systems (4):** The choice of measurement system can be a limiting factor for the choice of AFC and vice versa. For example, the resolution of pressure data by wall taps can be limited since interference effects with AFC systems could otherwise occur. These interactions should be observed carefully.
- **Fuel system (3):** Integrating measurement systems into the wing will require space and can lead to modifications of the initial fuel system.
- **Electrical system (3):** The choice of testing system determines the amount of power required. A PIV set-up needs a large amount of power to support the lasers, while pressure taps require very little.
- **Environmental control (4):** Test measurement systems require environmental control. For instance, the conditions influencing the refraction of the laser needs to be closely recorded in order to to correct for it.
- **Interior (3):** The consequence of positioning measurement equipment inside the fuselage is that the furnishing option are more constrained. This is again the case for the PIV system for both the main wing and for the pylon-mounted test section. The placement of these systems takes precedence over other fixtures.

Data handling system

- **Electrical system (3):** The amount of power the data handling system needs is a prerequisite to size for the electrical system.
- **Interior (3):** The data handling system, including the storage units need to be placed in the interior. Locating these heavy systems should in the interior may also affect the stability of the aircraft.

Interior

- **Wing group (3):** Interior configuration corresponds with the centre of gravity location. If it is not possible to achieve stability by changing the interior location, the location of the wing might be moved.
- **Test measurement system (3):** some test measurement equipment requires fixed locations with respect to view angles on the main wing/test section. This limits the placing of other interior furnishing.

II

Preliminary design

10

Weight estimations & cost breakdown

This chapter will elaborate on two procedures used to estimate several parts of the aircraft weight, such as the fuel weight and the individual component weights. Doing so, this chapter serves the purpose to provide an initial indication of what the effect of the redesign to be performed is on the weight of the aircraft. This chapter will start with the presentation of the method and results required for a Class I weight estimation. Following this the Class II estimation will focus on the component weigths of the aircraft, both before and after redesign. The amount of weight to be added will have its influence of the cost of the project. Therefore this chapter will conclude with the cost estimation of the project.

10.1 Class I weight estimation

The first step in the process of predicting the weight budget is to perform a Class I weight estimation. This estimation serves the purpose of predicting the fuel weight and doing so the fuel volume. During a regular Class I method the OEW and the MTOW need to be estimated using regression analysis. Since this project redesigns an already existing aircraft, the Class I estimation to be performed here will take the already existing OEW and MTOW as input. In order to perform this estimation, a typical flight mission needs to be defined. This mission is defined by Figure 10.1 [7].

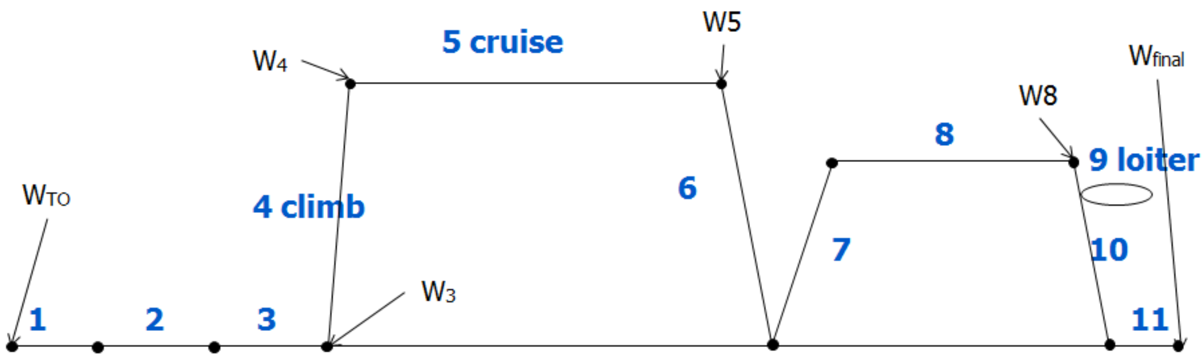


Figure 10.1: Typical flight mission

The following mission phases are distinguished:

- Engine start and warm up (1)
- Taxi (2)
- Take-off (3)
- Climb and accelerate to cruise (4 and 7)
- Cruise (5 and 8)
- Descent (6 and 10)
- Loiter (9)
- Landing, taxi and shutdown (11)

It should be noted that there are two cruise, two climb and two descent phases. This is due to the fact that the aircraft should be able to fly to a different airport than its initial destination, for example due to airport closure.

For this mission eleven fuel fractions can be defined, which when multiplied give the fraction of final weight over MTOW. This equation, as well as the fuel fractions resulting from statistics are given in Equation 10.1 and Table 10.1 [7]:

$$\frac{W_{final}}{W_{TO}} = \frac{W_1}{W_{TO}} \cdot \frac{W_2}{W_1} \cdot \frac{W_3}{W_2} \cdot \frac{W_4}{W_3} \cdot \frac{W_5}{W_4} \cdot \frac{W_6}{W_5} \cdot \frac{W_7}{W_6} \cdot \frac{W_8}{W_7} \cdot \frac{W_9}{W_8} \cdot \frac{W_{10}}{W_9} \cdot \frac{W_{final}}{W_{10}} \tag{10.1}$$

Table 10.1: Fuel fractions from statistics

$\frac{W_1}{W_{TO}}$	$\frac{W_2}{W_1}$	$\frac{W_3}{W_2}$	$\frac{W_4}{W_3}$	$\frac{W_6}{W_5}$	$\frac{W_7}{W_6}$	$\frac{W_{10}}{W_9}$	$\frac{W_{final}}{W_{10}}$
0.990	0.990	0.995	0.980	0.990	0.980	0.990	0.992

The remaining three fuel fractions can be calculated analytically by means of the Breguet equations for range and endurance. Rewritten to obtain the fuel fractions as output, these equations result in Equation 10.2 and Equation 10.3:

$$\frac{W_5}{W_4} = exp\left(-\left(R \cdot \left(\frac{g \cdot c_j}{V}\right)_{cruise} \cdot \left(\frac{L}{D}\right)^{-1}\right)_{cruise}\right) \tag{10.2}$$

$$\frac{W_9}{W_8} = exp\left(-\left(E \cdot g \cdot c_j\right)_{loiter} \cdot \left(\frac{L}{D}\right)^{-1}_{loiter}\right) \tag{10.3}$$

These equations require a number of inputs, such as the cruise speed, range and specific fuel consumption. The input values used are given in Table 10.2.

Table 10.2: Input values for Breguet equations

Component	**Unit**	**Value**
Range R	[km]	3000 (main cruise), 100 (divert)
Gravity acceleration g	[m/s^2]	9.81
Specific fuel consumption c_j	[$\frac{\text{lb}}{\text{lbf}} \cdot$ hr]	0.667 (cruise), 0.330 (loiter)
Cruise speed V_{cruise}	[m/s]	250
Aerodynamic efficiency $\frac{L}{D}$	[-]	15 (cruise), 16 (loiter)
Loiter time E	[s]	1800
MTOW	[kg]	56472

The values regarding the specific fuel consumption and the aerodynamic efficiencies are taken from literature [8]. Using the above input values Equation 10.2, Equation 10.3 and Equation 10.1 can be solved. The fuel weight can then be estimated by means of Equation 10.4:

$$W_{fuel} = W_{TO} \cdot (1 - \frac{W_{final}}{W_{TO}}) \tag{10.4}$$

Solving Equation 10.4 the fuel weight is estimated to be equal to 18700 kg. When using the assumption that the density of Jet-A fuel is 810 kg/m^3, the required fuel volume is equal to 23.1 m^3.

10.2 Class II weight estimation

With the Class I estimation completed the next step in the weight estimation process was to perform a Class II weight estimation. This method goes a level lower than the Class I method, determining the component weights of the aircraft. The Raymer method was chosen as the method to be used. To implement this method the aircraft was broken down into a number of parts, being the wing, the fuel system, the horizontal tail, the vertical tail, the fuselage, the main landing gear, the nose landing gear, the installed engines and furnishing and equipment. The weights as depicted in Table 10.3 were derived, using the input data as can be found in Table 10.4. These lead to an Operative Empty Weight (OEW) of 31811 kg. By using moment equillibrium around the nose of the aircraft it was also determined that the most forward and aft locations of the center of gravity of the aircraft are 12.71m and 13.53 m from the tip of the nose of the aircraft.

Table 10.3: Component weights of the original aircraft, in kg.

Component	**Weight [kg]**	**Component**	**Weight [kg]**
Wing	5622	Horizontal Tail	1175
Fuselage	4882	Vertical Tail	353.0
Main Landing Gear	1212	Fuel system	653.0
Installed Engines	6480	Furnishing and Equipment	11070

Table 10.4: Aircraft Component Weight Input Data

Component	**Unit**	**Value**	**Component**	**Unit**	**Value**
Wing span	[m]	28.88	Vertical tail thickness to chord ratio	[-]	0.611
Wing area	[m^2]	105.4	Fuselage length	[m]	29.75
Wing sweep	[deg]	25.00	Fuselage width	[m]	3.76
Aspect ratio	[-]	9.16	Fuselage heigth	[m]	4.01
Taper ratio	[-]	0.24	Fuselage wetted area	[m^2]	228.73
Thickness to chord ratio	[-]	0.15	Design gross weight	[kg]	64637
Wing fuel weight	[kg]	6480	Landing gross weight	[kg]	56254
Horizontal tail spanwidth	[m]	12.7	Ultimate load factor	[-]	3.75
Horizontal tail area	[m^2]	31.40	Lift over drag ratio	[-]	16
Horizontal tail sweep angle	[deg]	30.00	Mach number	[-]	0.75
Horizontal tail taper ratio	[-]	0.524	Total fuel tank volume	[l]	21578
Vertical tail area	[m^2]	23.13	Cabin pressure	[Pa]	$0.45 \cdot 10^5$
Vertical tail sweep angle	[deg]	35	Pressure volume	[m^3]	300
Vertical tail taper ratio	[-]	0.31	Number of people on board	[-]	10

Next, the component weights of the redesign were estimated. The redesigning process consists of several components. Amongst others, the amount of people on board was decreased to 10 pax. This 10 pax also accounts for the crew. The redesign of the wing has been explained in Chapter 11. Concluding from this chapter the weight increase for the wing is estimated to be 281 kg. After analysis it became clear that the fuselage had to be reinforced as well, increasing its weight by 490 kg. This lead to the aircraft component weights in Table 10.5 and an OEW of 30284 kg.

Table 10.5: Aircraft component weights of the redesigned aircraft, in kg

Component	**Weight [kg]**	**Component**	**Weight [kg]**
Wing	5903	Horizontal Tail	1175
Fuselage	5163	Vertical Tail	353.0
Main Landing Gear	1212	Fuel system	653.0
Installed Engines	6480	Furnishing and Equipment	8396

This means that in total 2840 kg will be removed from the fuselage, much of which originates from the redesign of the interior. To reinforce the fuselage, 490 kg extra material would be needed and 281 kg to reinforce the wing. Finally, the entire aircraft will become 668 kg heavier by placement of the pylon on top the fuselage, bringing the new OEW to 30854 kg.

10.3 Financial budget

Apart from the technical budgets, also a financial budget has to be made for the entire project. Naturally the costs are constrained. The allocated funds for the project are, as defined in requirement TLR-AD-06, €10 Million. The main expenses for the program will be the reinforcements and modifications of the existing airframe. Additionally there will be a significant proportion of the budget reserved for the detailed design, still to be performed later in the design process. It has been decided to use the method for cost estimation as described in [9].

For the method described by Torenbeek, it is necessary to know the weight of the to be designed and produced parts. Then, using an empirically determined formula, the engineering and tooling Man Hour (MHR) can be determined. From this, the cost can be determined by multiplying it with the engineering hour rate and the tooling hour rate. The results of this estimation can be found in Table 10.6.

The total program cost found is approximately €17.8 Million. It has to be noted that this does not account for excessively difficult design problems and possible overruns. For this reason, this figure should be approached with extreme cause and is better to be seen as a lower bound. This is reinforced by the fact that it does not fit in the allocated budget of €10 Million by quite a margin.

Table 10.6: Breakdown of program cost

Engineering & Development	EUR/USD = 1.3334	Eng hr rate: USD 103/hr	Tool hr rate: USD 51/hr
	Removing Interior	**Reinforcement**	**Placing**
$MHR_{Engineering}$	16534 hrs	29473 hrs	52638 hrs
$MHR_{Tooling}$	43149 hrs	79674 hrs	142845 hrs
Total Engineering Cost	USD 10.16 Million	=	EUR 7.62 Million
Total Tooling Cost	USD 13.55 Million	=	EUR 10.16 Million
Total Project Cost	USD 23.7 Million	=	EUR 17.8 Million

10.4 Recommendations

Both the weight estimations and the cost estimation can be improved on a number of points. The input data the Class I weight estimation method uses is based on regression analysis for 'classical' civil transport jets. The redesigned parts of this particular aircraft will change its properties in such a way that the input data should carefully be reevaluated. One particular parameter which should be reestimated in the detailed design phase (and maybe already in more iterations of the preliminary design) is the aerodynamic efficiency of the aircraft with its pylon mounted. It is expected that due to the fact that the pylon is mounted upside down the aerodynamic efficiency of the aircraft will decrease, which will result in an increase of the amount of fuel weight required.

The method used for the Class II weight estimation was Raymer's method. This method is already more than 60 years old. The base aircraft used for the formulation of the statistic regression equations are outdated and not really applicable to the Boeing 737-500. Adding the redesigned new geometry of the aircraft as another invalidating factor, more research should be done into if there are more modern Class II methods available based on more modern aircraft. For the weight estimations in general it can be said that more iterations would be required to finetune and harmonise both estimation methods, by for example using experimental data to formulate a calibration factor as follows:

$$W_{estim_2} = W_{calc_2} \cdot \frac{W_{experim}}{W_{calc}}_1 \tag{10.5}$$

In Equation 10.5, each new estimation is calibrated with the ratio of the previous calculated weight estimate and the figure from experimental data. This will improve new estimations [10].

Concerning the cost estimation a number of issues should be adressed. First of all, it has already been mentioned that the budget strategy done does not account for extreme design complexity and overruns, that is, the strategy is very agressive. This means that the figure given should be considered an absolute lower bound, rather than the actual project cost. For next iterations it is a good idea to give a more reliable figure. The method used for the cost estimation comes from Torenbeek. Just as with the Raymer weight estimation method, this method can be considered outdated. Further design iterations should look into if newer parametric cost models are available and usable for this experimental aircraft. When doing so the input data for the method should also be reconsidered, looking if the redesign details such as the mounting of the pylon can be expressed in more concrete figures. Here more iterations should finetune the obtained results to resemble reality more closely.

11
Wing redesign

To enable integration of AFC systems and test equipment, the wing will need to be redesigned. This chapter will present the results of the wing redesign, and the methods that were used. This chapter will start with a global overview of the wing modification. Then the test panels will be presented in more detail in Section 11.2. The structural implications and modifications are shown in Section 11.3. The wiring layout is presented in Section 11.4, and the modifications to the wing tip are shown in Section 11.5. This chapter will conclude with recommendation for future work.

11.1 Overall wing redesign

Multiple test sections will be identified on the wing. Each test section will allow accommodation of the desired flow control and measurement equipment. Test sections will only be placed in the outboard sections of the wing, from the engine mounting to the tip, as shown in Figure 11.1. The inboard sections of the wing will not be used. Fuselage interaction with the local flow makes it less interesting to perform measurements at the inboard section. Other reasons not to use the inboard section are the presence of landing gear and engine, which limit the structural freedom for modification, as well as the high structural loads being located close to the fuselage. Also this part of the wing remains reserved for fuel, limiting the modification possibilities as well.

As can be seen in Figure 11.1, the wing surface is covered in four different sections: A, B, C and D. In section A, space has been reserved for skin test panels. This section in its turn has been divided in four subsections. This has been done to create boundaries conditions for the geometry of the test panels and to specify a number of plug and play docks. Section B is issued with the wing tip section. Section C concerns the slats on the leading edge and section D the flaps on the trailing edge. Every section will be equipped with a docking station containing a certain set of connectors. More on the wiring in explained in Section 11.4.

Altogether, the following wing modifications have been identified:

- **Fuel tank redesign:** The fuel tank needs to be partially removed to enable accommodation of the required equipment. The inboard portion remains untouched.
- **Wing box structural redesign:** The wing box structure needs to be modified such that skin plate modules can be integrated easily.
- **Wing box interior redesign:** A convenient cable plan needs to be designed to create a 'plug and play' environment for the measurement and AFC systems.
- **Slat, flap, and wing tip redesign:** These should be designed such that they are modular and can equip the required AFC and measurement systems.

11.2 AFC integration in main wing

In the main wing, a modular panel system will be integrated such that AFC actuators can be removed and installed easily. To implement this modular system, the wing box has to be revised. A redesign shall

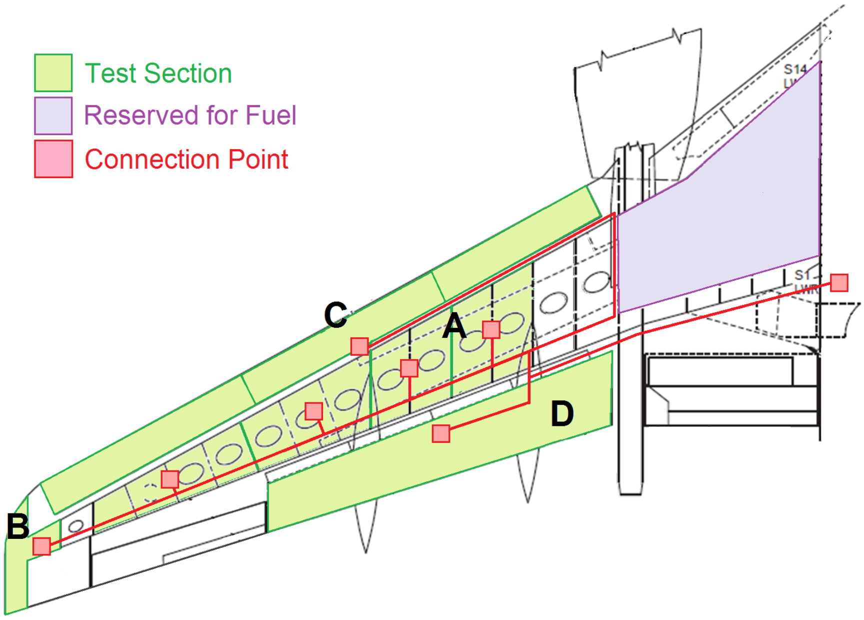


Figure 11.1: Detailed wing layout, including section division and cable plan

be performed to allow for unloaded test panels, to create an easily attachable and detachable wing tip and to modify the fuel tank to a feasible size.

The skin panels should be integrated such that they will not carry any loads, where the shape of the existing wing must be maintained. This is because of the following reasons:

- The panels will be manufactured by the customers. This creates high uncertainty about the structural quality of the panels.
- Loading of the panels cause deformation. These deformations may harm the AFC actuators.
- The client has more freedom in designing the part when it is not part of the structural body of the aircraft.

The structural redesign of the wing box will consist of lowering the upper surface by 5 cm. This initial guess is based on the requirement that the top skin should be able to accommodate AFC systems. This guess is checked and verified later on feasibility from a structural point of view in Section 11.3. By lowering the original skin, space is created for a second, non-load carrying top panel which can accommodate the actuators. As most AFC systems and measurement systems need only room in the order of millimetres to 2 centimetres, 5 cm of space will also allow extra space for connecting cabling and to accommodate for possible future systems that may require more space. The former top panel will be reinforced since it will experience higher stresses due to the decrease in height of the box. A schematic layout of the wing box layout is shown in Figure 11.2.

The panels will be fixed to the top skin by means of screws and nuts. The panels will have bolts on their bottom side which will fit in holes in the skin. Nuts are attached in the inside of the wing box that will fix the panels. Access to the inside of the wing box is possible by using the maintenance hatches on the bottom side of the wing, these were originally used for fuel tank maintenance. This solution will make it easy to attach and remove panels with minimal changes to the existing structure. The advantage of fixing the panels to the skin over fixing them to the spars is that smaller panels can be fitted in between panels in the front and rear.

All panels have a 5 cm offset from the load carrying skin and follow the curvature of the local wing. The dimensions of the test panels vary between locations on the wing. For example, the panel first panel in section A (see Figure 11.1) has a spanwise length of 1285 mm. Due to taper of the wing, the dimension in chord direction varies over the panel from 2225 mm at the inboard side to 1945 mm at the outboard side of the panel. Figure 11.3 displays the geometry of the panel at A. The top skin will not

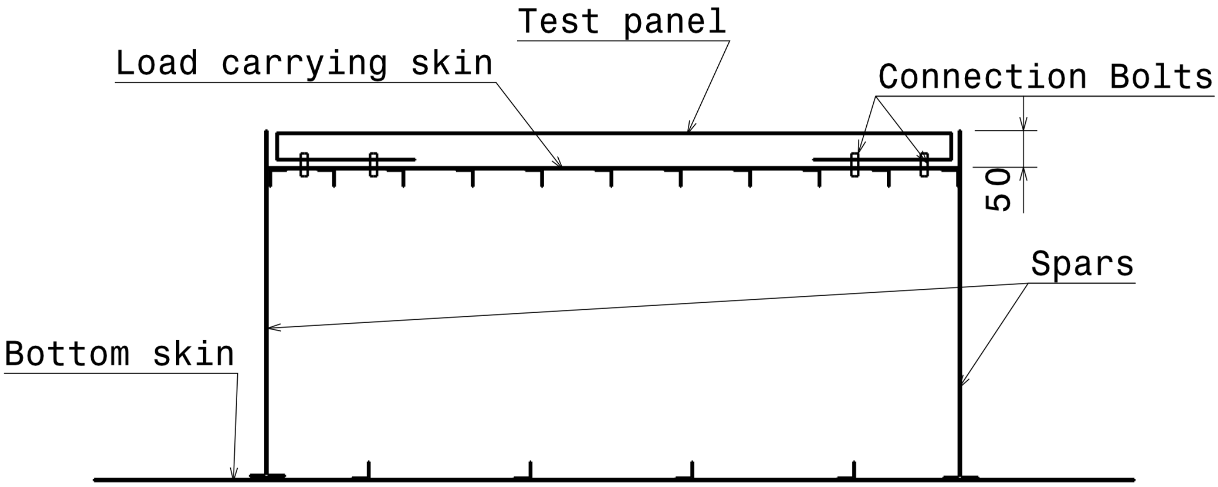


Figure 11.2: Schematic overview of the wing box layout. This figure is not to scale, the indicative dimensions are in mm.

carry loads, so the panel can be built thin-walled to save weight. The opening in the bottom side allows the passage of cabling to the systems and provides accessibility for e.g. placing the screws. The inner layout and dimensions of the panel can be varied as desired, adding AFC systems (a plasma actuator is shown for example) and measurement systems (PIV laser and pressure taps are shown in the picture). In the picture, the strain gauges are the transducers which turn the pressure difference in a wall tap into an electrical signal.

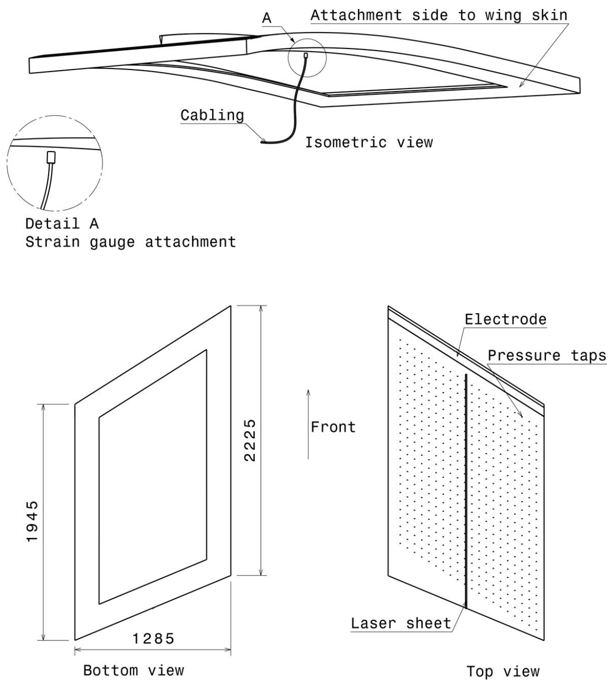


Figure 11.3: Example illustration of the geometry of a wing panel attached at wing section A. Dimensions shown are in mm.

11.3 Structural analysis

This section will focus on the structural redesign of the wing box. The AFC systems will be integrated only on the top skin of the outboard section of the wing. However, the top skin is an essential part of the wing box structure. It cannot be removed since it accommodates stringers, and it carries shear stresses. To enable unloaded modular skin panels, the effective load carrying skin will need to be lowered. To reduce peak stresses, a slope will be induced from the engine pylon to the first panel in section A, as shown in Figure 11.4. The wing box from the engine pylon inboard, is not redesigned. In this chapter, it will be checked how the redesign will affect the structural qualities of the wing box.

Recapping, the following are the main redesign efforts.

- The top skin plate will be lowered with dh at spanwise position 6.22 m$< y < 13.68$ m from the root.
- The top skin plate will have an additional slope of $tan^{-1}(dh/0.903)$ at spanwise position 5.32 m$< y < 6.22$ m from the root.

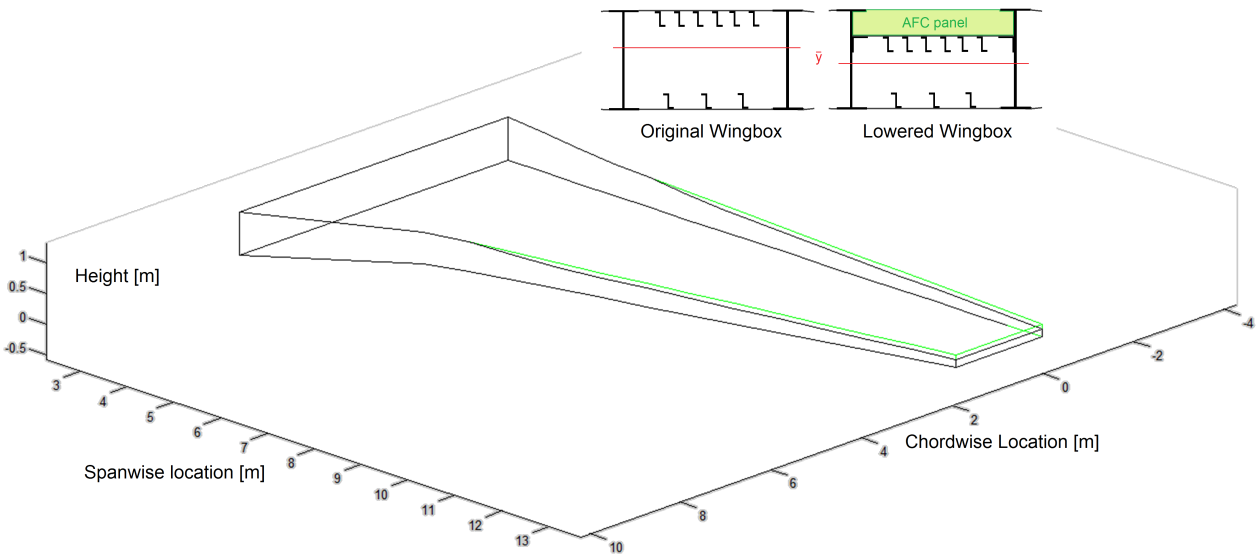


Figure 11.4: Three dimensional view of the wing box. The lowered skin panel is marked in green. In the top right two schematic cross sectional views are shown. The left one shows the original wing box, the right one shows the lowered skin panel. Here, n is 80.

The modifications will have the following impacts on the structure:

- The lowering of the top skin causes the bending resistance (moment of inertia) of the structure to decrease. The lowered top skin will need to be reinforced to preserve the bending resistance.
- The lowering of the top skin causes the torsional resistance (enclosed area) to decrease. and will result in an increase of shear flow. The skin panels should have an increase in thickness to cope with the increase of shear flow.
- The additional slope will increase the amount of induced shear force. This will result in an increase of shear flow. The skin panels should become thicker to cope with the additional shear stress.

The decrease in bending resistance is thought to be the most critical, since bending resistance decreases with a higher order term of height. To identify the additional structure required, the following steps were taken:

- The loads (shear force, bending moment, torsional moment) were identified along the span.
- The geometry of the existing structure was estimated. This structure was idealised using methods described in [11], the skin will carry no normal stress and booms are used.
- The top skin panel was lowered and the structure was analysed again. The required increase in inertia was calculated to maintain the same levels of stress.

- The required area to generate this inertia was used to estimate the weight increase of the wing box structure.

These steps now will be explained in the following subsections.

11.3.1 Wing loadings

The wing was divided into n sections. The average chord length (c_i) and surface area (S_i) were calculated for each section. For each section the lift and weight were estimated as well, as can be seen in Figure 11.5. Thrust and drag were not taken into account. This assumption is taken as the order of magnitude of these forces is less than those in the z-direction and therefore the stresses involved as well. Also, the moment of inertia in the x-direction remains unchanged when the upper skin is lowered. For each section the shear centre was assumed to be at $0.5c_i$. The forces are computed as follows.

- **Lift calculated:** Limit flight conditions were used with maximum take off weight ($W_{take-off} = 56472kg$) and the limit load factor ($n = 2.8$), such that $L = W_{take-off} \cdot n$. Then the lift per section was calculated as $dL_i = \frac{W_{take-off} \cdot n}{S_w} \cdot S_i$. It was assumed that the lift force applies at $0.25c_i$.
- **Weight calculated:** The structural weight of the wing was taken from the class II estimation. The structural weight was distributed in a similar way as the lift distribution; $dW_i = \frac{W_{wing}}{S_w} S_i$. The engine pylon weight was added at the engine location (5.32 m from the root). It was assumed that the weight is applied at the $0.5c_i$.

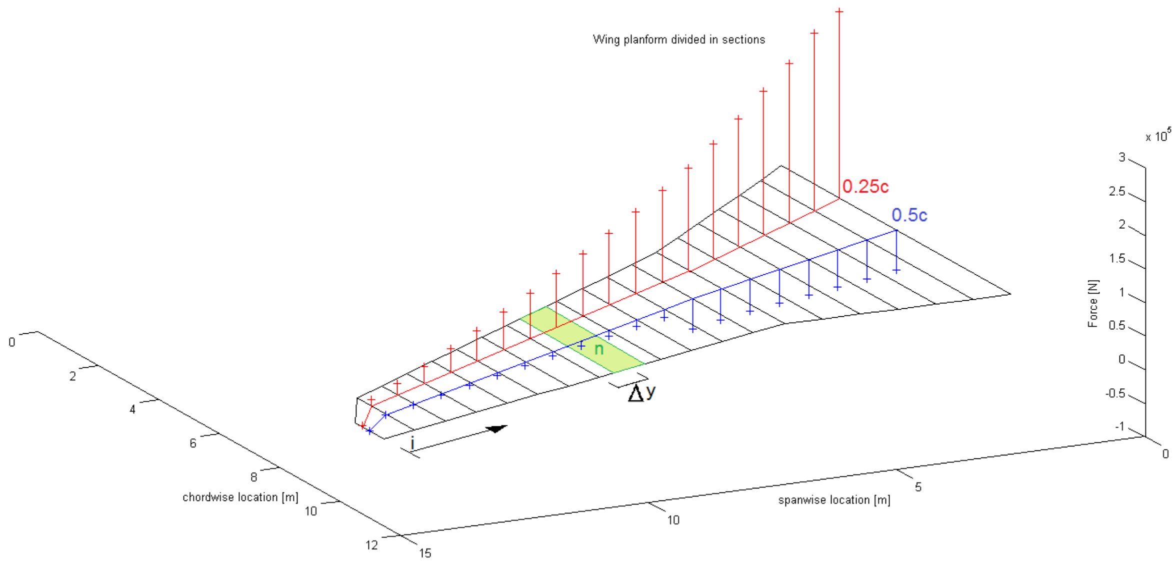


Figure 11.5: Three dimensional view of the planform. The planform has been divided into sections. For each section, the lift distribution is shown in red at $0.25c_i$ and the weight distribution is shown in blue at $0.5c_i$.

The shear force was calculated for each section using $dV_i = dL_i - dW_i$. The shear distribution, bending moment distribution, and torsional moment were calculated using Equation 11.1, Equation 11.2, and Equation 11.3. The resulting loading diagrams are shown in Figure 11.6.

$$V_n = \sum_{i=1}^{n} dV_i \tag{11.1}$$

$$M_n = \sum_{i=1}^{n} dV_i \cdot (n-i)\Delta y \tag{11.2}$$

$$T_n = \sum_{i=1}^{n} [-0.25c_i - (x_n - x_i) + 0.5c_n] \cdot dL_i \tag{11.3}$$

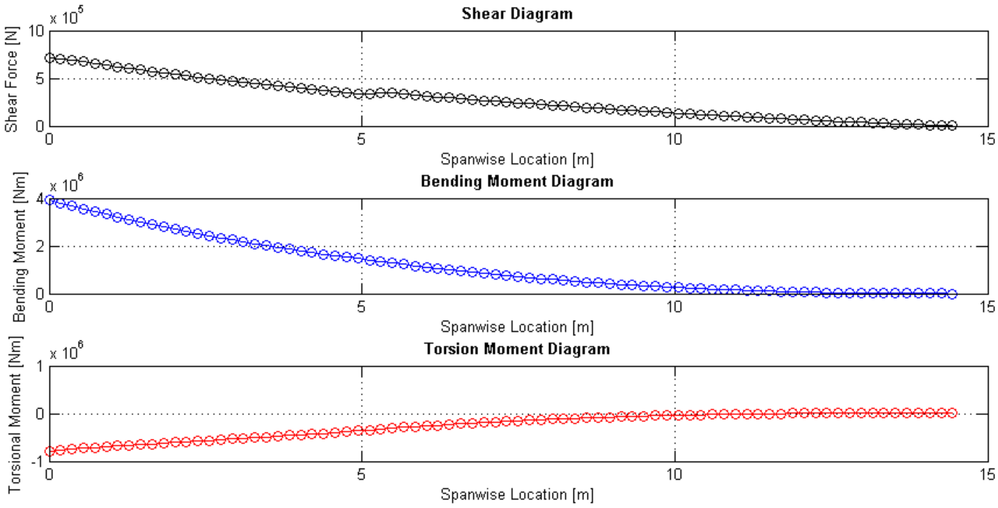


Figure 11.6: Shear force diagram, Bending moment diagram, and Torsional moment diagram showing the value of force/moment along the span. The value for n was 80.

Now the load distributions are known for the original wing box, a comparison can be made with the new wing box with lowered top skin.

11.3.2 Impact of lowering top skin

The geometry of the original structure was estimated. A simplified cross section was used with constant height. The following variables were used:

- Front spar thickness of $0.001c_i$
- Rear spar thickness of $0.0007c_i$
- Top skin thickness of $0.0003c_i$
- Bottom skin thickness of $0.0002c_i$
- Varying height and width as function of chord length and spanwise location (from [12])
- Varying amount of stringers on top and bottom skin (from [12]) with varying area from 250 mm^2 near the root to 80 mm^2 towards the tip

For each section the local cross section geometry was calculated. The structure was idealized using the methods described in [11]. The location x_{B_j}, z_{B_j} and the area B_j of each boom were calculated. The neutral axis and the moment of inertia were then determined for each section using Equation 11.4, and Equation 11.5. The new configuration with the lowered skin panel was then applied, and the required moment of inertia to maintain the same stress levels was calculates using Equation 11.6. The results are shown in Figure 11.7.

$$\bar{z} = \frac{\sum_{j=1}^{n} B_j \cdot z_{B_j}}{\sum_{j=1}^{n} B_j} \tag{11.4}$$

$$I_{xx} = \sum_{j=1}^{n} B_j \cdot \left(z_{B_j} - \bar{z}\right)^2 \tag{11.5}$$

$$\frac{dI_{xx}}{I_{xx}} = \left(\frac{z_{max_2}}{z_{max_1}} - 1\right) \tag{11.6}$$

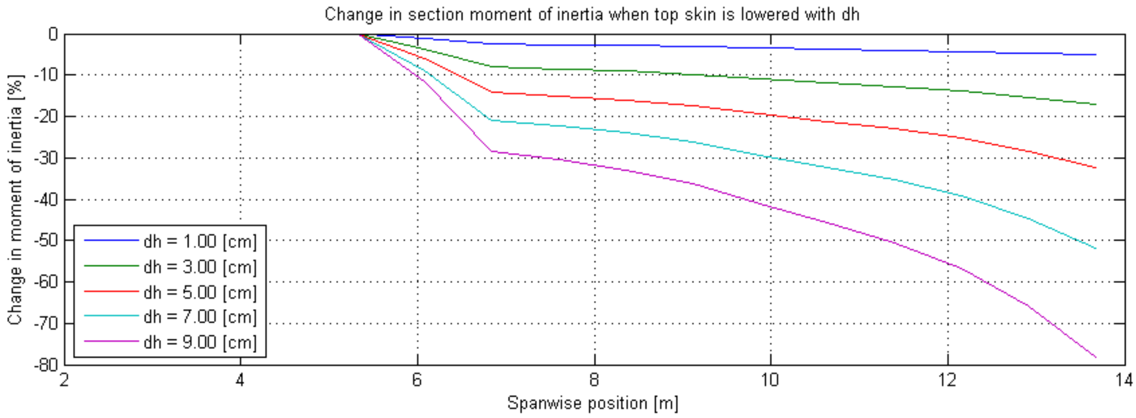


Figure 11.7: Decrease of moment of inertia over the span for different panel depths dh.

The moment of inertia decreases very fast in the sloped section, and decreases somewhat slower near the tip. To make up for the inertia, additional area is required. First the additional area was calculates using Equation 11.7. The change in mass was then estimates using Equation 11.8. The result is shown in Figure 11.8.

$$dA = \frac{dI}{\left(z_{max_2} - \overline{z_2}\right)^2} \tag{11.7}$$

$$dm = \left[\frac{dA}{A}\right] \cdot m\Delta y \tag{11.8}$$

The result provides an indication of additional structural weight to overcome the reduction in bending resistance. A panel depth of 0.05 m will cause structural weight to increase by 2.66% of the original structural weight. Further analysis should be done to check if additional structure should be used for shear and buckling loads.

11.4 Wiring installation in the wing

The actuator and equipment power, generated in the fuselage, needs to be conducted to the application area in the wings. Following from the discussion on the wiring installation in the fuselage (Section 13.6), this section will explain how the wiring is installed in the wing. Because of the different nature of cable purposes, this section consists of three subsections: the installation of power cables, pressure tubes and optic fibres. The optic fibres will be used for both the data transfer to the fuselage as for the laser

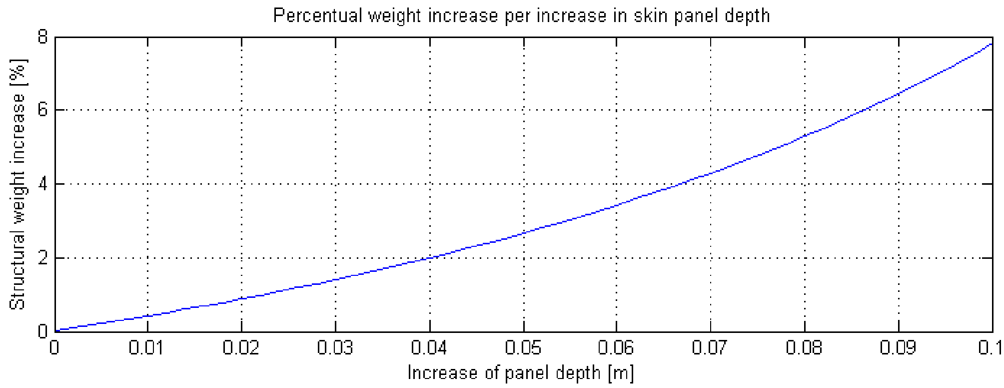


Figure 11.8: Decrease of moment of inertia over the span for different panel depths dh.

distribution coming from the fuselage. It should be noted that in the following subsections one wing is discussed, however all systems are installed in symmetry. In Table 11.1, an overview has been given for each section which kind of plugs will be available.

Table 11.1: The amount of plugs incorporated into the wing, indicated per section and subsystem.

Subsystem	**Section A**	**Section B**	**Section C**	**Section D**	**Total per system**
AFC	12	2	2	2	18
Power plug	8	1	1	1	11
Pressure plug	4	1	1	1	7
DH	16	2	2	2	22
Optic fibres	16	2	2	2	22
MS	12				12
Optic fibres	4				4
Power plug	8				8
Total per section	40	4	4	4	52

11.4.1 Power cables integration

Following from the discussion on wiring in the fuselage two types of power cables will go to the wing, being power for the AFC actuators and power for the measurement systems. From the Central Power Point (CPP) a cable bundle will run to the wing root fairing, crossing the data handling cable channel. This cable bundle will then enter the wing at the trailing edge due to the presence of fuel in the integral wing fuel tanks. The cabling will be fed into the flap hinge line. After the cabling to section D has branched off, the other cables will enter the wing through the flap actuator. After consultation with the stakeholders it was found that the initial idea of creating a cut-out in the rear spar would cause such a decrease in structural stiffness that this simply was not an option.

Once in the wing the cabling will dive underneath the wing box and enter the wing box structure through the lower skin. Once the cabling has entered the wing box structure, the sets will split again going to the individual test sections.

At first it was considered to use a power splitter station. In this way, only one set of cables needed to run from the power generator to the wing, from where the cables could be split further. It was however found that the amount of power required in the wing (about 6 kW for actuators and measurement systems combined) was too large to be split by a regular splitter. Normal four way splitters can accept mean power up to about 1 kW [13]. This was one of the reasons to abandon the idea of a power splitter.

Another reason for not using a power splitter, is that when using a splitter it is very difficult to separately control the individual actuators and measurement devices without shutting down an entire branch.

Following the previous argumentation it was decided to abandon the idea of a power splitter and lay cable bundles from every test section to the respective power generators. With the power cables split into the three remaining categories, each cable bundle will run to their respective section. The cables to section C will be fed into the hinge line of the slats. This way the connection between the slats and the wing structure does not need to be compromised. The other two cable bundles will run to sections A and B (the wing skin and wing tip sections, respectively), using the already existing holes in the ribs.

The cabling to section A will have to cross the load carrying wing skin at four locations. The cut-outs to be created will have a similar geometry and reinforcement as those already present in the ribs. Nevertheless, the wing skin needs to be locally reinforced to cope with the stress concentrations occurring around the holes. Here it should also be taken into account that different loads act on the skin of the wing box than on the ribs, which will have its influence on the way the cut-outs need to be reinforced.

The power line running to section A will again split into four branches to feed the four test sections present on the wing skin. When a set of cables has entered a section, it will be connected to a box attached to a rib. This box will connect the cables from the fuselage and the cables from the actuators and measurement systems in a secure way. An example of such a box can be found in Figure 11.9.

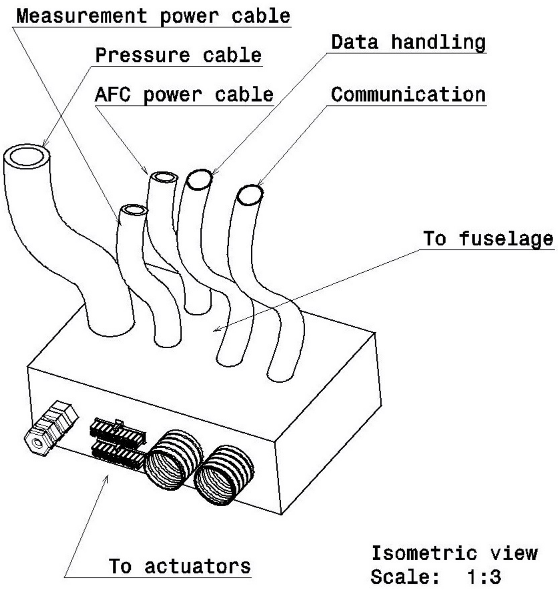


Figure 11.9: An example of a cable dock. This dock will be attached to a rib, right under the lowered skin. Through holes in the skin, the cables will be lead to the actuators

11.4.2 Pressure tubes

Besides power, some actuators are fed by pressure instead. Pressure will be available at all four sections, and will be distributed in the same way as the power cables, described in the former section. The pressure will be obtained from a pressure pump inside the fuselage. Differential pressure of less than 1 bar has to be produced, which can be easily obtained by a small electric pump. The pneumatic (pressure) system of the aircraft will remain untouched, mainly because this system may not be designed for pressure tapping by additional components and because the current solution is more simplistic than adapting the current system.

11.4.3 Optic fibres

Optic fibres are required for three things: data transfer from transducers to storage, communication between engineer booth and the actuators and provision of light (laser which will be used for Particle Image Velocimetry (PIV)) to the panel sections. The data transfer will consist of one line per transducer to a multiplexing device situated right before section A, span wise. From there on, all data will be combined and the amount of cables will be brought back to a minimum. Inside the fuselage, another multiplexing device will then spread the data again such that each transducer output can be stored separately. Also, a set of two cables is required to provide two way communication between the controllers in the engineer booth and the actuators. Lastly, one set of optic fibre cables is required to guide the laser produced inside the fuselage to the different test sections. All optic fibres will be lead through the wing together with the other wires and cables.

11.5 Wing tip integration

Winglets at the tip improve the performance of the aircraft by reducing induced drag. This reduction in drag in turn leads to lower fuel consumption (in the order of a couple of percent), lower emissions of CO_2 and NO_x and a decrease of noise.

For the wing tips to be modular, the panels as seen on Figure 11.1 in Section 11.1 are affected. The chosen Boeing 737-500 has the possibility to be retrofitted with a different style of winglets once the wing box is reinforced. Apart from the wing box redesign, the impact of the made modifications is small. Because the 500-series has the wings of the larger 737-400, therefore these are structurally overdesigned. This means that the strengthening of the wing box should only be minimal, since most of the larger wing loading induced by the winglets can be compensated for by the stronger structure. The partial redesign of the wing box should also account for the replacement of the outer (closure) ribs and a fitting for the wing tip at the tip of the spars. Once these structural modification are done, the replacement of the tip itself can be performed in a matter of hours [14].

Although the structure needs to be reinforced, the overall aircraft take-off weight with wing tip devices is reduced with around 1 % of total weight [15]. In the case of the 737-800, each kg added in structural weight saved 2.5 kg of fuel per flight [15].

To address unwanted effects such as flutter, the weight and centre of gravity location of the winglet needs to be taken into account in the design of the wing tip device. In some cases, wing tip ballast eliminates the need for wing box redesign [14]

11.6 Recommendations

The structural work of the wing box redesign has still been done fairly globally. Without details on the skin thickness, rivet spacing, etc. the calculations are still estimations. Detailed dimensions of the wing box of the 737-500 should be obtained to obtain more accurate results. With these results, the lowered skin dimensions can be determined, as well as the required reinforcement around the cable holes. The same goes for the wing skin panels. No details are known yet about the number of bolts required to fix a panel to the wing box, or about the bolt dimensions. This would be the following step to perform in the structural analysis.

Also, more detailed example test panels should be made. This is to give the designers better insight in the requirements which should be given when a client is about to make such a test panel. Also, yet unforeseen problems can be detected and solved by the designers, in terms of e.g. skin panel integration.

12
Pylon design

This chapter will present the design process performed to arrive at a suitable pylon and test section design. It will chronologically present the individual topics of this design cycle. First, characteristics with respect to the stability and control of the entire aircraft is discussed. In the next section the pylon wake is discussed. According to this wake, the test section sizing and pylon sizing are presented. And finally the pylon sizing is performed. A detailed technical drawing is added in Chapter 15.

12.1 Stability and control

Placing a pylon mounted test section on top of the aircraft will affects stability and control. During testing, lift and drag forces produce moments around the centre of gravity. These must stay within reasonable boundaries to avoid an unstable or uncontrollable aircraft. Therefore, the sizing of the test section and pylon must be performed carefully.

12.1.1 Moment balance

The location of the centre of gravity is the most important variable, when designing for stability and controllability. Lift and drag forces will produce a moment around this point, meaning the position of the center of gravity directly influences the aircraft moment coefficient. The equation for C_{m_α} can be deducted using a simple free body diagram. The result is visible in Equation 12.1, below.

$$C_{m_\alpha} = C_{L_{\alpha_w}}\left(\frac{x_{cg}-x_w}{\bar{c}}\right) - C_{L_{\alpha_h}}\left(1-\frac{d\epsilon}{d\alpha}\right)\left(\frac{S_h}{S}\right)\left(\frac{V_h}{V}\right)^2\left(\frac{x_h-x_{cg}}{\bar{c}}\right) + C_{L_{\alpha_t}}\left(\frac{S_t}{S}\right)\left(\frac{x_{cg}-x_t}{\bar{c}}\right) \quad (12.1)$$

Where:

- C_{m_α} is the derivative of the aircraft moment coefficient with respect to the angle of attack [-]
- The subscript w is the abbreviation for the main wing, h for the horizontal tail and t for the test section
- C_{L_α} is the derivative of the lift coefficient of a surface to the angle of attack [-]
- x_{cg} is the location of the centre of gravity with respect to the most forward point of the aircraft [m]
- x is the location of the aerodynamic centre of the mean aerodynamic chord of a lifting surface with respect to the most forward point of the aircraft [m]
- $\bar{c}$ is the length of the mean aerodynamic chord of the main wing [m]
- $\frac{d\epsilon}{d\alpha}$ is the downwash of the main wing [-]
- $\frac{S_h}{S}$ is the ratio between the surface area of the main wing S and the surface area of the horizontal stabiliser S_h [-]
- $\frac{S_t}{S}$ is the ratio between the surface area of the test section S_t and the surface area of the main wing S [-]
- $\left(\frac{V_h}{V}\right)^2$ is the ratio between the velocity of the main wing and over the horizontal stabiliser squared [-]

For Equation 12.1 it is assumed that the moment caused by the drag of the pylon and the test wing is negligible to the moments caused by the lift of the different lifting surfaces. This assumption is assumed valid for this phase as the drag of the pylon and the test wing is more than an order of magnitude smaller than the lift of the test wing. It should be noted that the struts and the joint housing are designed to keep the frontal area to a minimum, and are shaped as airfoils to minimize drag. The moments caused by the drag of the main wing and the horizontal tail are neglected because the drag is more than an order of magnitude smaller than the lift of those surfaces, and the moment arm is for these moments is relatively small.

From statistical data [16] and information obtained from XFLR 5, the x position and the mean aerodynamic chord of both the main wing and the horizontal stabilizer were determined, and the ratio between the main wing and the horizontal stabilizer was found. The squared ratio between the velocity at the tail over the velocity at the main wing was found to be 0.85 from statistics [17]. The class II weight estimation predicted the x position of the center of gravity of the aircraft and the weight op the pylon group, which was found to be 668.2 kg Section 12.3. With these parameters known, only the downwash, lift coefficient derivatives and the moment coefficient derivative remain to be determined to solve Equation 12.1. The next subsections will explain their deduction and computation.

12.1.2 Downwash

The downwash is computed according to Equation 12.2[18].

$$\left(\frac{d\epsilon}{d\alpha}\right) = \frac{K_{\epsilon_\Lambda}}{K_{\epsilon_{\Lambda=0}}}(A + B \cdot C)\frac{C_{L_{\alpha_w}}}{\pi AR_w} \tag{12.2}$$

$$\begin{aligned} A &= \frac{r}{r^2 + m_{tv}^2}\frac{0.4876}{\sqrt{r^2 + 0.6319 + m_{tv}^2}} \\ B &= 1 + \left(\frac{r^2}{r^2 + 0.7915 + 5.0734 m_{tv}^2}\right)^{0.3113} \\ C &= 1 - \sqrt{\frac{m_{tv}^2}{1 + m_{tv}^2}} \end{aligned} \tag{12.3}$$

Where:

- K_{ϵ_Λ} accounts for the wing sweep angle effect [-]. Finding this value can be done using Equation 12.4
- $K_{\epsilon_{\Lambda=0}}$ accounts for the wing sweep angle effect [-]. Finding this value can be done using Equation 12.5
    - Λ is the sweep angle at half chord of a lifting surface [deg]
- r represents the horizontal distance between aerodynamic centres of the main wing and horizontal tail [-]
- m_{tv} represents the vertical distance between the zero-lift line of the main wing root chord and the horizontal tail aerodynamic center [-]
- AR_w is the aspect ratio of the main wing [-]

$$K_{\epsilon_\Lambda} = \frac{0.1124 + 0.1265\Lambda + 0.1766\Lambda^2}{r^2} + \frac{0.1024}{r} + 2 \tag{12.4}$$

$$K_{\epsilon_\Lambda} = \frac{0.1124}{r^2} + \frac{0.1024}{r} + 2 \tag{12.5}$$

Using the planform drawings of the aircraft, the unknown distances can be estimated. This results in a value for the downwash in front of the horizontal stabiliser.

12.1.3 Lift coefficient derivative

Next, the lift coefficient derivatives need to be found. This can be done using the DATCOM formula depicted in Equation 12.6 [18].

$$C_{L_\alpha} = \frac{2\pi AR}{2 + \sqrt{4 + \left(\frac{AR\beta}{\eta}\right)^2 \left(1 + \frac{tan^2\left(\Lambda_{0.5C_h}\right)}{\beta^2}\right)}} \tag{12.6}$$

Where:

- β is the Mach correction factor, computed according to $\beta = \sqrt{1 - M^2}$ [-]
- η is the airfoil efficiency and is assumed equal to 0.95 [-]

The sweep angles are previously known and the Mach number is taken to be 0.5, the maximum speed at which testing may occur. This results in that only the derivative of the moment coefficient is unknown.

12.1.4 Moment coefficient derivative and results

The moment coefficient was found by first computing the stability without the pylon. This initial stability must always be guaranteed and it was decided that the pylon could only influence it marginally. This margin was set at of 10%, regardless if this is an in- or decrement, it results in the same value. A straight, non tapered test wing was chosen for this analysis. The wing has an aspect ratio of 8, a surface area of $8m^2$, and is build up from the B737-500 root airfoil. With the lift gradient of this wing, calculated with Equation 12.6, the pylon wing position was found to be 14.31 meters from the nose. This calculated position was expected, as the pylon mounted test wing will be close to the center of gravity, making the moment arm and hence the moment small.

12.2 Pylon wake

Now that the location and the size of the test section are known, the must of the pylon must be computed. The pylon sizing will proceed for the most extreme case. It will be designed for the situation where maximum lift and or drag occurs, to ensure the pylon will be able to resist every load it will ever encounter during its lifetime.

Placing a pylon in front of the empennage could be risky, as wake of a test section could interfere with the performance of lifting and controlling surfaces. Due to the lack of CFD (Computational fluid dynamics) tools, the wake could not be investigated in detail. Therefore, a different approach by Raymer [19] was used. The rule states that the horizontal tail must be carefully positioned to avoid blanking of the rudder, when designing a fuselage mounted tail. As can be seen in Figure 12.1, some rules of thumb can be used to size the wake of the stabiliser. These rules can be applied to the test section and then its wake can be visualised. This results in boundaries between which no operating surfaces should be placed. This means that in these areas, the rudder can not be positioned.

Raymer also provides information on T-tail situations. In this case, the wing can produce a wake which can screen the stabiliser. The angles found in Figure 12.2 can also be applied to the test section. This results in two extra boundaries, in which it is advised not to design control surfaces. As the test section will be placed upside down, the wake is turned upside-down as well. This results in an upwash with respect to the rest of the surfaces. It is assumed that this upwash does not influence the horizontal tail. The distance is too large and the wake will mostly be intercepted by the fuselage. The influence on the main wings will be assumed negligible as well.

As the x-position of the pylon is already fixed, only the height remains variable. Using drawings, the height is adjusted so that a minimal portion of the vertical fin is affected. The result is shown in Figure 12.3. This situation occurs when the test section is placed at a height of 3 metres. The two lines directed upwards constrain the green area in which it is advisable to place no control surfaces, as dedicated in Figure 12.1. As can be seen, a minimal portion of the rudder is affected.

The two lines pointing down confine another area which can also blank surfaces. In this region no surfaces are present, and the wake will be reduced by the presence of the fuselage in this section. The region is shown for an angle of attack of the test section of 10 degrees and 20 degrees. They bound the blue and red area respectively.

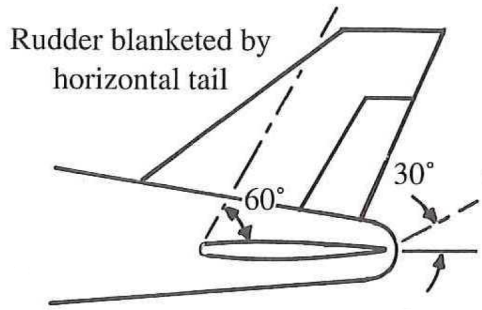


Figure 12.1: Tail geometry for spin recovery.

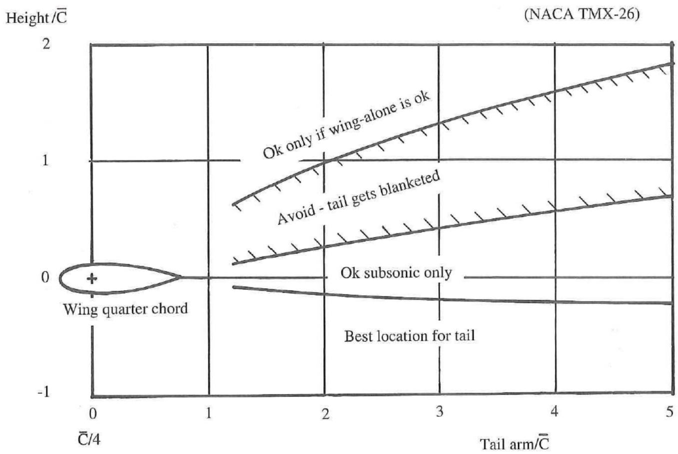


Figure 12.2: Aft tail positioning.

As can be seen, the blue area completely screens the rudder. A miscalculation had been made during the design process. Therefore, the test section could possibly affect the rudder. However, it must be noted that this situation occurs at an angle of attack of 10 degrees. Most airfoils would not have been stalled at this point, so the area will likely move upward. Also, an airfoil wake is not discrete and dissipates over a distance. This means that the boundaries are not precisely as shown.

Lastly it must be mentioned that the pylon cannot be placed any higher due to structural reasons. The interference of the structure and housing is assumed to be at a minimum at this pylon height. A solution would be to increase the distance between the test wing and the vertical fin. This could be investigated in further design iterations. An extreme solution would be to use an other base aircraft, with a longer fuselage.

12.3 Test section and pylon sizing

This section will discuss the method used to compute the main dimensions of the test section and the pylon. First, the dimensions of the test section are determined. Subsequently, the loads on the pylon can be calculated in order to determine the final pylon dimensions.

The loads are dependent on the aerodynamic forces, which are unknown at this point. However, as mentioned before, a worst case scenario is simulated in which the lift and drag force are maximised. The wing span and surface area have been derived from the aspect ratio, found in Section 12.1. The chord of the test section was set at 1 m in order to provide sufficient space for accompanying systems. The dimensions of the test section are given in Table 12.1.

The test section is carried by a truss system as indicated with the free body diagram in Figure 12.4. The four joints of the truss are simplified to make the system statically determined. The test section is mounted upside down to enable the PIV system to measure the flow. This results in a lift force in the positive z-direction, additional to the weight of the test section. The weight of the truss system itself has been neglected in this structural calculation since it is much smaller than the other forces involved.

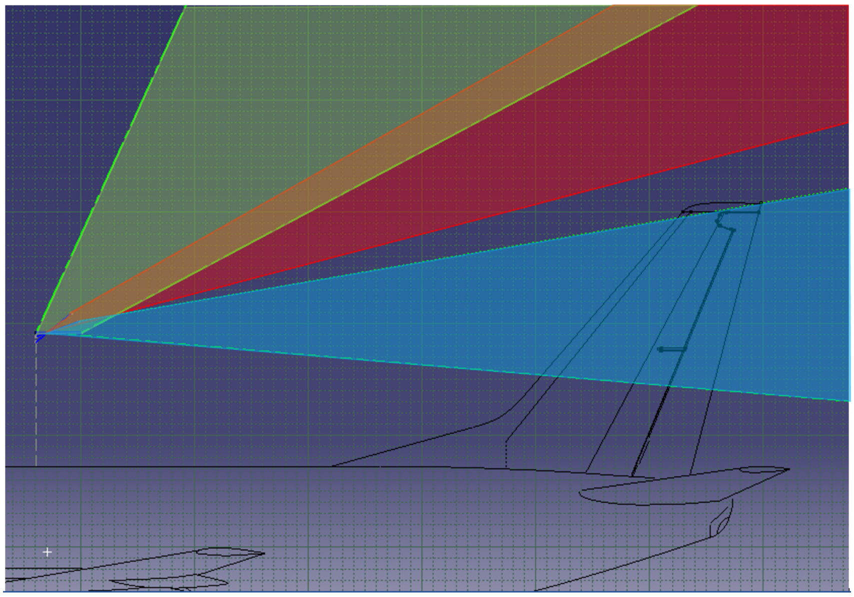


Figure 12.3: Boundaries which visualise the wake of the test section.

Table 12.1: Dimensions of the test section

Parameter	b_t [m]	c_t [m]	S_t [m^2]	AR_t [-]
	8.0	1.0	8.0	8.0

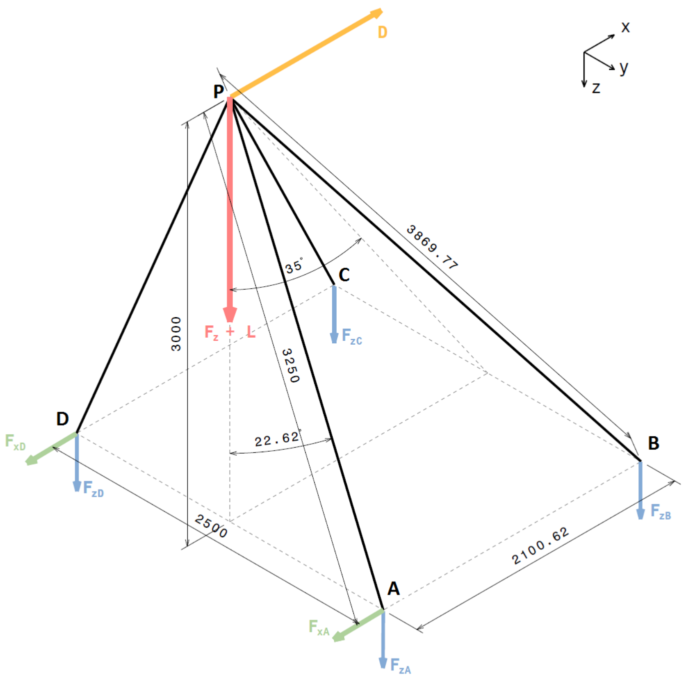


Figure 12.4: 3D free body diagram of the pylon.

The angle between the xz-plane and the strut AP is determined by the z-location of the test section and the width of the fuselage. This angle is introduced to minimise the influence of the test section on the flow in front of the tail and maintain a high tail efficiency by helping it keeping clear of turbulent air. The angle between the yz-plane and the rear of the mid section has been introduced to compensate for the drag force. The larger this angle will be, the better the struts BP and CP can compensate for the drag. However, since the struts have to be mounted to the fuselage, the angle is restricted. In order to find the optimal angle in terms of the mass of the strut, the following procedure has been used.

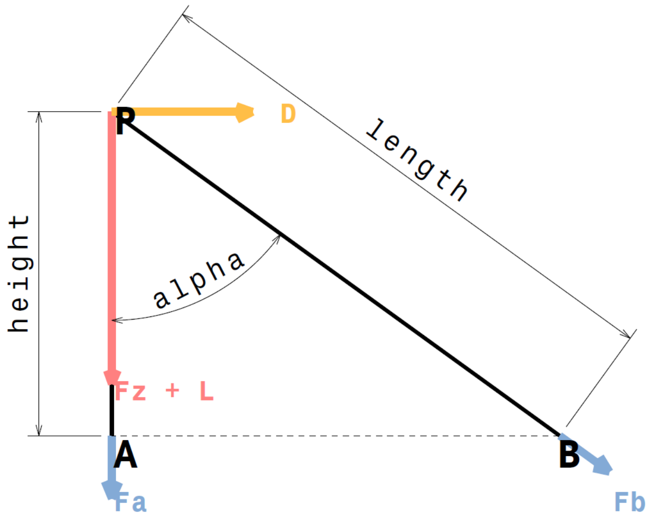


Figure 12.5: Truss system pylon 2D.

1. The truss has been simplified to a 2D case, presented in Figure 12.5, since the truss is completely symmetric in the xz-plane. The forces in the struts AP and BP have been determined for varying angles of attack and different ratio's of lift and drag forces.
2. With Equation 12.7, the required cross-sectional area A has been calculated for a given length L of the strut. This length is dependent of the angle alpha, visible in the figure. The displacement δ and Young's modules E have been held constant.
$$\delta = \frac{PL}{EA} \tag{12.7}$$
3. The cross-sectional area has been multiplied with the strut length to find the volume V. This value can be multiplied with a given material density to calculate the mass.
4. The angle between the struts AP and BP has been plotted against the volume, to find the angle for which the volume is minimal. The plot is shown in Figure 12.6. It can be seen that the angle for which the strut volume is minimal, is 35 degrees.

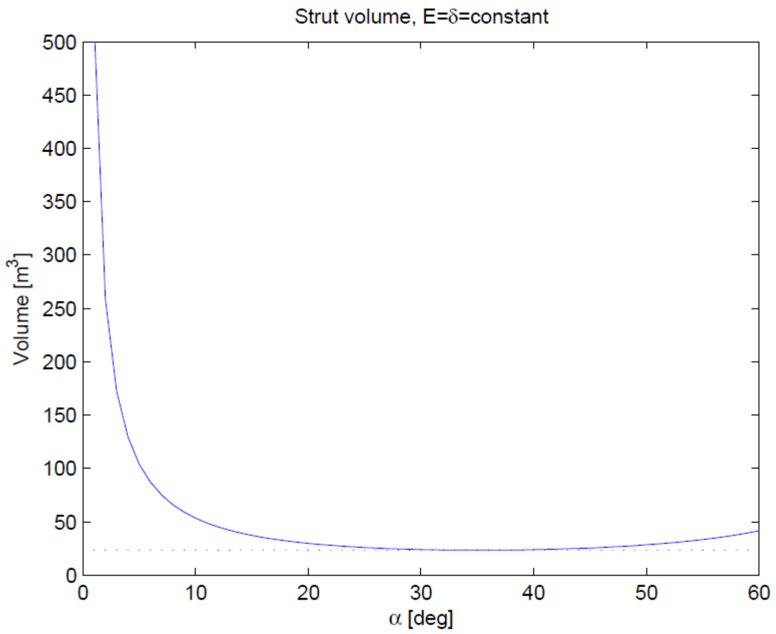


Figure 12.6: Strut volume per angle of attack. δ=E=constant.

The pylon sizing is continued by calculating the loads in the struts to determine the necessary cross-sectional area to withstand the loads. The geometry of cross-sectional area has been chosen to be circular for simplicity. Such a cross section is only variable with one parameter, as long as the thickness is fixed. The required thickness and radius of the cross-section have been determined by meeting following three load cases:

- Normal yield strength, using Equation 12.7[11]
- Buckling load, using Equation 12.8 [11], where P_{cr} is the critical buckling load in N, E the Young's modules in Pa, I the moment of inertia in m^4, K the effective column length factor and L the length of the strut in metres.
$$P_{cr} = \frac{\pi^2 EI}{(KL)^2} \tag{12.8}$$
- Shear yield stress, using Equation 12.9[11], where τ_{max} is the maximum shear yield stress in Pa, S the shear force in N, R the outer-radius of the cross-section in m and t the thickness of the strut in m.
$$\tau_{max} = \frac{S}{\pi R\ t} \tag{12.9}$$

The maximum aerodynamic loads the pylon has to withstand, have been calculated using Equation 12.10 and Equation 12.11. The situation when this occurs has the parameters as shown in Table 12.2. A thick airfoil, the B737-500 root chord airfoil, has been chosen to obtain large lift and drag coefficients. Using *XFLR5*, the maximum lift (C_L) and drag (C_D) coefficients have been computed. The surface area S has been determined previously and the test velocity V has been set to Mach 0.5 at 3000 m altitude. Finally, the weight of the pylon wing (F_z) has been computed by downscaling the weight of the main wing of the B737-500 linearly to the matching dimensions of the pylon wing, resulting in a mass of 570 kg. The free body diagram can be seen in Figure 12.4

$$L = C_{L_t}\frac{1}{2}\rho V^2 S_t \tag{12.10}$$

$$D = C_{D_t}\frac{1}{2}\rho V^2 S_t \tag{12.11}$$

Table 12.2: Worst case aerodynamic forces B737-500 root chord airfoil

Parameter	**M [-]**	**h [m]**	α_t **[deg]**	**C$_{L_t}$ [-]**	**C$_{D_t}$ [-]**
	0.5	3000	20	1.41	0.22

The final dimensions of the struts and magnitude of the forces in the joints are presented in Table 12.3. The values in the table, like the shear and normal forces, have been computed using Matlab. The radius and thickness of the struts have be determined, while a safety factor of 2.5 was applied to the load calculations mentioned previously, and using the material properties of aluminium 2024. The lift and drag forces used to compute these strut forces and stresses, form constraints on the test wing design a client could use. The maximum lift and drag must be adhered in order to safely conduct tests. This means that the combination of the aerodynamic coefficients and surface area is limited. Also the wingspan is bounded. This is a regulation which is communicated to the client.

Table 12.3: Dimensions and forces of the struts and joints

Joint forces	[N]	**Dimension**	**Unit**	**AP/DP**	**BP/CP**
Fx_A	14446.59	P$_{cr}$	[N]	1136967	801944.7
Fx_D	14446.59	S	[N]	32176.76	21336.15
Fz_A	-74753.5	N	[N]	-69003.2	-69003.2
Fz_B	-20631.9				
Fz_C	-20631.9	R$_{out}$	[m]	0.075	0.075
Fz_D	-74753.5	t	[m]	0.007	0.0045
		l	[m]	3.00	3.86
		m	[kg]	27.6	21.5

It had been decided to ignore some of the forces in the process of arriving at this solution. If all forces in the 3D problem need to be taken into account, the indeterminate problem was to strenuous and time-consuming to solve. Therefore, it has been chosen to replace the two rear fixed joints with joints on rollers. This means that the forces in x- and y-direction are ignored. However, the ignored forces are not neglected. When the simplified problem is solved, the ignored forces, in the x- and y-direction, were equalled to those forces in the z-direction. This means that, in the end, all the forces are larger than they actually are. The result is that the joints are overdesigned, which is beneficial as this acts as a safety factor in the design.

12.4 Test wing integration

The test section is designed to be completely modular. The pylon will feature a joint on which the client can mount a desired test wing. The dimensions and properties of this wing are constrained by the pylon capabilities. Each test section will therefore be checked and tested in order to make sure the maximum load on the pylon is not exceeded.

12.4.1 Mounting

The joint on which the section will be mounted pivots around an axis longitudinally parallel to the wing. In this way, the angle of attack of the test wing is adjustable. The revolving axis is mounted in a housing on top of the struts. The joint consists of a number of bolts on which the section can be mounted. To mount the test wing, the central skin panels will need to be detachable to grant access to these bolts, as can be seen in Figure 12.7. These detachable skin panels also provide access to the cabling, so that the cables can be guided to the outer sections of the test wing correctly.

On the surface of the joint, a hole is present as can be seen in Figure 12.7. The cabling for the test wing, for power, a laser beam, differential pressure and data handling, will run through this hole. To maintain structural strength, the rotating axis will be solid. Therefore, the cables will exit the joint block on the rear side, which is not visible in Figure 12.7. From here the cables will run to the support structure, and through the support system to the fuselage. A flexible protective pipe will house the cables, between the joint block and the strut, to protect the cables from the outside environment.

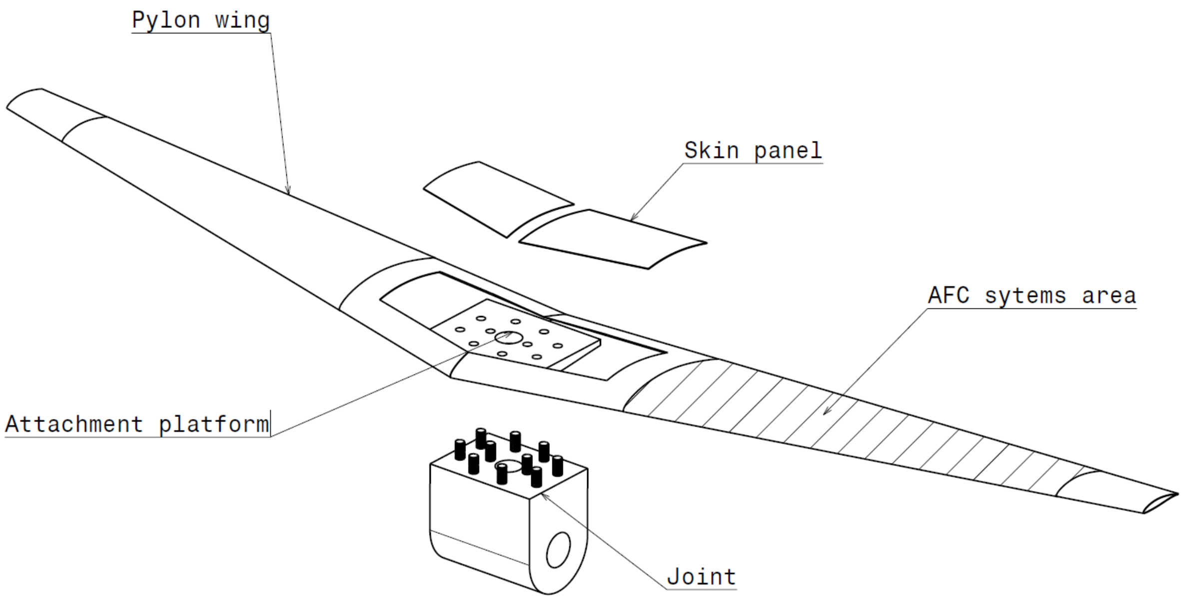


Figure 12.7: 3D view of the test section.

12.4.2 Actuators

Every type of actuator can be mounted on the test wing. As the client develops its own test section, they are free to choose the different types. As the test section is not essentially a lifting surface, and the integrity of the construction is of less concern, more structurally demanding actuator types are possible to mount. Since the influence of any electromagnetic fields is negligible at this location, systems that produce a significant amount of electromagnetic fields such as plasma actuators can be mounted safely.

It is necessary that every wing section and actuator placement is symmetric. Asymmetrical shapes and loading will result in moments for which the pylon is not designed. This will be one of the requirements communicated to the client. Shapes other than wings are can also be tested, i.e. an engine pylon or an empennage structure.

12.4.3 Measurement systems

The pylon can accommodate several measurement systems. As data transfer cables are readily available, other systems, desired by the client, can be used as well. The systems which are permanently mounted on the pylon are a PIV system, load cells and accelerometers.

To visualise the boundary layer on the top surface of the test wing, a PIV system is present. The wing is placed upside down such that a camera placed in the joint housing can record the flow. The required laser sheet will be aimed downwards from the surface of the wing. The laser beam will be guided along optic fibres to the required location and then be transformed to a sheet by optics. A thin transparent window in a skin panel of the test section will let the laser through.

The lift and drag forces are very interesting to record. To this end, load cells will be present on the joint. The displacement of the attachment to the test wing can be measured. These can be translated to forces and moments. Lift enhancement and drag reduction can be measured with tests in which the actuators are turned off and on. By using this system, for instance the drag of skin panels can be tested on a model as well. A new type of surface material or paint can be investigated using such a set up.

A sudden gust can result in forces which produce noise in the measurements of lift and drag. In order to be able to filter these out later, these forces need to be known. Therefore accelerometers are placed on the pylon. They are placed as close to the test section as possible, while remaining on the housing itself. The movement of the aircraft itself can be monitored and this data can take the influence of gusts on the movement of the test wing into account. If these were placed at the bottom of the struts, the deformation of the struts would need to be compensated for as well.

12.5 Fuselage integration

The test section produces aerodynamic loads, which have to be carried by the fuselage structure. This section will elaborate upon the integration of the pylon into the fuselage structure. The reinforcements and integration of the pylon are based on Airborn Warning And Control Systems (AWACS) aircraft, which carry large radar systems on top of the fuselage [20]. The weight of these radar systems can be compared to the lift produced by the the test section, since the test section is mounted upside down. Therefore already existing solutions of AWACS have been explored and analysed.

The section is structured as follows. First, the joints will be discussed, which transfer the loads into the fuselage structure. This will only be discussed on conceptual level and will not include the sizing en dimensioning of the joints. Second, the load transfer from the joints into the frames and wing box will be presented.

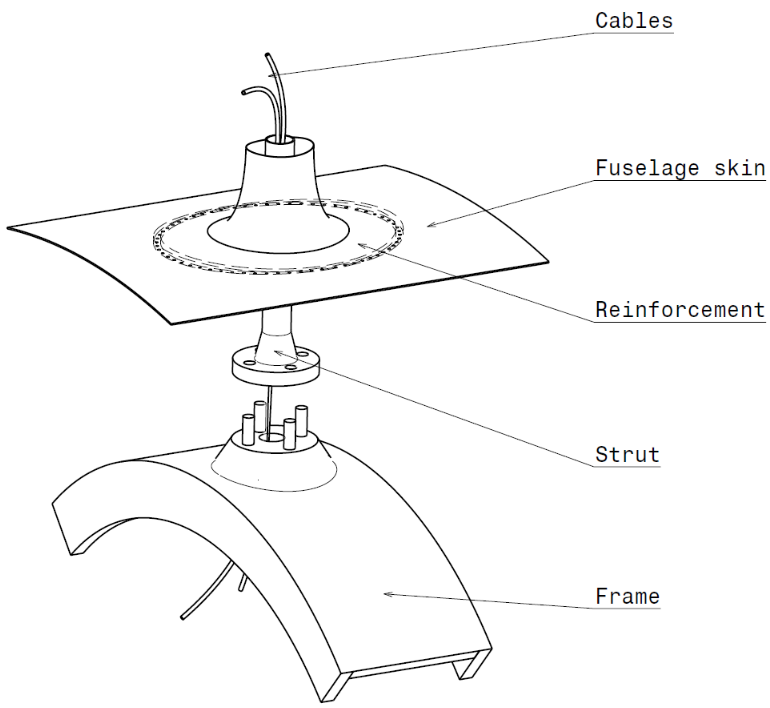


Figure 12.8: Pylon-fuselage integration

The pylon is mounted directly onto circular frames, similar to the ones already in the fuselage. The frames will be added to the initial structure and are designed to carry the loads, produced by the test section. Large detachable joints are installed on top of the frames to allow attaching and detaching of the pylon. To realise these direct connections, holes in the skin panels on top of the fuselage have to be constructed. These holes are reinforced similar to fuselage windows. To improve the aerodynamic properties of the joints, fairings have been installed. These fairings will not carry any stresses and can easily be attached or detached. The concept of this joint is presented in Figure 12.8

In order to lead the stresses introduced by the pylon through the joint to the rest of the fuselage, the existing fuselage structure needs to be reinforced. Wider flange frames and extra circular flange frames are introduced to cope with the loads underneath the location of the pylon struts. The technique of decreasing the frame pitch and using more frames locally is similar to that used for AWACS surveillance aircraft. The complete modification is carried out by first detaching the upper fuselage section, as can be seen in Figure 12.9, from the rest of the airframe.

Figure 12.9: The upper fuselagepart as it is mounted during production. [21]

After the cut-outs, on top of the fuselage to accommodate the struts, are made, the modified frames are attached to the structure. These are connected to the wing box through the floor and with each other with stringers. The exact location and amount of the frames needs to be specified during the detailed design. During the interior design, it was made sure that adequate space was reserved to accommodate these additional supports.

12.6 Recommendations

The designed pylon and test section are the results of a first design iteration. This means that the current parameters are only estimates of the final characteristics of the design. Detailed situations, as an aircraft stall or dive, have not been taken into account. Due to the limited time available, these iterations will not be performed. Even though, some recommendations can be made to do so.

Multiple design iterations can also lead to a better shaped joint housing on top of the pylon. The droplet shaped housing can be refined, looking at the amount of drag produced. Computational Fluid Dynamics (CFD) also could come in handy when redesigning.

When CFD software is available, details about the test section and wing wake can be found. This could have impact on redesign of the empennage. Conversely, if no redesign of the tail is wished, this could impact the test section sizing. An extreme case for the test wing could be computed to check the influence on the vertical fin and the horizontal stabiliser.

In the last weeks, some simple CFD computations became available through a tutor. The wing could be modelled and its wake computed in three dimensions. Due to insufficient time, the results can not be interpreted. However, the figures below will visualise the recommendation for future work.

In Figure 12.10 the velocity in the x-direction is shown at a free stream velocity of 150 m/s. The airfoil is the same as in previous calculations and is set at an angle of 20 degrees. The blue and white areas represent a lower velocity. This could result in loss of efficiency of the rudder, because the dynamic pressure will be reduced. Figure 12.11 shows the velocities in three dimensions. The intersecting planes are placed at 5, 10 and 15 metres behind the wing. In the legend, the x-velocity component is depicted.

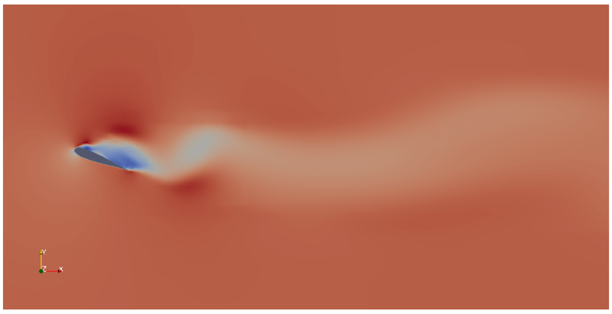


Figure 12.10: CFD visualisation of the wake of the test section airfoil, in x-direction.

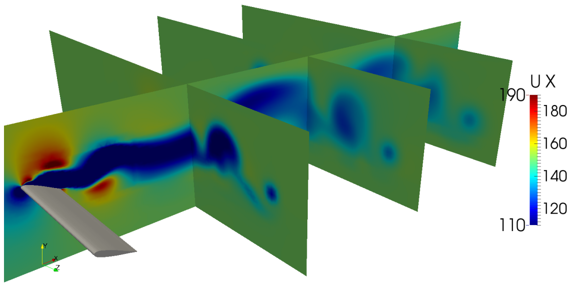


Figure 12.11: 3D CFD visualisation of the wake of the test section airfoil

13 Interior design

This chapter will focus on how the interior of the base aircraft needs to be modified to fit its new purposes. Due to the fact that the aircraft no longer needs to have the ability to transport passengers, the original furnishing dedicated to the passengers is taken out of the aircraft. This chapter is structured as such that the starting point of the design is an empty fuselage.

13.1 Engineering booth

An important areas in the aircraft is the engineering booth. This area is reserved for the test engineers and the client to regulate, monitor and verify the tests and some of the results during flight. The booth will run from the first entrance door behind the cockpit, until the beginning of the wing area. A schematic representation of the engineering booth is given in Figure 13.1.

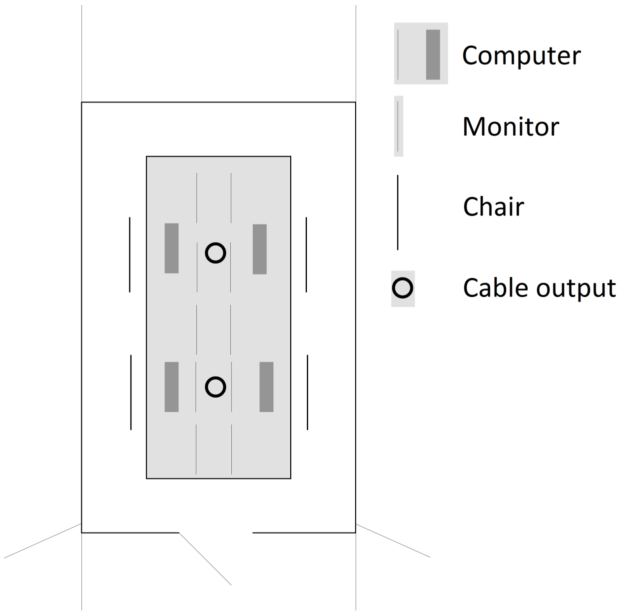


Figure 13.1: Schematic representation of the engineering booth

The booth area is comparable with a conference room, having a conference table with a large amount of monitors. The preliminary processing and storage will be done at another location in the aircraft. The visual verification of the data will be done by the engineers.

A single long table in the middle has been chosen for communication purposes. This gives a central location to monitor the testing process and the engineers can directly talk to each other during the test. Next to that, the central location of the table leaves space on both sides to walk to other areas in the aircraft. The table in the engineering booth has room for up to six engineers. In Figure 13.1 places for four engineers have been drawn, with a total of ten monitors for verification. The two remaining seats are optional. Also the option is given to the customer to mount additional screens above those already installed for additional processing. An example of how the engineering booth could look like with all these monitors installed is represented in Figure 13.2.

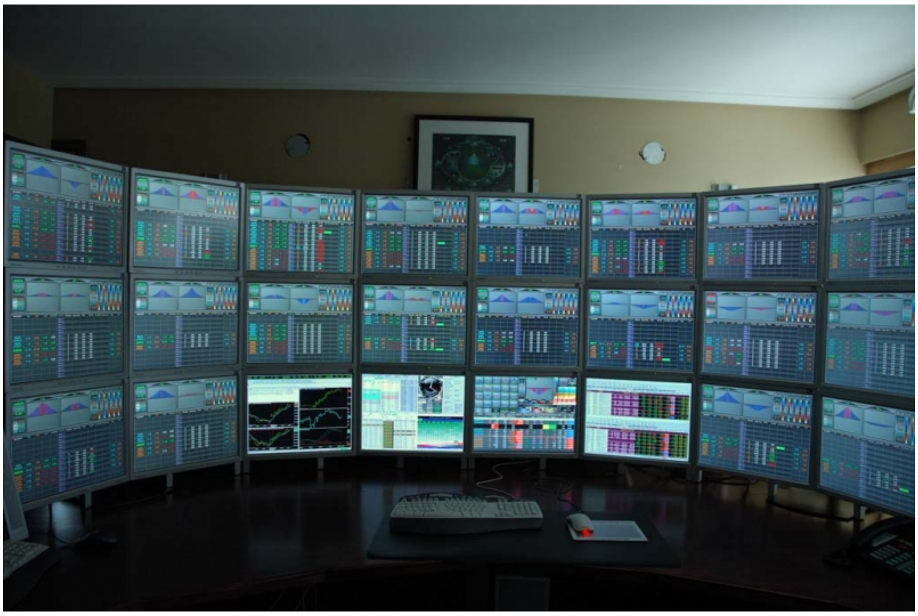


Figure 13.2: Example of monitor wall for test engineers [22]

In the booth two central points will be created were the most of the cables will be lead up to the table. These two locations can be seen in Figure 13.1 as the two circles. From there all the wiring for monitors, keyboards mouses and other input devices will be spread over the table.

Because the aircraft accommodates measurement systems in a modular fashion, multiple screens are accommodated for. Each system can then be viewed separately. This is an advantage as future measurement systems can be monitored easily. The amount of data these systems generate can be large, especially when measurements are done over an entire wing. The aircraft will give the possibility to check all the data generated by these systems, hence the large amount of screens. Next to that, if the client wants to be present during tests, extra screens and input devices should be added. This way the client can verify the generated results independently.

The above option will be complemented by the use of special software developed for this particular aircraft. This can be used to easily check all the systems at the same time on the same screen, using a smart representation. This extra feature will ensure that not every separate system needs a separate screen. For the development of software the time, money and visual space required should be taken into account.

The engineering booth has a total length of 4 meters and is confined by the cockpit pressure bulk and a wall to be installed just before the leading edge root connection of the wing. The purpose of this wall is to keep noise and interference out of the engineering booth and should therefore be designed accordingly. This way, the test engineers can communicate with each other, the pilots and the clients without problems.

The floor of the engineering booth is about 10 cm higher than the actual floor of the passenger area of the fuselage. The wiring for the engineering booth with respect to power, measurement data, communication etc. will be placed underneath this floor to create an accessible wiring system. For more information on wiring see Section 13.6.

13.2 Fuel system design

During the conceptual design phase it has been decided to integrate part of the fuel tanks within the cargo hold of the fuselage. This is done in order to facilitate the installation of AFC systems in the wing skin. This section will describe how the integration of the fuel tank within the fuselage is done.

13.2.1 Fuel tank location and sizing

Volume driver

Following from the Class I weight estimation (Section 10.1), it has been found that 18700 kg of fuel needs to be carried onboard to perform the mission. Assuming that Jet-A fuel is used, it follows that the volume needed is 23.1 m^3. In order to limit the amount of required redesign, it would be very convenient to place some of the fuel tanks in the fuselage. Therefore it has been chosen to place the fuselage fuel tanks within the already existing cargo holds. The these are empty anyway, considering the new purpose of the aircraft.

Two possibilities remained to integrate the fuel tanks: either integral or removable. In the first case the fuel is contained by the cargo hold walls. In the second case a tank is fitted within the cargo hold, following the cargo hold contours as closely as possible. As the presence of cargo doors might pose leakage problems and sloshing effects should be kept to a minimum, the latter option is chosen.

Of the total 23.1 m^3 of fuel required, only 4 m^3 needs to be stored in the cargo hold of the fuselage. The rest of fuel is still stored in the existing wing based fuel tank. Only a a small redesign is necessary, which enables the connection and pumping system between both the wing tanks and the fuselage tank. Next to the amount of fuel required to perform the flight, an additional 1 m^3 will be added to be able to power the APU. It needs to be operative for a flight of 3000 km, which comes down to running the APU for approximately 7.5 hours [23].

Weight driver

Next to the amount of space required, there is an additional driver on both the fuel tank and its system. This driver states that the total weight of the fuel and fuel tank should not exceed the weight capacity of the cargo hold, otherwise the structural integrity of the cargo hold cannot be guaranteed. By spreading the fuel tank over the floor of the two cargo floors the fuel load is distributed in such a way that the cargo hold does not have to be reinforced.

The maximum amount of luggage both the cargo holds can accommodate without reinforcements was found to be 4471 kg [24]. The total volume to be stored in the cargo hold, has been estimated to weigh about 4050 kg. This is 5 m^3 of Jet-A fuel, with a density of 810 kg/m^3 [25]. This means that the designed tank should not exceed a total of 421 kg, or the cargo hold should be reinforced.

The first estimation on the dimensions of the fuel tank can be seen in Figure 13.3. The fuel tank is relatively flat and spans most of the cargo hold. The construction is rather simple to keep the weight low. The weight of the fuel tank is estimated to be 190 kg for the entire fuel tank. This is the weight without reinforcements, connections and piping, since time only allowed modelling and design of the outer shell. Taking those into account it is estimated that the weight will at least double to a weight of 380 kg.

The margin left to prevent reinforcements is 41 kg. The weight needs to be monitored closely during the detailed design of the fuel tank. Nevertheless, the preliminary design concludes that there is the possibility to include a fuel tank in the cargo hold without having to reinforce the fuselage.

Lastly, the cargo hold doors need to be redesigned. Unfortunately the fuel tank will block the cargo hold door as these open inwards. The option chosen to still be able to open the door is to decrease the height of the door. By decreasing the height of the door with about 350 mm, the same height as the fuel tank, it will open just above the fuel tank. At the same time the fuel tank will resemble a second floor and is easily accessible for maintenance.

13.2.2 Fuel tank protection

Placing the fuel tanks in the lower fuselage poses a problem in terms of safety. In case of a belly landing or a tail strike the fuel tank might be punctured, causing the fuel to ignite and explode. In order to prepare for such a situation three measures have been taken to prevent the fuel tank from tearing apart. First of all, the lower wall of the fuel tank is reinforced (that is, made thicker) to provide more structural stiffness to the tanks. Second, the tanks are lined with explosion suppressing foam from the inside. Common for these problems, polyurethane is used [26]. This foam will limit the impact of an explosion. As a final measure the space between the cargo hold floor and the lower fuel tank wall will be lined with an energy absorbing foam. As material aluminium foam is chosen due to the fact that it is not flammable. These three measures together should ensure the fuel protection.

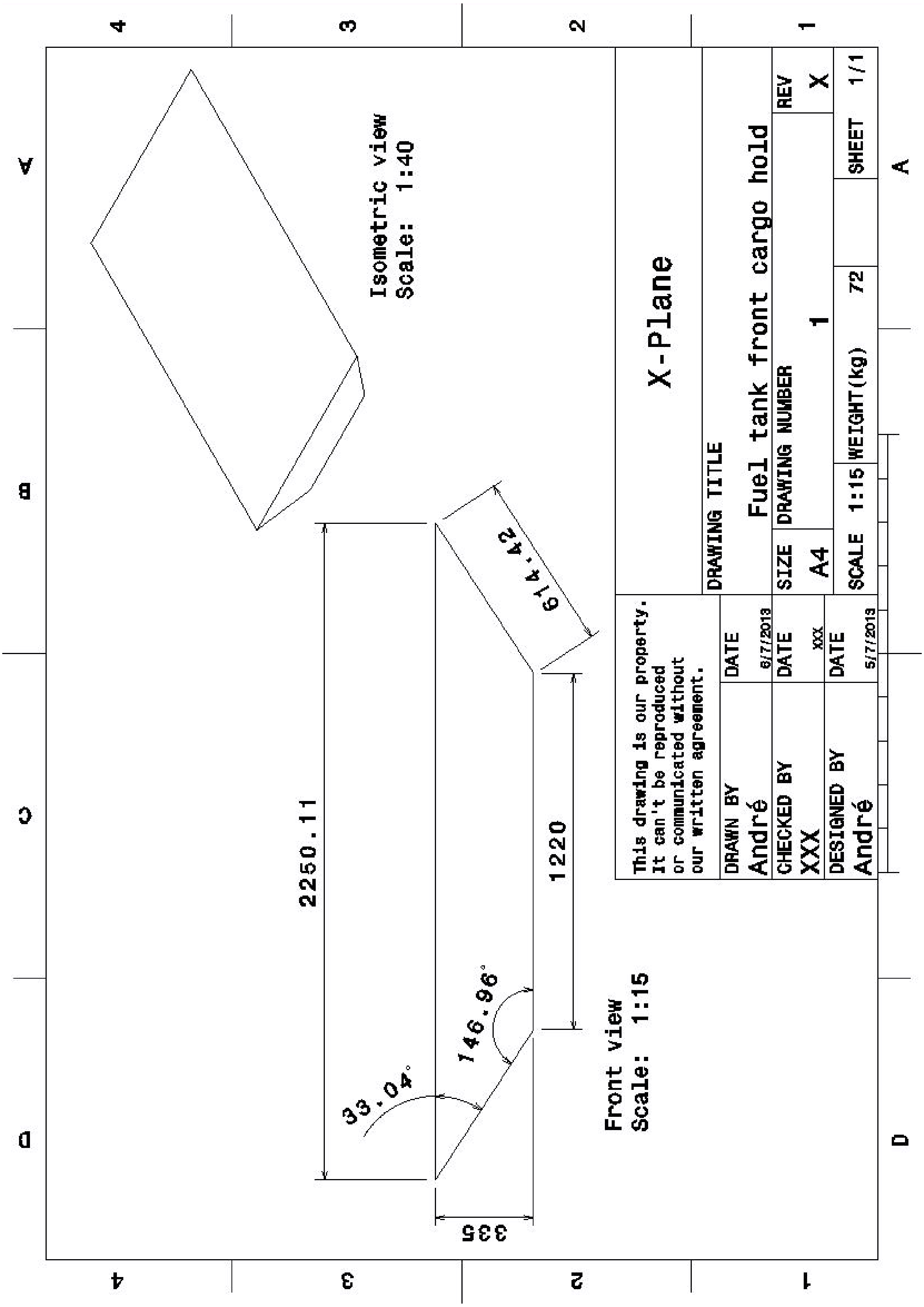


Figure 13.3: Fuel tank dimensions

13.3 Integration of PIV system

In order to measure the velocity profile over the wing, the aircraft will feature a PIV system. This system will consist of the following subsystems:

- A camera
- A laser
- The connection between the laser and the test section
- The lenses setup and test section, the optics
- The cooling system

Light inherently diffuses related to the distance between its origin and perception. The strength of the laser diminishes when longer distances are involved. Also the camera has a limited region in which it can efficiently capture the phenomena. Therefore, it has been chosen to install the PIV related systems as close to the measurement location as possible. This section will now continue with the explanation of all subsystems chosen.

13.3.1 Implementation of the camera

The PIV measurements require a high level of accuracy to produce the required results. Also the amount of data the PIV system generates is much larger than with other flow measurements system. This means that it would be ideal to have a camera with integrated RAM memory. In this way some computing stress is taken from the data processing computers. As has already been indicated in previous reports the design philosophy followed is to limit the amount of redesign to a minimum. Concerning the environmental control system it would therefore be of great ease if the cooling of the camera could already be integrated within the device itself. After consulting with the stakeholders the pco.1600 Charge Coupled Device (CCD) camera system [27] has been chosen, which also achieves all of the above. Due to the fact that this camera has a Universal Serial Bus (USB) 2.0 interface it can easily be connected to the computers located in the engineering booth making it easy to actuate. In order to describe the further integration of the cameras in the other subsytems, four parameters are of main interest. These are the weight, the power usage, the amount of volume the device takes and the costs. For the two cameras combined, these parameters are as given in Table 13.1.

Weight [kg]	**Power usage (maximal) [W]**	**Volume [m^3]**	**Costs [€]**
3.6	80	0.005	Unknown

Table 13.1: Four main parameters for both PIV cameras combined

Concluding from the above it can be seen that due to its low weight, power usage and volume occupied this camera can easily be integrated within the existing fuselage.

Integration for main wings It is desired to perform PIV measurements on both aircraft wings. This means that two cameras will need to be taken onboard to look at each respective wing. These cameras will be placed at one of the rear windows along the wing root fairing to be able to measure along the swept wing. To prevent distorted images, the window through which the camera is pointed has to be without curvature and as thin ass possible. Possible materials and redesign has to be investigated to find the most desired image quality.

The cameras will be mounted on a tripod with a tough spring on it. This will shield the camera from vertical vibrations during use. To protect the cameras from horizontal vibration when it is not in use two clamps will be mounted in the fuselage wall, which restrict the camera from movement. The cameras will get their power from cables coming from the CPP in the reinforcement area of the fuselage. The power for the cameras is generated by the APU, as explained in Section 13.6. More details about the power connection for the wing can be found in Section 11.4.

Integration for test section The test section is mounted upside down. The housing in which the joint is present, also presents room to place the camera, pointing either side. Two orientations are possible for the camera, ensuring it to look both forwards and backwards. The structural stiffness of the pylon should ensure that the camera will be subjected to small vibrations only. The power cables for the camera will come from the CPP. These cables will be combined with the power cabling for the other measurement systems (the modular plugs) going up into the pylon.

13.3.2 Implementation of laser

In order to create a laser sheet for the PIV measurements a laser will be carried onboard. This laser will be placed in a cabinet with warning signs on the outside, due to the danger involved when looking directly into the laser. This cabinet will be connected to two air vents, one will supply cold air to the cabinet and the other will drain hot air. After consulting with the stakeholders the Evergreen 200 laser has been chosen for usage on the aircraft [28]. One of the major advantages of this laser is that it comes with an integrated cooling system. The laser power cables will originate from the CPP. Again the low weight of only 18 kg as well as the low volume it occupies ($0.048m^3$ with integrated cooler) it can be easily integrated within the already existing structure.

13.3.3 Laser - test sections connection

Due to the fact that the laser beam should not diffuse between its generation and the test sections a connection needs to be made between these. This also enables the possibility to use the same laser for both the three desired test sections, two on the main wing and one on the pylon. For this connection optical fibre cables will be used. To keep dispersion to a minimum, multi-mode graded optical fibres will be used. This type of cable has a lining with a decreasing index of refracture, which bends the light in a sinusoidial pattern. A graphical illustration can be found in Figure 13.4.

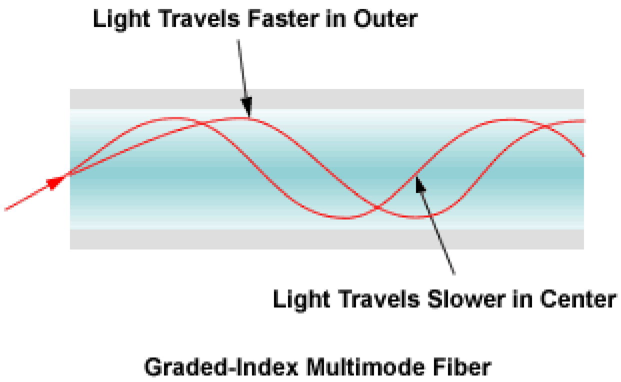


Figure 13.4: Light pattern within a multi mode graded cable

The cables will be connected to the laser located in the laser cabinet. The cables will run into the fuselage floor directly from the cabinet, going to both wings as well as the test section on the pylon. The cables will run up to the respective optic instruments at the testing locations. This setup will exist of several optical devices. The laser diffuses when it is guided along the fibre, a phenomena which cannot be prevented, even with the most efficient cables. To focus the beam, a lens is used. Now, the focussed beam is guided through a lens which transforms it into a thin sheet. This sheet is projected outside, through a thin slit in the upper surface of a skin panel.

13.4 Pressure pump

To enable the provision of differential pressure to AFC systems a decent air compressor is needed on-board. For this purpose it has been decided to use the Ingersoll Rand 50-HP Rotary Screw Total Air System, which is rated at 150 psi (10.35 bar) [29]. It has been decided to place the pump in the rear part of the fuselage as will be indicated in Section 13.6. Its dimensions are 167.64 x 134.62 x 134.62 cm and it costs 19799 US$.

Since it can produce a volume flow rate of 205 ft^3/min (0.0967 m^3/s) at 90 psi (6.21 bar) this is one of the best units available for this purpose and therefore should be able to provide current and future AFC systems with sufficient differential pressure. A major disadvantage is its weight, since 1190 kg may require special reinforcement measures to accommodate the unit. Due to time limits the design of these possible reinforcements is left to the next design iteration.

For the pressure pump and corresponding cooling systems it is necessary to have additional air from outside the cabin be sucked into the cooling and pressure systems. It has been decided to allow for this by placing a NACA duct in the top fuselage skin, directly under the pylon. It has also been decided to place it there since the skin at that location has to be reinforced at this location anyway to withstand the loads introduced by the pylon mounting. The hot air exiting the cooler will be exhausted to the outside environment by placing the exhaust of the coolers onto the exhaust system of the additional generator, which in turn feeds into the APU exhaust.

13.5 Power budget

This power budget is a first estimation for the power requirements of the entire system. To facilitate the power systems in the aircraft and to determine the high end capabilities, the first step was to create a power plan. Such a plan describes what would be connected to each other and what would not be featured on the aircraft. For the first estimation, this plan consists of a list of all systems requiring power. In future design iterations this should be expanded with the connections between the individual systems. The system will be divided in three different sections:

- **Flight power system**
  The flight power system consists of the original system installed in the Boeing 737-500. This system powers the Primary Flight Systems (PFS) in the aircraft and consists of the normal system plus the APU connections. This system will be separated from AFC and measurement systems to ensure safe flight. If the measurement system or the AFC systems encounter a problem the flight power system will not be affected.
- **AFC power system**
  A system to accommodate the power requirements of the AFC system will be incorporated. Due to the requirement of the high variability in the power supply this system a separate system is required.
- **Measurement power system**
  A separate power system for the measurement equipment and data handling will be provided. This system has no connection with the AFC or flight system and will thus not be influenced by power peaks or other interfering electric and magnetic effects. This ensures high quality measurements.

13.5.1 Flight power system

This system is unchanged in the aircraft. The main systems remain the same except that the cabin will be redesigned and thus less lighting and internal environmental control is required. This system will not be elaborated upon further.

13.5.2 AFC power system

The power system for the AFC will not power any other systems. This system is powered by a separate generator which will be capable of fulfilling the following requirements:

- **FR-AR.1-02:** The electrical power system shall be able to provide 10 kW for the AFC systems.
- **SR-AD-01:** The electrical power system shall be able to provide TBD kW peak power for TBD seconds
- **SR-AD-05:** The electrical power system shall provide direct current capabilities up to TBD V
- **SR-AR.1-04:** The electrical power system shall provide variable power to the AFC systems up to 10 kW

- **SR-AD-14:** The electrical power system shall be able to accommodate a waveform transformer
- **SR-AD-15:** The electrical power system shall be able to accommodate a voltage transformer
- **SR-AD-16:** The electrical power system shall be able to accommodate a frequency transformer
- **SR-AD-17:** The electrical cable system shall be able to handle at least voltages up to 15 kV
- **SR-AD-18:** The electrical cable system shall be able to handle at least frequencies up to 15 kHz
- **SR-AD-19:** The electrical cable system shall be able to handle sine, square, triangular, pulse and sawtooth waveforms

These requirements are intrusive on the design of the power system and could be reasons for noise over the power cable. Since all the systems of the aircraft rely on electricity and are influenced by it, it has been decided to separate this system from all other power systems. Unexpected power peaks, strange power waveforms or even other strange phenomena on vital non-AFC systems should be avoided with the separation.

Table 13.2: Power requirements AFC

Actuators	Power requirement	Total power estimated [W]
Vibrating ribbon devices	Unknown	-
Piezoelectric vibrating parts	5 Watt per actuator	-
Pulsed jet	Maximum of 1200 Watt per actuator	5000
Power resonance tube	No electrical power required	-
ZNMF Piezoelectric	0.9 Watt per actuator 21 Watt per actuator	90
ZNMF Electrodynamic	5 Watt per actuator	500
Active dimples	Unknown	-
Suction actuators	No electrical power required	-
SDBD	3,5 Watt per actuator 20 W/linear ft	500
Rotating surfaces	Unknown	-

Looking at the estimated power usages from Table 13.2, it can be concluded that at least 5 kW is required. This is just enough to sustain the most power consuming AFC system. However taking into account the requirement to sustain at least two AFC systems at the same time, a larger amount of power needs to be supplied. Next to that, taking into account possible future systems, the total power delivered by the AFC power system shall be 10 kW.

13.5.3 Measurement power system

Measurement system power is slightly different from the AFC power. This system needs low interference due to the high accuracy requirements from the measurement systems. The power system for the measurement system consists of both the measurement equipment power and the data handling power. Considering this it has the following requirements:

- **FR-TS.1-04:** The electrical power system shall be able to provide 15 kW for the measurement system.
- **FR-TS.1-05:** The electrical power system shall be able to provide 5 kW for all on-board aircraft segments of the data handling system.

The total power requirement for the measurement power system has been estimated at 15.8 kW, following from Table 13.3. This however is a power system that has to be monitored closely due to the difference in power usage of different PIV laser models considered.

Table 13.3: Power requirements measurement systems

Type	Power requirement [W]		#	Power total [W]
Pitot tube	450	max per tube	2	900
GPS	10.5	max per reciever	1	10.5
Accelerometers	0.012	max per piece	6	0.072
Gyroscopes	0.105	max per piece	3	0.315
Flow-vanes	0.006	average per piece	2	0.012
Data handling	2500	estimated total	1	2500
PIV camera	220	max per piece	4	880
PIV lasers	5500	max per piece, Nano TRL 400-20 PIV	2	11000
Thermal anemometers	0.045	average per piece	300	13.5
Other	500	max total	1	500
Total estimated				15804.4

13.6 Electrical system

This section will focus on the electrical power generation and regulation on board the aircraft. The first two subsections will discuss how power is generated and how electric wiring is installed in general. The following subsections will go into more depth and focus on each electric circuit in itself, discussing the measurement systems, the AFC systems and the data handling systems.

13.6.1 Power generation

For the testing equipment and AFC systems to be able to operate without disturbing each other, it is necessary to have two independent sources of electrical power. This is needed to make sure that activating or having activated the AFC systems will not influence the obtained test data. It has been decided to use the already present 90 kW Auxiliary Power Unit (APU) for the provision of electrical power to the primary equipment. This includes the measurement systems, the data handling and the data storage systems. In emergency situations (one engine inoperative, gliding flight) the APU will provide emergency power to the primary flight controls. As an additional power source a 10 kW generator will be installed to provide power to the AFC systems. It has been decided to use the Honeywell 6033, Air cooled 15kW Kerosene Generator for this purpose, which will cost $ 3699 [30].

In addition to these power generation systems, other systems are required to transform the generated power from one power form to another. This includes, but does not limit itself, to changing frequency of A/C, an AC/DC converter and a transformer. To a certain extent these will be provided on-board by the aircraft as will be discussed in Subsection 13.6.4. Any further wishes could be provided for by the client and done in close collaboration with the engineering team.

13.6.2 Electrical wiring installation

To facilitate the different types wiring two main channels are designed. The first one is located in the middle of the fuselage under the fuselage floor. This is where the power cables will run. The second channel will be in the left corner of the fuselage. This will be used for data handling cables.

The power cables will run through the middle of the fuselage to the middle of the wing root. At the middle of the wing root, the power cables are split in the Central Power Point (CPP) to different areas within the aircraft. The power cable channel will itself consists of two sub-channels: one for the AFC systems and one for the measurement systems for reasons as discussed before. Insulation in between prevents interference.

The power cable channel shall be about 10 cm high and as wide as required. It shall be designed such that an engineer can walk over the channel, as it will be the main path of walking. Also the top panels of the channel should be removable for easy access to the power cables in case of malfunctioning and maintenance.

The data handling cables will go through the left side of the fuselage to separate them as much as possible from the power cables and it will give easy connection to the processing system. Due to multiplexing the amount of cables is reduced, but due to the large amount of measurement systems the amount of cables will still be substantial.

The final area for which wiring is important is the engineering booth. This area is a bit more complicated due to the possibility of changes, constant presence of engineers and the spreading of equipment. To facilitate all of this a double floor system will be incorporated. A new floor will be created about 10 cm above the main floor. Below the new floor all the cables will be placed. By removing floor panels, the cables will be accessible and changes can be made easily.

The general fuselage lay-out with the wiring and connections between the installed systems can be found in Figure 13.5 and Figure 13.6.

13.6.3 Measurement systems electrical circuit

This section will elaborate in more detail on the electrical circuit concerning, the measurement systems. The electrical block diagram for the measurement systems can be found in Figure 13.7.

The measurement systems are powered by the APU, presented in Figure 13.7 as an alternating current source. Right after the source there is a master switch between the PFS and the measurement systems. This means that the APU either powers the PFS or the measurement systems. In case of an emergency this switch ensures that all measurement systems are shut off, directing all power to the PFS. Following this switch is a master fuse preventing the entire power generation circuit from overloading. Next a power regulator controlled from the engineering booth will regulate the power given to the measurement systems, since the maximal power the APU can give of around 90 kW is too much to be fed directly into the system [16]. After the power regulator, a transformer will transform the input voltage to 230 V, such that the power going to the PIV installations and the modular measurement systems plugs is the same as in any ordinary household: 230 VAC at 50 Hz. This transformer also separates the power generation circuit from the circuit to which the measurement systems are connected.

After the current has passed the transformer it will run through an attenuator. This attenuator will damp out the noise induced in the power signal, giving high quality signals to the measurement systems. Following this, the first junction splits cables going to the cooling and data handling systems (which are then also grounded) and cables to the CPP.

The CPP serves the function of splitting all the cabling in the proper direction. Part of the cabling will run to the two wings (only one wing has been drawn in Figure 13.7), a part will run to the pylon and a final part will remain in the fuselage. The part to the wing and test section will provide the modular power sockets and the power for the pylon mounted PIV camera. The other part of the cables will power the fuselage mounted PIV camera and the laser. All measurement switches are protected by fuses and switches. Fuses are chosen rather than circuit breakers because fuses tend to burn through faster which provides more protection for the sensitive equipment such as the PIV camera and laser. The switches are controlled from the engineering booth and enable selection of the desired power sockets.

13.6.4 AFC electrical circuit

The AFC electrical circuit handles all the power requirements for the AFC systems on the wing and pylon. The AFC electrical installation exists of four parts which all can be seen in Figure 13.8:

- Power supply
- Central power point
- Wing AFC power
- Pylon AFC power

Power supply

The first element is the power supply. The power supply main objective is to generate the required amount of power, in the right frequency, at the right voltage and with the right waveform. This is done through multiple systems: a variable frequency changer, a variable transformer and a waveform changer.

Both the frequency changer and waveform changer can be circumvented, since it is not always required to change the waveform or frequency during testing. In that case the standard sinusoid waveform and the frequency range of the power generator is sufficient.

Next to that, the power is split over two circuits. Both circuits can handle all low voltage requirements, but only the dashed version in Figure 13.8 can handle the high voltage requirements for plasma actuators.

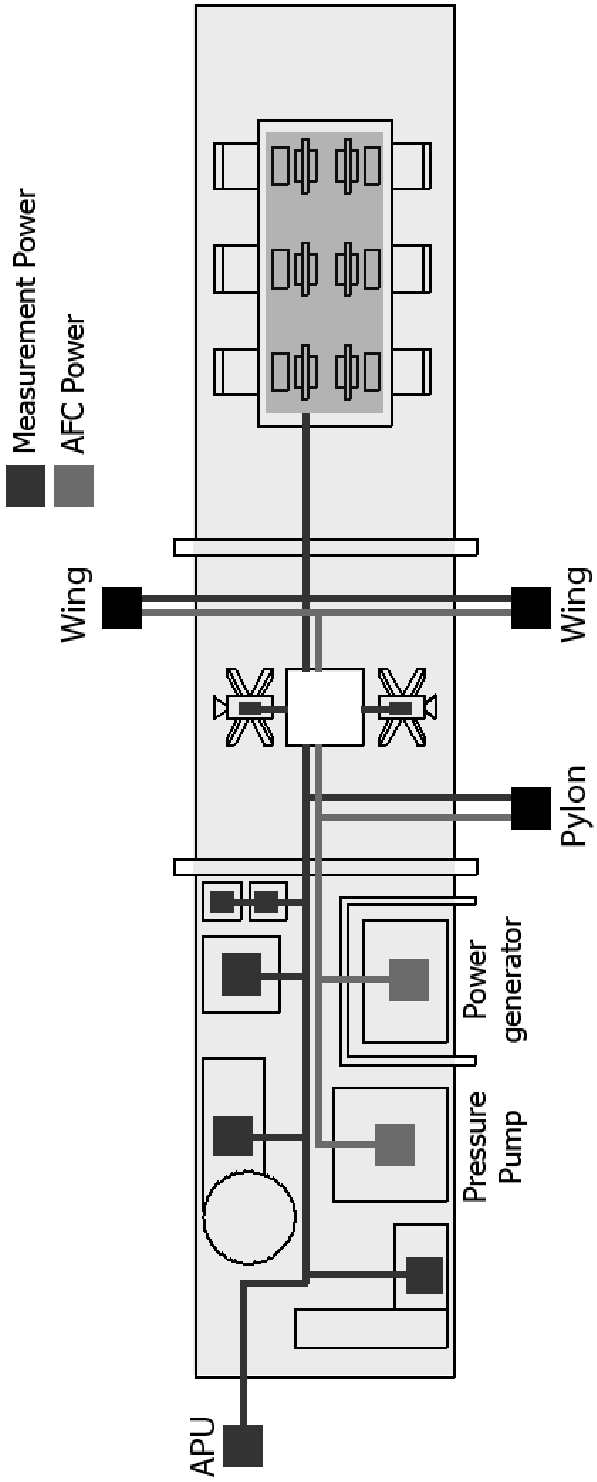


Figure 13.5: The fuselage lay-out with the location of the power wiring and installed equipment

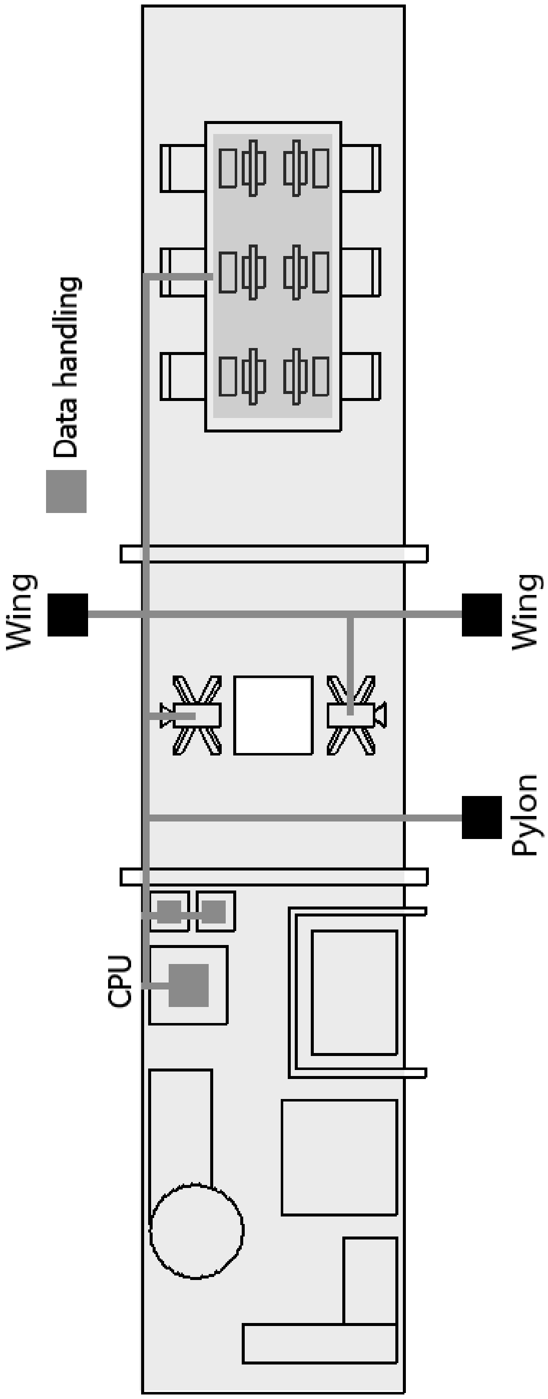


Figure 13.6: The fuselage lay-out with the location of the data handling wiring and installed equipment

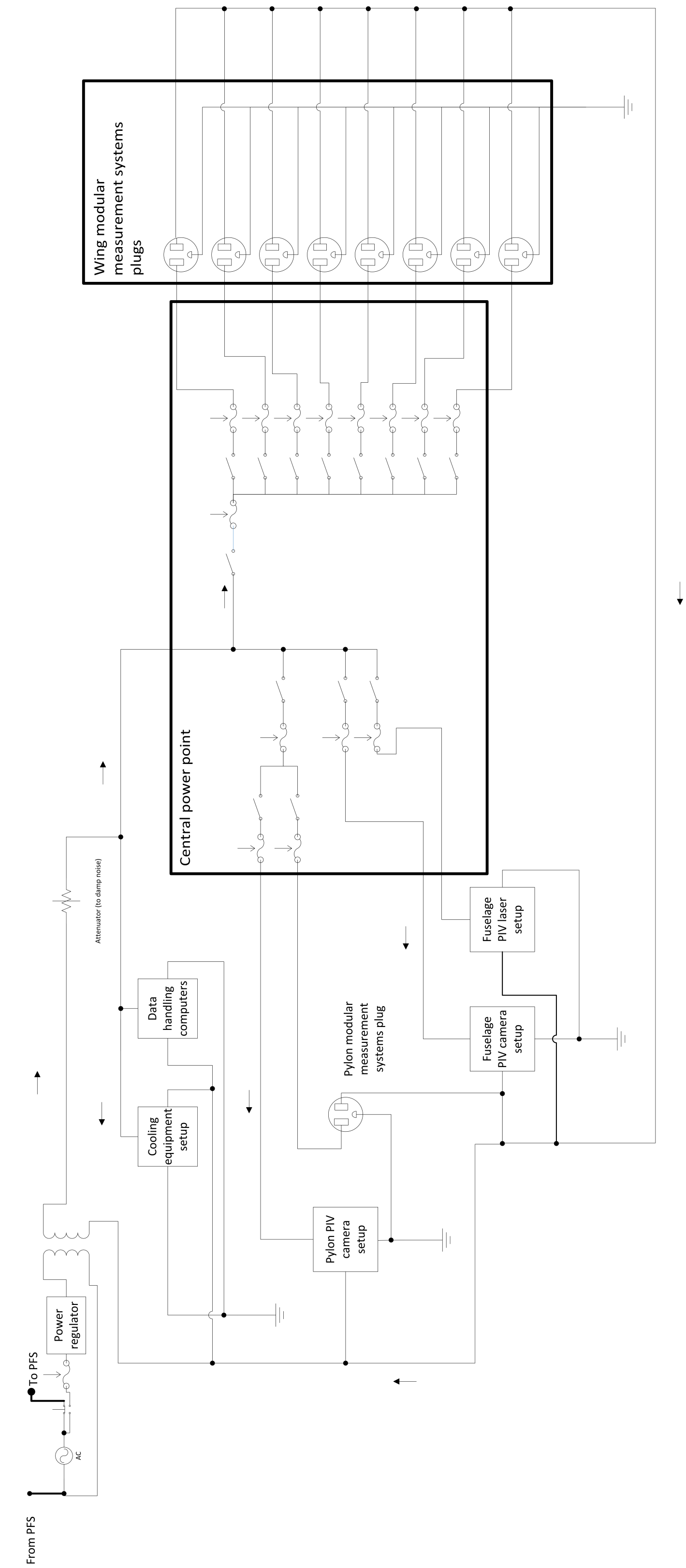


Figure 13.7: Electrical block diagram for the measurement systems

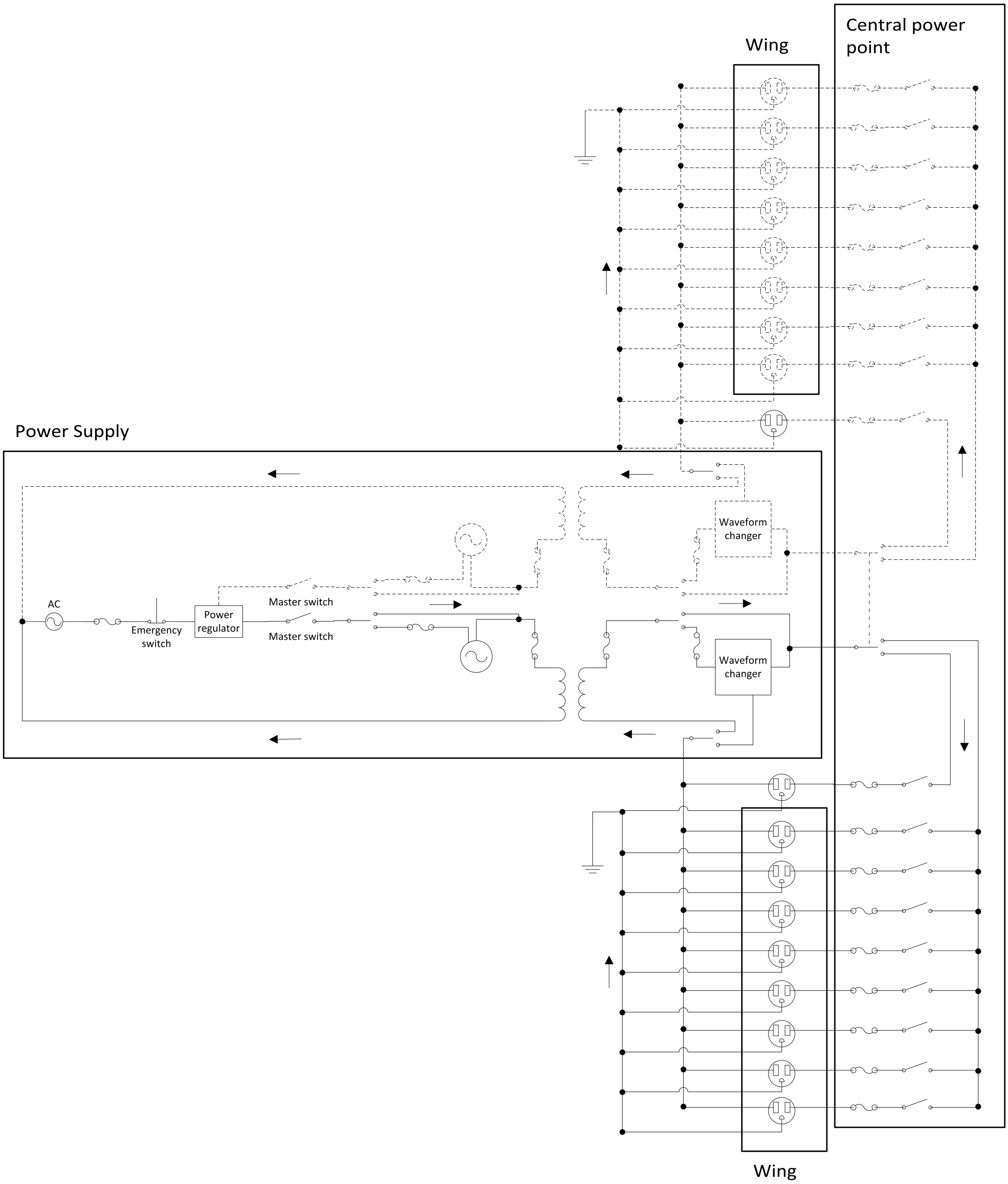


Figure 13.8: Electrical block diagram of the AFC power system

With this system, two different AFC systems can be connected to the power supply and be actuated at the same time, even though the systems have different power requirements.

Central power point
The first area after the power supply is the CPP. This is not necessarily a system, but mainly a power hub where all the fuses and switches are located. All the switches in the CPP are actuated from the engineering booth. This way the engineers can choose which parts of the wing they are going to test. Of course taking into account that either the wing or the pylon can be chosen. The pylon and wing cannot be active at the same time.

The chosen fuses are variable since the power requirements per actuator can differ. Fuses are chosen instead of circuit breakers, as the type of fuse can be changed relatively simple compared to changing a circuit breaker. Also fuses are available in more variety and can be customized more easily.

13.6.5 Electrical grounding

To provide a safe electrical system, it is necessary to have sufficient grounding. This is needed to assure the systems will never be overloaded, which would destroy it. Also, this way it is easier to cut off the power with an overload kill system, in case of a short-circuit.

It has been decided to install, in accordance with the dual electrical system, a dual ground. It has to be made sure that these two grounds can not contact each other. For this reason it has been decided to keep the main ground intact. Also, since it is already part of the APU-circuit, the existing ground part of the APU circuit is kept.

A new ground must be created for the AFC-circuit.A large sheet of copper is placed in the core of the floor separating the cargo hold and the cabin section. This sheet will function as the main ground for the AFC system. Further there will also be a fin on the belly of the fuselage to allow discharging of the plate. When the aircraft lands, a grounded plug will have to be placed onto this fin, to allow full discharging of the AFC ground.

13.7 Data handling diagram

The data handling diagram, shown in Figure 13.9, gives the structure on how data flows throughout the aircraft systems. The build up of the data flow starts with the sensors and ends with the client, however a lot of control and calibration is required to make sure good quality work is generated.

Figure 13.9 is a schematic representation of how data handling happens. The first step is the sensors and their multiplexing unit. Every module has a a set of sensors which go through a multiplexing unit to reduce cabling and go to the processor. This is not only for the wing and pylon, but also for extra sensors added in the fuselage. Take into account that this does not represent the total amount of sensors or multiplexing units. The total amount could be more, but since those systems are relatively simple not all of them have to be drawn.

After the multiplexing comes the main preprocessing unit. This unit converts the data from raw data into a format which can be stored and used for further analysis. This is done for all the data generated by the sensors and power measurement equipment. The PIV system, flight systems and the power supply do not have this preprocessing, since it is already done internally by the device itself or by another processor. During preprocessing a time stamp is added to all data. To efficiently store data and retrieve it for analysis later on, it is important to know at which time what data is generated.

After a time stamp has been added, all the data should be synchronised. At this point, power data, PIV data and flight data is added to the stream of information. It is all synchronised according to their stamps.

The synchronised data is now ready for storage and further processing for verification. For analysis purposees, the raw data, obtained after synchronisation, is stored directly. This way no information will be lost. Next to that, a first graphical analysis is done such that the engineers in the engineering booth can do some preliminary verification. This way large errors can be spotted easily and corrections can be made during testing.

The corrections can be made by either recalibrating the equipment or commenting on the results such that the corrections can be done efficiently afterwards. Although calibration during flight is preferred for

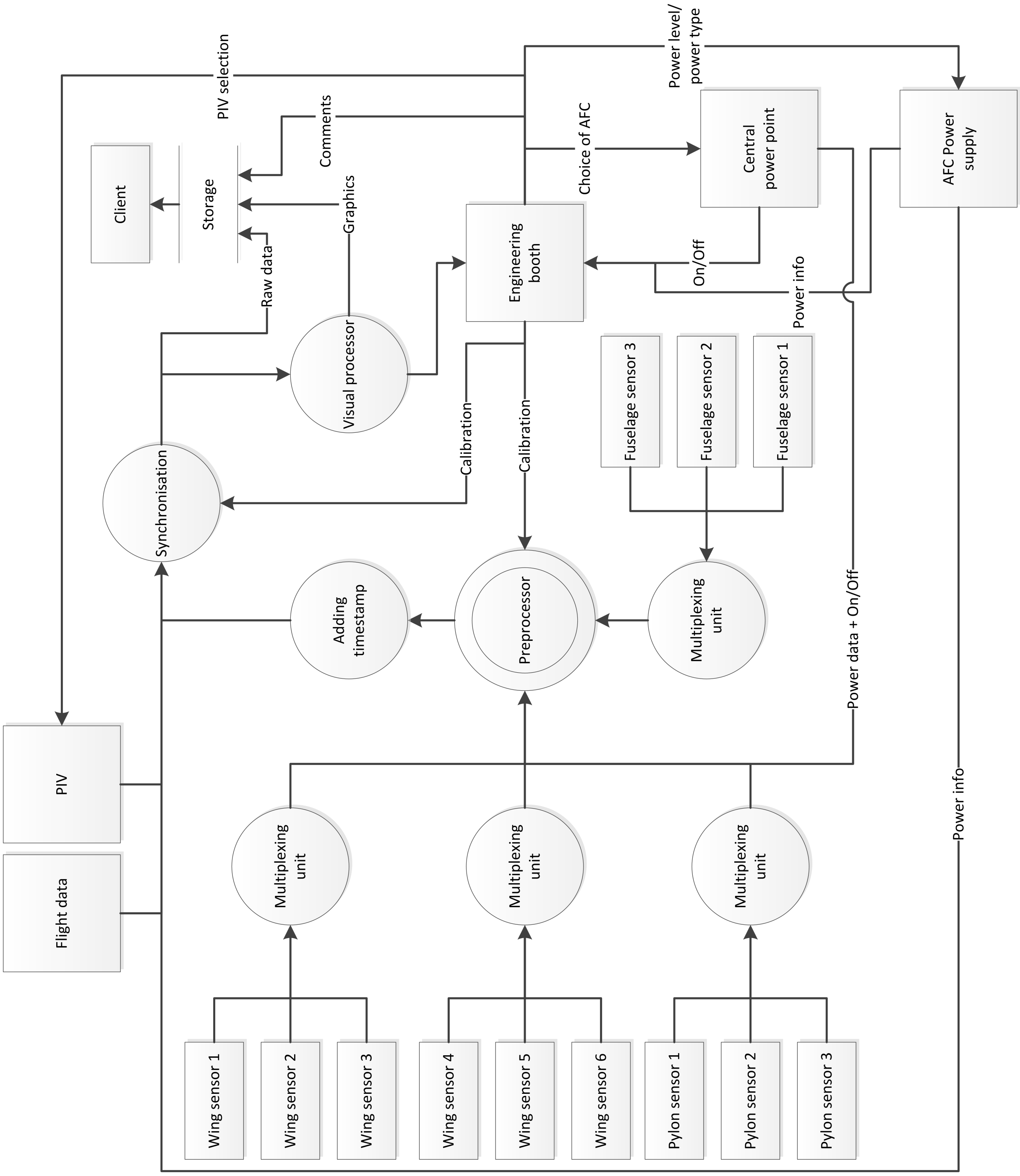


Figure 13.9: Data handling diagram for the aircraft

better data it is difficult to reconfigure testing equipment outside of a specified facility. Therefore, the most logical option is either commenting on the results, ignoring that part of the results or stopping the test to recalibrate on the ground.

The engineering booth is the main control area of the system. Meaning that it can choose the AFC systems to activate, but can also determine which measurement systems are turned on or off. This is done via a connection to the CPP, which has some control software. The control has to be actuated, even when the entire data handling system is off line, meaning that a separated control is going back from the CPP to the engineering booth for feedback and control. By doing so, the engineers are able to determine which systems are on line and off line, without the use of the data handling system. The same applies for the power supply systems, which also need to be checked without the help of the data handling systems.

13.8 Recommendations

For the interior design, there are three elements that need a lot of attention in the next phase of the design: Power system, the fuel tank and pylon integration. Throughout this chapter, the Data Handling (DH), power and fuel systems have been estimated. The estimations all have been done based on information of what is available on the market, but not on actual operating systems.

The power system is a very complex system since it is constrained by a lot of different requirements. It has been investigated in the power budget how much power is actually required, however no definitive answer on the amount of equipment supported on board has been decided. Therefore, the power budget will most probably change drastically and will influence the power requirements. Although an initial investigation was done and the fuselage sensors are for the most part decided, the wing Measurement Systems (MS) and DH systems are still open. So the first step for the power budget is to determine the total amount of supported test systems on the aircraft.

For AFC systems the power budget was estimated and a total amount of power was given for the AFC systems. The problem with the AFC systems are the amount of different power requirements. Although the electrical block diagrams take into account the possibilities, an extra investigation has to be done for the actual way of integration and space required to be able to create module power systems.

Initial investigation shows that it is feasible to fit the fuel tank in the cargo hold. However there are two elements that need to be investigated. First is the continuation of the initial design process, to obtain better weight information. Second, the Rogerson-Installed Auxiliary fuel tank [12] can be investigated. Both options are possibilities, however the extra fuel tank can create certain problems with stability and control and the requirement to be able to influence the stability of the aircraft. So a further investigation has to be done to determine the feasibility of both options.

Pylon integration has not been taken into account yet for fuselage design. Although the pylon group made some initial plans, some extra communication is required between fuselage and pylon design to create a good system. Especially the CPP and the PIV system are influenced by the integration and to create an efficient system, the communication and co-operation between the fuselage and pylon design teams is essential.

14
Aerodynamic characteristics

During the preliminary design phase an estimation of the aircraft's aerodynamic characteristics has to be made, in order to size the aircraft. Namely, the aircraft has to be as stable and controllable as desired. The sizing is an iterative process for which a standard method is made to estimate the aircraft's aerodynamic characteristics. This method is described in the Section 14.1 and the found results in Section 14.2. Section 14.3 will apply the aerodynamic results, found in Section 14.2, to determine the stability of the aircraft.

14.1 Method

Due to the limited availability of complex, but accurate tools, to estimate the aerodynamic characteristics, they were estimated using two methods. The lift and moment coefficient were determined by XFLR 5 and the drag was estimated using statistical relations. XFLR5 is an analysis tool for airfoils, wings and planes operating at low Reynolds numbers.

Before XFLR 5 could be used, the wing planform needed to be determined. The wing planform will be presented first in this section. The method to determine lift and moment characteristics with XFLR 5 is described second. Last, the drag characteristics estimation method will be explained.

14.1.1 Planform

In order to analyse the wing of the B737-500, the wing planform was modelled. However, the exact geometry of the wing was not available, which resulted in a model which is partly based on estimations. The planform geometry was based on the geometric parameters presented in Table 14.1 [16], and on estimations which are based on the geometry of the B737-500 wing. The wing planform used in XFLR 5 is presented in Figure 14.1. The wing consist of the four airfoils, according to the data base of the University of Illinois at Urbana Champaign (UIUC) [31], which are presented in Figure 14.2. The spanwise positions of these airfoils are based on the analysis of different views of the B737-500 wing.

Table 14.1: Wing planform parameters, including wing span b, wing surface S, aspect ratio AR, chord c, dihedral Γ, and sweep Λ. [16]

b [m]	S [m^2]	AR	c$_{root}$ [m]	c$_{tip}$ [m]	Γ [deg]	Λ [deg]
28.88	105.4	9.16	7.32	1.62	6	25

The wing of the B737-500 has multiple high lift devices installed, which are listed below. However, these high lift devices are not modelled in XFLR 5, since they are not deployed in cruise flight. Another reason they are not modelled is due to the lack of information on the exact geometry of the high lift devices. What is known are $\frac{flap\ span}{wing\ span} = 0.72$ $\frac{flap\ area}{wing\ area} = 0.29$, and unfortunately that is not enough for accurate modelling.

High lift devices which are normally installed, are the following.

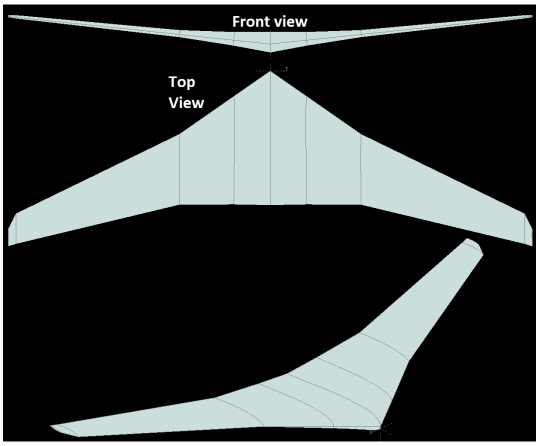


Figure 14.1: Front view, top view and isometric view of the wing planform

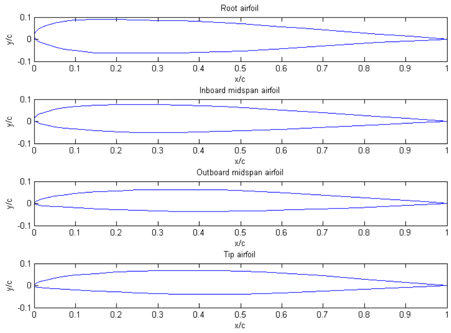


Figure 14.2: B737-500 airfoils

- Leading edge devices
  - 4 Krueger flaps, inboard of the engines
  - 6 slats, outboard of the engines
- Trailing edge devices
  - triple slotted flaps

14.1.2 Lift and moment coefficient

After the model was build in XFLR 5, the aerodynamic characteristics of the wing were analysed. 4922 3D panels were fitted to the wing, which represent the wing when it is analysed using the panel method. The panel method can be based on one or more fundamental solutions to the Prandtl-Glaubert or Laplace's equation. The basic idea is to use known flow solutions and add these together. For instance, using uniform flow with point sources. When the panels are determined, the flow at each panel is determined, associated with the source, doublet, or vortex of another panel. Afterwards, boundary conditions are enforced, with which the singularity strength of each panel is determined.

There are a few drawbacks when using XFLR 5 for the analysis of the aerodynamic characteristics of the B737-500. The first drawback is that XFLR 5 does not model the entire aircraft. XFLR 5 analyses only one wing at a time. This analysis does not consider fuselage interference, interaction with the engines, or any forces and moments generated by the other parts of the aircraft. Therefore, only the lift coefficient and the moment coefficient obtained from XFLR 5 will be used. The lift generated by the part of the wing that is normally within the fuselage accounts for the lift that is normally generated by the fuselage itself [18]. The moment coefficient, generated by XFLR 5, represents only the wing. However, this moment coefficient can be used for the stability analysis of the aircraft.

The second drawback is that XFLR 5 does not simulate situations were the Reynolds number exceeds 10 million. The cruise Reynolds number will be approximately 40 million over the wing. However, the Reynolds number, especially for the flow over the wing tips, is lower than the actual cruise Reynolds number. As the Reynolds number increases, the ratio between inertial forces to viscous forces increases. Hence, the lift coefficient calculated by XFLR 5 will be an underestimation of the actual lift coefficient.

14.1.3 Drag coefficient

To provide an initial estimation of the drag, the technique proposed by Sadraey in [32] was used. This was done, rather than taking the value from XFLR5, since this enables other contributing factors such as the fuselage, empennage and engines to be taken into account. The method has a reported accuracy of 95% for subsonic aircraft. The lift coefficient (C_L) will be taken from the XFLR5 data in order to determine the induced drag. The zero-lift drag coefficient (C_{D0}) will be the sum of the coefficients of the separate parts. The total drag is the addition of induced and zero-lift drag. The C_{D0} of the main wing, fuselage, empennage, engine & nacelles, test wing and pylon mountings is discussed next.

Main wing
To determine the Zero-lift drag coefficient for the wing (C_{D0_w}), Equation 14.1 is used. Below, each variable is explained.

$$C_{D0_w} = C_{f_w} f_{tc_w} f_M \left(\frac{S_{wet_w}}{S}\right)\left(\frac{C_{d_{min_w}}}{0.004}\right)^{0.4} \tag{14.1}$$

The first parameter is the Skin friction drag coefficient of the wing (C_{f_w}) and is a non-dimensional number. In general, for laminar flow, the Skin friction drag coefficient (C_f) is given in Equation 14.2 and for turbulent flow in Equation 14.3.

$$C_f = \frac{1.327}{\sqrt{Re}} \tag{14.2}$$

$$C_f = \frac{0.455}{\left[log_{10}(Re)\right]^{2.58}} \tag{14.3}$$

A combination of these two is made in the case that both types are encountered in the flow over the surface. General values for the portion of laminar flow are around 5% [32]. For these equations, the Reynolds number (Re) is required. To calculate Re Equation 14.4 can be used

$$Re = \frac{\rho V l}{\mu} \tag{14.4}$$

With Atmospheric density (ρ), True airspeed (V), Length (l) and Dynamic air viscosity (μ) of air. The length l is the component in the flight direction, in this case the Mean Aerodynamic Chord (MAC). The flight altitude determines the ρ and μ, since they are dependent on temperature.

The second parameter is f_{tc_w} and is computed using Equation 14.5. It is mainly a function of the maximal Thickness-to-chord ratio ($\frac{t}{c}$). However since the wing consists of several airfoils along the span, an average value for the maximal Thickness-to-chord ratio ($\frac{t}{c}$) was taken.

$$f_{tc} = 1 + 2.7\left(\frac{t}{c}\right)_{max} + 100\left(\frac{t}{c}\right)^4_{max} \tag{14.5}$$

The third variable, Mach factor (f_M), is calculated according to Equation 14.6 and is only a function of the Mach number of the aircraft.

$$f_M = 1 - 0.08M^{1.45} \tag{14.6}$$

The next to last parameter is Wetted area (S_{wet_w}), or actually the ratio between S_{wet_w} and surface area (S). To determine the ratio between areas, the S_{wet_w} needs to be calculated, since S is already known. This was done for the wing using Equation 14.7

$$S_{wet_w} = 2\left(S - c_r d_f\right)\left(1 + 0.25\left(\frac{t}{c}\right)_r \frac{1+\tau\lambda}{1+\lambda}\right) \tag{14.7}$$

Equation 14.7 is a function of S, Fuselage diameter (d_f), Taper ratio (λ). Subscript 'r' denotes the values are taken at the root, and $\tau = \frac{t_r/c_r}{t_t/c_t} = 1.42$ for the main wing.

Finally, $C_{d_{min_w}}$ is the minimum drag coefficient of the cross section of the wing of tail. It is extracted from the drag polar of the airfoil using XFLR5.

Fuselage
The Zero-lift drag coefficient for the fuselage (C_{D0_f} is given by the following equation:

$$C_{D0_f} = C_{f_f} f_{ld} f_M \left(\frac{S_{wet_f}}{S}\right) \tag{14.8}$$

The Skin friction drag coefficient of the fuselage (C_{f_f}) is calculated with the specific Reynolds number, with the fuselage length as l in Equation 14.4. Instead of the maximum thickness, a parameter based on the fuselage fineness ratio is used, which is the division of the fuselage length l_f by the fuselage diameter d_f. The coefficient is defined as follows:

$$f_{ld} = 1 + \frac{60}{\left(l_f/d_f\right)^3} + 0.0025\left(\frac{l_f}{d_f}\right) \tag{14.9}$$

The Mach factor (f_M) is the same as previously determined. The wetted area for the fuselage is calculated as follows:

$$S_{wet_f} = \pi d_f l_f \left(1 - \frac{2}{\frac{l_f}{d_f}}\right)^{\frac{2}{3}} \left(1 + \frac{1}{\left(\frac{l_f}{d_f}\right)^2}\right) \tag{14.10}$$

Empennage
For the drag estimation of the empennage, the horizontal and vertical stabilizer were considered separately. The equations used are similar to the equations which were used to calculate the C_{D0_w}. However, to compute the wetted area of the main wing, an equation can be used. Such equations are not available to compute the S_{wet_w} of the control surfaces. Therefore, they are assumed to be twice as large as their reference areas. The thickness of the tail wings is based on statistics.

Engine & nacelles
For the estimation of the drag coefficient contribution to the total value, Equation 14.11 was used.

$$C_{D0_e} = n_e C_{f_e} f_{ld_e} f_M \left(\frac{S_{wet_e}}{S}\right) \tag{14.11}$$

The Number of engines (n_e) is taken into account, along with again a factor for the skin friction, fineness ratio, Mach term and specific wetted area. The Factor of fineness ratio of the engine (f_{ld_e}) makes use of the following statistical relation, which is used for blunt bodies:

$$f_{ld_e} = 1 + \frac{2.2}{\left(l_e/d_e\right)^{1.5}} + \frac{3.8}{\left(l_e/d_e\right)^3} \tag{14.12}$$

The wetted area of the engine consists out of the wetted areas of the cowlings of the fan and gas generator and of the spike at the end. The specific formulas can be found in [32] but will not be repeated here for brevity.

Test wing & pylon mounting
For the test section, the procedure is identical to Equation 14.1, except that the fuselage contribution in the calculation of the wetted area is neglected, since this obviously has no influence on the test wing.

The pylon mountings are considered to be struts having airfoil sections, which means the $C_D 0_i$ equals 0.1 [32] in Equation 14.13.

$$C_{D0_s} = \sum_{i=1}^{n} C_{D0_i} \left(\frac{S_{wet_s}}{S}\right) \tag{14.13}$$

Again, also the wetted area of the strut needs to be calculated. The total component of the pylon mounting is the sum of all values for the different struts.

Total zero lift drag
To overall C_{D0} is the sum of the C_{D0} of the separate parts. However, these values do not account for other drag sources like pitot tubes, antennas, rivets and screws, leakage, etc. To account for these factors, a correction factor is applied of 1.1 is applied [32]. This value of 1.1 is valid for passengers aircraft, which is what this aircraft was originally.

Lift induced drag
The drag induced by lift is calculated with the lift coefficient determined by XFLR 5. The so called induced drag coefficient (C_{D_i}), seen in Equation 14.14 also depends on Oswald's efficiency factor (e) and the aspect ratio (AR). e is a correction factor that represents the change in drag with lift of a 3D wing, which is described by Equation 14.15 [33].

$$C_{D_i} = \frac{C_L}{A * \pi * e} \tag{14.14}$$

$$e = \frac{1}{(\pi * A * k) + \frac{1}{u * s}} \tag{14.15}$$

'k' is the viscous drag due to the lift factor, and is described in Equation 14.16, where Λ represents the sweep angle. The variable u is the planform efficiency factor which usually has a value of 0.99. 's' is the induced drag factor due to the effect of the on the spanwise lift distribution, and 's' is presented in Equation 14.17.

$$k = (0.38 + 57e - 6 * \Lambda^2) * C_{D_0} \tag{14.16}$$

$$s = 1 - 1.556 * \frac{d_f}{b}^2 \tag{14.17}$$

14.2 Results

The aerodynamic characteristics of the aircraft, estimated using the method described in Section 14.1, are presented in this section. Three design analyses have been done on the aerodynamic characteristics of the aircraft. The first and second analysis focussed on non testing flight, while the third analysis is concerned with testing flight using the test wing.

14.2.1 Non testing flight

During the first analysis, the pylon and the pylon mounted test wing were not considered when estimations of the aerodynamic characteristics were done. The parameters in Table 14.2 and Table 14.3 were used to represent the aircraft during the analysis. An altitude of 3000 m and a Mach number of 0.5 were used during the analysis.

Table 14.2: Lifting surface parameters [16]

	wing span (b) [m]	S [m^2]	AR [-]	Taper ratio [-]	MAC [m]	$\frac{t}{c}$ [%]	C$_{d_{min}}$ [-]
Wing	28.9	105	9.16	0.240	4.34	11.3	0.00513
Horizontal tail	12.7	31.4	4.04	0.260	2.75	10	0.00495
Vertical tail	6.15	23.1	1.81	0.310	4.11	10	0.00490

Table 14.3: Aircraft parameters [16]

	l [m]	Height [m]	Width [m]	Diameter [m]	Slenderness ratio [-]
Fuselage	29.8	4.01	3.76	3.89	8.02
Engine	5.34	2	2	2	2.67

The analysis of the wing, specified in Section 14.1, resulted in the lift and moment coefficient of the wing. The lift and moment coefficient, determined with XFLR 5, are shown in Figure 14.3 and Figure 14.4 respectively. It can be noted that analysis stops at an angle of attack of 10 degrees, since XFLR 5 did not converge for higher angles of attack, meaning XFLR 5 could not provide an answer. The drag coefficient of the entire aircraft is shown in Figure 14.5, where the C_{D_i} and C_{D0} are shown as well. The lift drag polar of the aircraft is shown in Figure 14.6.

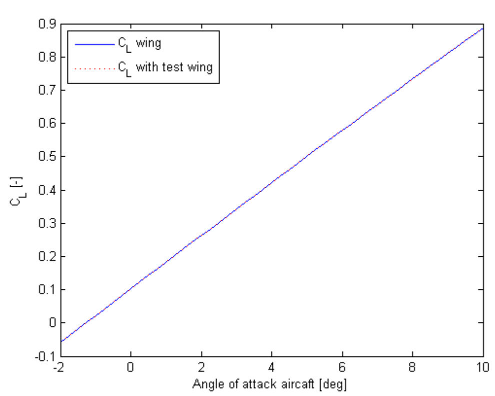


Figure 14.3: Lift coefficient B737-500

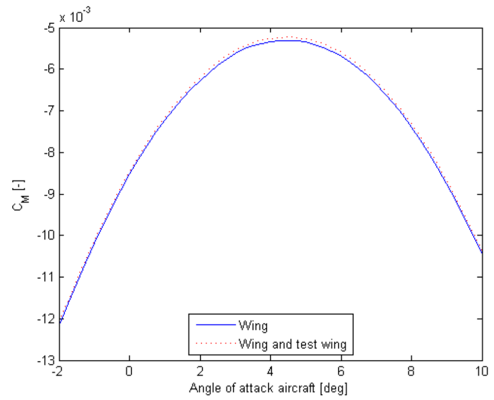


Figure 14.4: Moment coefficient B737-500

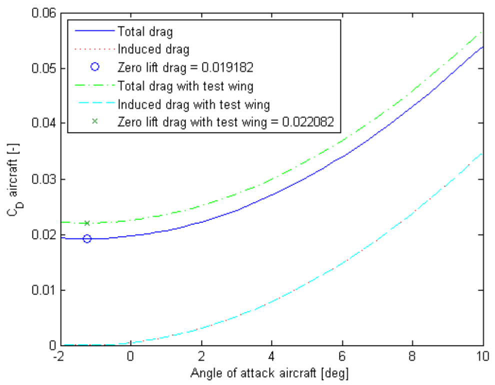


Figure 14.5: Drag coefficient of the B737-500

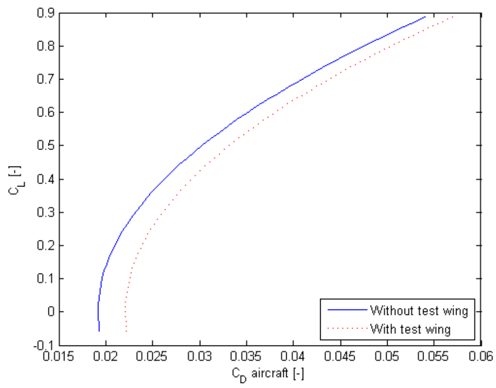


Figure 14.6: Lift-drag polar of the B737-500

The second analysis included the pylon and the test wings. The aircraft parameters, presented and used in the first iteration, were used for this analysis as well. The test wing used, had a chord of 1 m, a span of 8 m, no sweep, and was build up from the root airfoil of the B737-500. This relatively thick airfoil was chosen to represent high drag and lift forces. This test wing was inversely mounted on the pylon. Hence, when spoken of a positive angle of attack of the test wing it will rotate nose down.

This analysis, of the aircraft with the pylon mounted test section, focussed on non-testing flight. Under these conditions, the test wing would be actively set its minimum drag position. In Figure 14.3, the lift coefficient of the aircraft is presented over a range of angles of attack. As mentioned, the test wing is set at a fixed angle of attack for minimum drag, and will rotate such that if the angle of attack of the aircraft changes, the angle of attack of the test wing remains constant and therefore needs to change as well.

It can be seen that the lift coefficient of the aircraft with the test wing in minimum drag position is nearly the same as the lift coefficient of the aircraft without the test wing and pylon. The moment coefficient of the wing and test wing are added and shown in Figure 14.4. It can be noted that the moment coefficient changes a little due to influence of the test wing. The moment analysis is not representative for the moment coefficient of the aircraft since only the wing and the test wing are considered.

The drag coefficient, the zero lift drag, and the induced drag coefficient are shown in Figure 14.5. The induced drag is nearly the same for both configuration, with and without test wing and pylon. However, the zero lift drag coefficient is 15.1% higher for the aircraft with the pylon and test wing. Even though the total drag coefficient of the aircraft with the pylon and test wing is higher, the Mach number will go down from the transonic regime to 0.5. Therefore, more powerful engines will not be needed as the total drag will be much lower for a lower flying speed. However, at take off the drag is increased for the same speed. This means that a longer take off distance is required.

The result of the altered lift and drag coefficient can be seen in the lift drag polar in Figure 14.6. Here it can be seen that the lift coefficient spans the same range for both configurations, but the drag coefficient is higher for the aircraft with test wing. The lift to drag ratio, over a range of angles of attack, of both configurations is presented in Figure 14.7.

At the Mach number of 0.5 Mach at an altitude of 3000 m, the lift coefficient needed is calculated using Equation 14.18. The weight determined with a Class I weight estimation of 56472 kg was used for this calculation. The angle of attack corresponding to a lift coefficient of 0.429 is 4 degrees. At this angle of attack, the lift to drag ratio of the aircraft with test wing is 14. This lift to drag ratio is lower than the lift to drag ratio of 15 used during the Class I weight estimation in Section 10.1. This should be altered for the next design iteration.

$$C_L = \frac{2 * W}{\rho * (M * a)^2 * S} \longrightarrow C_L = \frac{2 * 56472 * 9.81}{0.9090 * (0.5 * 328.58)^2 * 105.4)} = 0.4285 \tag{14.18}$$

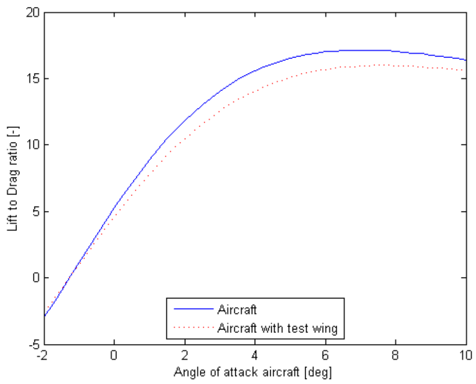


Figure 14.7: Lift to drag ratio

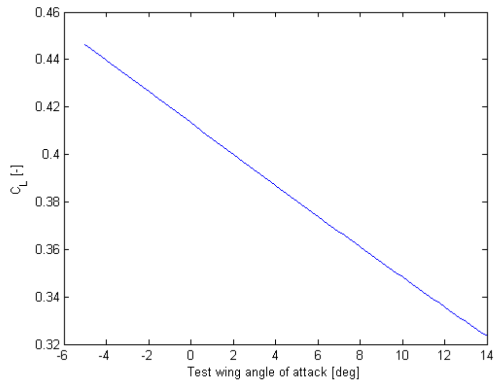


Figure 14.8: Lift coefficient test flight. Angle of attack is positive for aircraft nose-down

14.2.2 Testing flight

The last analysis of the aircraft with the pylon mounted test section was focussed on testing flight. In testing flight, the aircraft is kept in a horizontal, straight, steady flight, while the test wing is tested over a range of angles of attack. In Figure 14.8, the lift coefficient of the aircraft is given over a range of angles of attack of the test wing. It should be noted again that the test wing is inverted, meaning a positive angle of attack is nose down for the test wing. In Figure 14.8 it can be seen that the test wing influences the lift coefficient of the aircraft significantly. To maintain steady flight, the aircraft's angle of attack needs to be altered. In Figure 14.9 it can be seen that the total drag coefficient of the aircraft increases significantly. To maintain steady flight, the thrust level of engines needs to be adjusted. The moment coefficient of the aircraft during testing is shown in Figure 14.10. It can be noted that the changes in the moment coefficient are small when compared to values in Figure 14.4. Hence, little action will be required to keep the aircraft in horizontal, straight, steady flight.

14.3 Stability

After the aerodynamic characteristics of the aircraft are determined, an estimation of the neutral point of the aircraft can be made. In combination with the centre of gravity, the neutral point provides information about the longitudinal static stability of the aircraft. The neutral point is defined as the point where the resultant of the lift forces variations due to a perturbation is applied [17].

By definition of the neutral point, the variation of the moment around the neutral point, due to a perturbation, is zero. Assuming the drag perturbation moments of the wing and the tail are much smaller than the other perturbation moments, Equation 14.19 was used to determine the position of the neutral point. This assumption is based on the fact that the change in drag coefficient with a change in the angle of attack is significantly smaller than the change in lift coefficient with a change in the angle of attack. The drag perturbation moments caused by the test wing are considered, since the moment

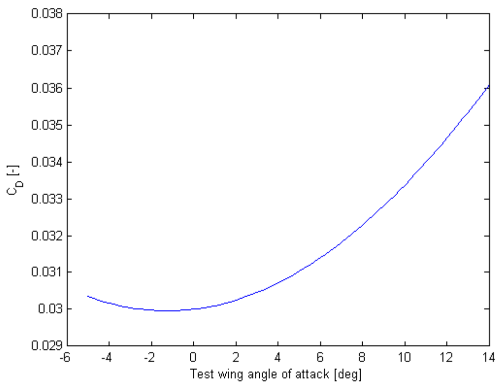


Figure 14.9: Drag coefficient test flight

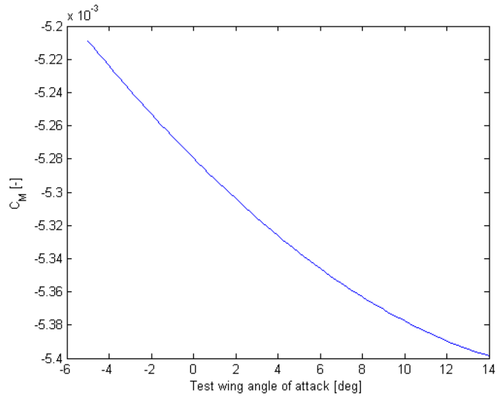


Figure 14.10: Moment coefficient test flight

arm is relatively big and the angle of attack of the test wing can be very high. It is also assumed that the test wing is in clear air, meaning it does not experience downwash from another wing nor does V_t differ from V.

$$\Delta L_w * (x_{np} - x_{ac_w}) + \Delta L_h * (x_{np} - x_{ac_h}) + \Delta L_t * (x_{np} - x_{ac_t}) - \Delta D_t * (z_{np} - z_{ac_t}) = 0 \tag{14.19}$$

The subscript 'w' stands for wing, 'h' for horizontal stabilizer/tail, and 't' for test wing. This equation can be rewritten to Equation 14.20, where non dimensional aerodynamic parameters are used to determine the position of the neutral point.

$$\begin{aligned} \bar{x}_{np} = \bar{x}_{ac_w} - \frac{C_{L_{h_\alpha}}}{C_{L_{w_\alpha}}}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\frac{S_h}{S_w}\frac{x_{np} - x_{ac_h}}{mac}\left(\frac{V_h}{V}\right)^2 \\ -\frac{C_{L_{t_\alpha}}}{C_{L_{w_\alpha}}}\frac{S_t}{S_w}\frac{x_{np} - x_{ac_t}}{mac}\left(\frac{V_t}{V}\right)^2 + \frac{C_{D_{t_\alpha}}}{C_{L_{w_\alpha}}}\frac{S_t}{S_w}\frac{z_{np} - z_{ac_t}}{mac}\left(\frac{V_t}{V}\right)^2 \end{aligned} \tag{14.20}$$

$$\varepsilon = \frac{2 * C_{L_w}}{\pi * AR_w * e} \longrightarrow \frac{d\varepsilon}{d\alpha} = \frac{2 * C_{L_{w_\alpha}}}{\pi * AR_w * e} \tag{14.21}$$

Table 14.4: parameters used for stability analysis

	C_{L_α} [-]	C_{D_α} [-]	S [m^2]	x_{ac} [m]	$V_{subscript}$/V [-]
Wing	0.0788		105	14.2	1
Horizontal tail	0.0667		28.9	27.83	0.850
Test wing	0.0857	-0.0007 to 0.0025	8.0	14.56	1

One quarter of the height of the fuselage was chosen as the z-position of the neutral point. The test wing is mounted 3 m above the fuselage, meaning it is 6.00 m above the neutral point. The change in downwash is defined in Equation 14.21. With all the parameters specified in Table 14.4 known, Equation 14.20 could be solved solved for the x-location of the neutral point. The neutral point was found to be between 16.48 m and 16.49 m after the tip of the aircraft. The variation in the neutral point is caused by the variation of $C_{D_{t_\alpha}}$. During the Class II weight estimation the centre of gravity was found to be between 12.71 m and 13.53 m after the nose. Using the most forward neutral point, the static margin, expressed as a percentage of the MAC, is 86.1% and 67.3% respectively. This means the centre of gravity can be moved backwards, in the next design iteration, to improve controllability of the aircraft. This would also relax the requirements on the position of the fuselage fuel tank.

15
Design overview

This chapter will present a brief overview of the results of the design. First the weight estimation will be presented in Section 15.1. Then the results of the wing redesign will be given in Section 15.2. Third, Section 15.3 presents the design of the pylon, continued by the interior design in Section 15.4. Last, Section 15.5 will address the aerodynamic properties. This chapter is limited to presenting the results of the iterative design process, rather than presenting the methods and results in chronological order. A final technical drawing of the modified B737-500 is presented in Figure 15.1.

15.1 Weight estimations

The implementation of the two weight estimation methods has given two parametric models to determine the fuel weight and the component weights of the aircraft. It was found that $18700kg$ of fuel was required to perform the defined mission. Assuming Jet-A fuel was used, the fuel tanks required a volume of $23.1m^3$. The results of the Class II method can be found in Table 10.5, where it was found that the OEW is $30854kg$.

15.2 Wing redesign

The wing redesign encompasses the accommodation of AFC systems and measurement systems. The top skin will be lowered by $5cm$ to create space for modular skin panels. These panels can be placed and removed at will on ground. The wing box is thickened to account for the loss in inertia caused by the reduced height of the load carrying box. The testing panels will contain both measurement systems and AFC systems, and each panel will have its own connection for pressure, power for AFC systems, power for measurement systems, and a data handling cable for measurement systems. Flaps, slats, and the wing tip are all removable and can be replaced to accommodate AFC devices on those locations. Systems that require modifications not possible within a panel, like morphing wings or rotating surfaces, have to be mounted on the pylon test section and cannot be implemented on the main wing.

The fuel tank originally covered most of the wing in between the spars, but the fuel tank is now sealed off after the engine pylon. The remaining space is modified to accommodate the test panels.

In total, seven 'plug 'n play' docks will be installed per wing, providing electrical power and pressure. Optionally, optic fibres for communication and data handling may be present as well, depending on the location of the dock. The wiring will run from the fuselage, through the landing gear hold and the flap hinge line to the flap actuator. There the cables will go into the wing box, penetrating the lower wing skin panel. This is done at the flap actuator to protect the cabling from the outside environment. A technical drawing of the wing is presented in Figure 15.3

15.3 Pylon design

The x-location of the test wing was found using the longitudinal moment balance of the aircraft. The lift gradients of the wing, tail, and the test wing build up from the root airfoil of the B737-500 with a geometry as described in Table 15.1, were used. A location of $14.31m$ after the nose was found for the leading edge of the test wing.

The height of the test wing above the fuselage was found using a visual method of Raymer. This method visualised the wake of the test wing, and hence the areas which can screen a control surface. It was found that the test wing needed to be placed $3.00m$ above the fuselage.

The dimensions of the pylon struts, and the corresponding maximum forces are found in Table 15.1. The maximum testing velocity and angle of attack for the test wing are also found there. The test wing is mounted using bolts and nuts, and it can pivot around an axis to change its angle of attack. The housing of the joint between the test wing and the struts provides space for a PIV system and the necessary cabling. The pylon struts are mounted on the inner frame work of the fuselage, using bolts and nuts. This joint is designed such that the fuselage skin does not carry any extra loads. A technical drawing of the pylon section is presented in Figure 15.2.

Table 15.1: Characteristics of the test wing and pylon struts.

Test wing		**Pylon struts**	
Wing area [m^2]	8.00	Length front struts [m]	3.25
Aspect ratio	8.00	Length back struts [m]	3.87
Maximum lift [N]	$-1.85 \cdot 10^5$	Radius front struts [cm]	7.50
Maximum drag [N]	$2.89 \cdot 10^4$	Radius back struts [cm]	7.50
Maximum testing Mach number	0.500	Thickness front struts [mm]	7.00
Maximum angle of attack [deg]	20.0	Thickness back struts [mm]	4.50

15.4 Interior redesign

The interior design focussed on the integration of the power, data handling and pressure pump subsystems. All of the original cabin furnishing has been removed to make room for these subsystem.

The upper fuselage section holds the engineering booth, the data handling system, the PIV system, and the pressure and power generation subsystem. The engineering booth has been located between the cockpit and the wing root. The PIV system is located above the wing root, where two cameras and one laser are placed. The cameras will be placed between clamps to shield them from vibrations, and optical fibres will run from the laser to the test section in order to form a laser sheet. The rear part of the fuselage will house the pressure and power generation subsystem, as well as the cooling systems and the data handling system. A separate power generator is located in the rear part of the fuselage to power the AFC system. The measurement system are powered by the already present APU.

The fuel tanks, which will be removed from the wings, will be placed in the cargo holds as two flat fuel tanks. The amount of fuel stored in the fuselage tanks comes down to $5m^3$, or 4050 kg. A preliminary estimation predicts that these fuel tanks would weight 380 kg, which below the maximum weight the cargo floor can hold. To protect these fuel tanks, multiple layers of energy absorbing and explosion suppressant foam will be applied to the inside of the fuel tank and underneath the cargo hold floor.

15.5 Aerodynamic properties

The lift and moment coefficient of the aircraft, with the test wing and pylon installed, do not change significantly in comparison to the configuration without the test wing and pylon. In this analysis the aircraft is in non testing flight, where the angle of attack of the test wing is fixed to its minimum drag position. The lift coefficient does differ significantly during testing mode, when the angle of attack of the test wing is altered.

It was also found that the zero lift drag coefficient was increased by 15% when the pylon and the test wing were installed onto the aircraft. However, the lift induced drag of the aircraft with and without test wing is almost equal. The difference is thus mainly caused by the relative difference in drag decrease with increasing induced drag. During non testing flight, at cruise conditions, a lift to drag ratio of 14 was found. During testing flight, the drag coefficient can differ significantly from the non testing flight drag coefficient. To maintain straight, horizontal, steady flight during testing, the angle of attack of the aircraft as well as the thrust setting of the aircraft need to be changed. The graphs presenting the aerodynamic coefficients can be found in Section 14.2

During the stability analysis it was found that the neutral point of the aircraft lies between $16.48m$ and $16.49m$ from the nose of the aircraft. With the centre of gravity being between $12.71m$ and $13.53m$ from the nose, the static stability margin was found to be between at 86.1$\%$ and 67.3$\%$.

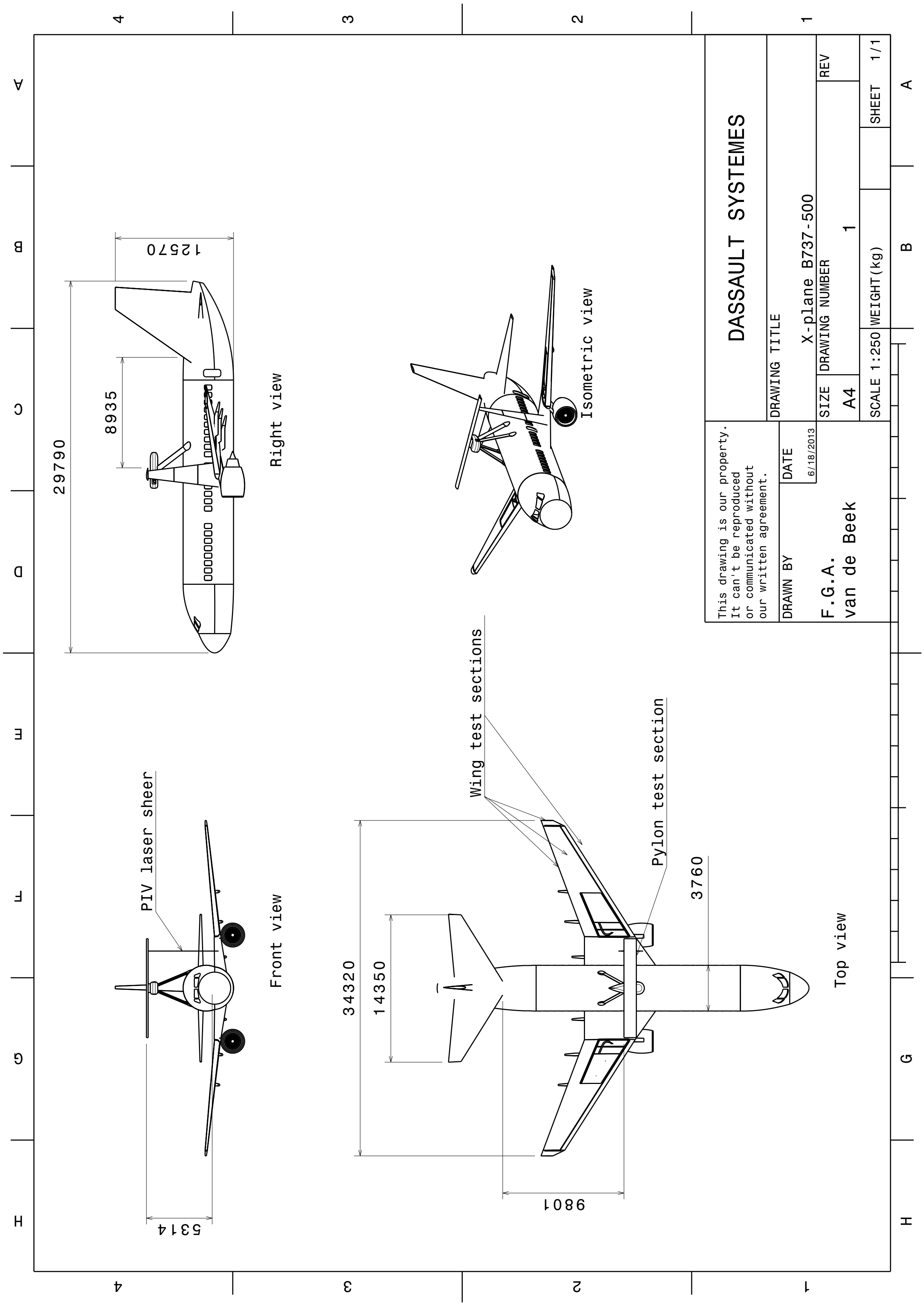


Figure 15.1: Technical drawing X-plane B737-500 design

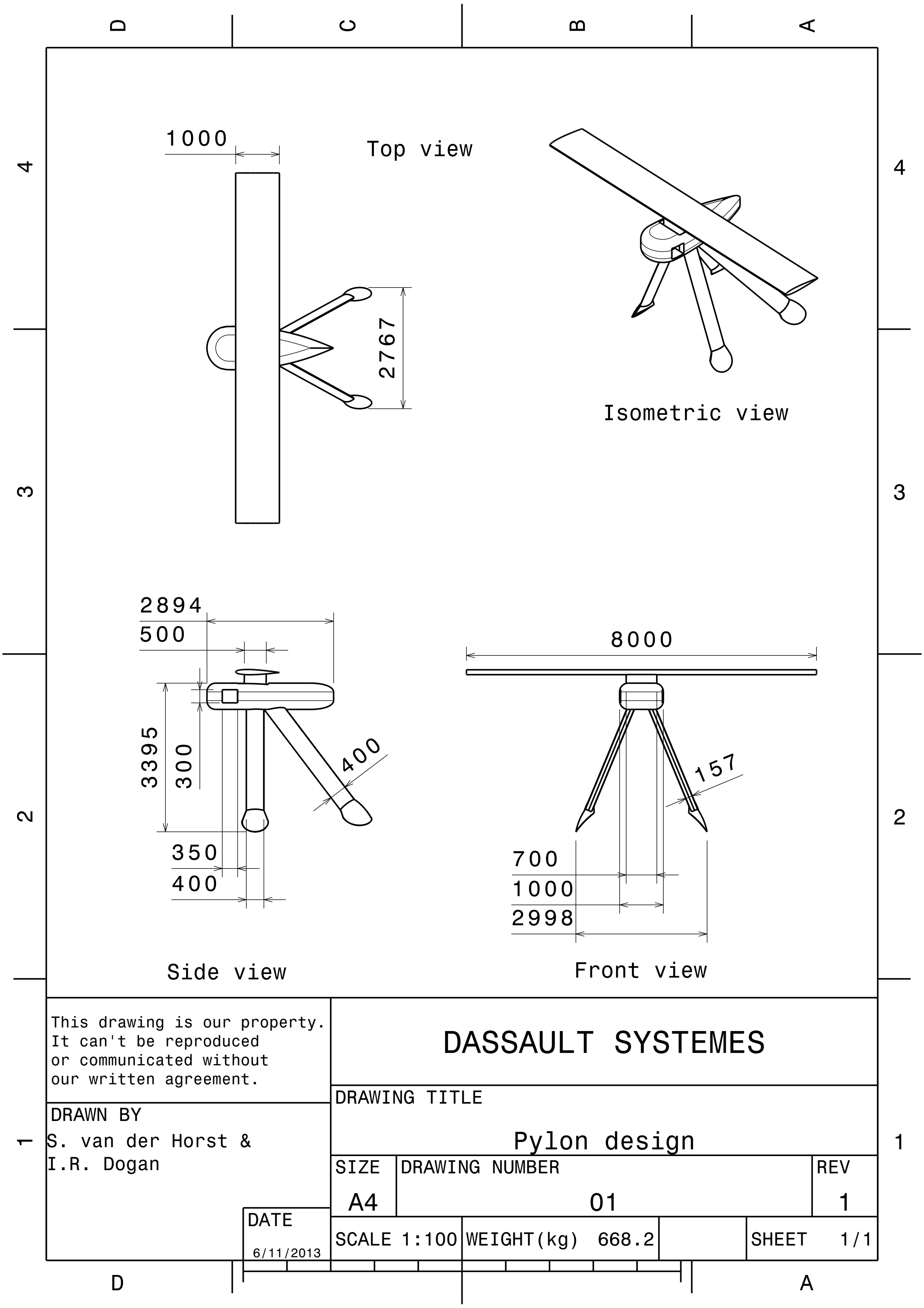


Figure 15.2: Technical drawing pylon design

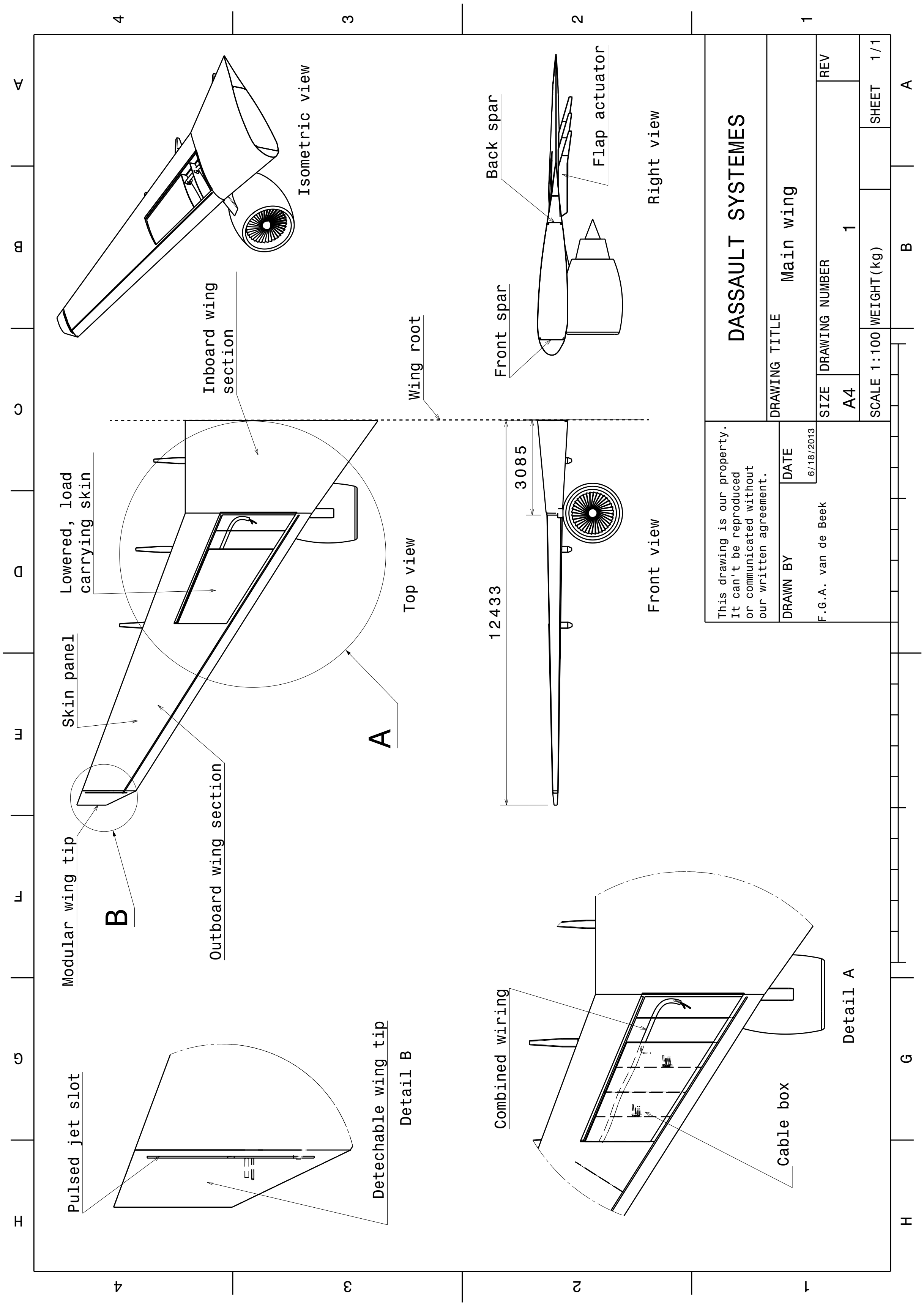


Figure 15.3: Technical drawing wing design

III

Design analysis

16

Flight test manual

This chapter contains the test flight manual for the Flow Control X-plane. The purpose of this chapter is to cover all procedures, rules, requirements, and guidelines associated with the practical use of the test platform. As such, this chapter will serve as an indication for the client what to expect, as well as instruct the client how to prepare their test panels.

16.1 Subject of testing

This section provides an overview of the systems which need to be implemented to perform the test. First the actuators will be briefly discussed, then the division of different levels of performance MS is repeated in this section.

16.1.1 Actuators to be tested

The platform will perform tests of integrated wing-based AFC technology in-flight. The specific AFC development team (the client) will deliver the system to be tested in a prescribed module (see for example Figure 11.3). This module can then be mounted on the test aircraft. The wing tip and pylon can be customized to test any AFC system, within the limits of weight and power requirements. Among others, the following actuators can be integrated in the main wing:

- Located near the leading edge: vibrating ribbon devices, piezoelectric vibrating parts, pulsed jet, zero net mass flux systems, fluidic oscillator.
- Located downwind of the leading edge: active dimples, suction actuators, Single Dielectric Barrier Discharger (SDBD).
- Located at the wing tip (vortex counteracting): pulsed jet, SDBD, suction and blowing actuators, rotating surfaces.

The following actuators can be integrated in the pylon test section only:

- Rotating surfaces, plasma heating, morphing structure

Any AFC system not listed above is not automatically infeasible. The compatibility of other kinds of AFC systems will be investigated on a case-by-case basis.

16.1.2 Performance to be tested

The effects caused by an AFC system, or a combinations of AFC systems, will be measured on three levels, namely:

- Aircraft performance level: lift enhancement, drag reduction, stability & control effects, noise effects, and other.
- Boundary layer level: transition delay, separation delay, vortex reduction, and other.

- Energy consumption level: power, pressure and fluid usage of AFC systems.

Boundary layer MS (PIV-system, pressure tabs, temperature sensors) are the most complex of the three level MS, and have a higher risk of failure. The overall performance and energy consumption MS are less complex. Therefore, these are less susceptible to any failure.

16.2 Integration of test panels on the wing

In order to assure plug-and-play capability, it is necessary to have an easy attachment possibility of the wings. Now briefly it will be explained how these are installed, and what the requirements are.

16.2.1 Installation procedure

The test panels are fitted to the wing by means of bolts and nuts. The client has to prepare panels of appropriate size. The panel is placed from the top side into the wing. After placing the panel, the bolts should fit through holes in the load-carrying skin of the aircraft, as explained in Chapter 11. Nuts are placed on the bottom side of the top skin from the inside of the wing box to fix the panel to the wing. The inside of the wing box is accessed through the maintenance hatches on the bottom side of the wing.

16.2.2 Requirements on test panels

The test panels have to follow requirements on size, power consumption, and required pressure for pressure driven actuators. The outer dimensions of the panels should fit exactly in the wing. The top side should follow the exact aerodynamic shape of the wing. The height of the panel should be 5 cm. The horizontal dimensions are limited by the spars in chord direction, and due to the taper of the wing depend on the spanwise position of the panel.
The power consumption of all the AFC systems on one wing together is limited to 5 kW. This excludes measurement systems, which are powered separately. The power cables for the measurement systems should be kept separate from the power cables for the AFC systems. The required pressure should not exceed 1 bar in total per wing.

16.3 Integration of test section wing

Next to the panels also some requirements and procedures apply to the test section wing. These will be dicussed here.

16.3.1 Installation procedure

The attachment of the test section wing consists of bolts and nuts. The bolts are attached to the joint on the pylon housing. The nuts are tightened inside the wing. To do so, an upper skin panel needs to be removed. This also grants acces to the ends of the cabling provided here. The client needs to make sure all of the power supply and data cabling ends under this panel. As such they can easily be attached to the ends of the cables at the joint. Also, the optic fiber for the laser ends here and can be attached to the optics inside the wing.

16.3.2 Requirements on the test section

Not every test section can be mounted. Regulations formulated during the design process constrain the characteristics of the test wing. As the pylon and its joints are designed to resist limited loads and stresses, the maximal forces produced by the test wing are limited as well, consisting of the lift and drag force. This means that at a given height and velocity, the combination between the aerodynamic coefficients and the surface area of the test section is bounded. Therefore, the airfoil and its corresponding area needs to be chosen carefully. The height at which can be tested is an altitude of at least 3000 metres.

Also, the wingspan of the section is limited as well. A maximum span of 8 metres is possible. This boundary has been set from a safety point of view. The aircraft is not designed to cope with the larger moments produced by wider wings than this. Also, the controllability of the aircraft will be affected too much by wings of larger sizes. Therefore this limit is communicated to the client. The rest of the test wing parameters are more relaxed. The client has more freedom in the choice of these parameters, such as the test wing sweep angle, taper ratio, dihedral and other. The exact influence of these parameters has not yet been properly assessed due to time issues.

16.3.3 Installation of the pylon

The pylon is detachable as well. When doing performance tests on the main wing only, it would be beneficial to completely remove the pylon. Otherwise the generated pylon drag would negatively influence the performance of the aircraft. The pylon has been designed in such a way that it can be removed from its supports easily, leaving only the connection points in the fuselage.

Using bolts and nuts located at the interior frame of the fuselage, the joints can be uncoupled as well. The fairing of the fuselage skin is detachable to grant access to the attachment (see Figure 12.8). The skin is reinforced, reducing the loads carried by the fairing. This means that it could consist of an easily detachable, minimal drag shape. If the aircraft is flown without the pylon, this small skin part is replaced by the original, flat fuselage skin.

As the pylon is one integrated structure, it can not be taken apart when it is detached. This means, that a crane is required to lift the pylon from the fuselage. The bolts are designed to be vertical, so the pylon is easily lifted from the attachment. The weight of the pylon is minimal compared to what such cranes can maximally lift. The only requirement is that the crane can lift high enough.

16.4 Testing procedure

The specific testing procedure will be explained by first stating the general approach. This explains how this project testing program is set-up. Then the testing management controlling this process is explained. Furthermore, it is stated what the outcome of the test should be in Subsection 16.4.3, when the test is considered successful in Subsection 16.4.4 and what should be done in case the mission is jeopardised in Subsection 16.4.5.

16.4.1 Testing levels

The testing of the specific active flow control system in general will consist of unit, system/integration and acceptance test levels. This project is responsible for creating a platform that enables system/integration level testing of AFC systems. The three just defined test levels are defined as follows:

- **Unit:** Testing on this level will be performed and approved by the developer of the AFC system(s). Proof of unit testing (such as test case list, output, data printouts, defect information) must be accepted before integration testing.
- **System/Integration:** Testing on this level will be performed by the X-plane test team with assistance from individual developers. The platform is the major test tool and will be prepared for the specific test. During testing, personnel such as engineers are required to handle the tools. AFC systems will enter System/Integration tests after all critical defects have been corrected. Minor defects in the AFC systems are only allowed as long as safety of the test can be guaranteed and as they do not impede testing of the program.
- **Acceptance:** Testing will be performed by the actual end users (aircraft manufacturing companies) with assistance of the developers. This entails certification of the new aircraft with implemented AFC systems. The test plane will enter into acceptance test after all critical and major defects have been corrected.

16.4.2 Management control

The program of the testing procedure will be controlled by the Testing Management. The testing management can be split up in various teams: an AFC Development Team (the client), Management & Planning, Preparation, Execution and Verification Team. This division is visualised in an N2-chart, which is shown in Figure 16.1.

AFC development (client) All unit and initial system testing will be performed by the AFC-development team. They will deliver unit test results, test requirements, and a AFC system module that can be implemented into the base aircraft.

Management & planning: Planning of the test will be performed by management & planning in combination with the client. They will write the test plan. It includes a description of the configuration and a test script which will be used by the other teams. The test script describes the mission profile and the strategy for the on-board scientists/engineers.

AFC Development Team (Client)	• Unit Test Results • AFC System module • Test Requirements		• Real Time Monitoring	
• Test Plan	**Management & Planning**	• Test Configuration	• Test Script (Manual)	
		Test/Configuration Preparation Team	• Test Setup	• Post-Flight Configuration Check
			Test Execution Team	• Real Time Measurements • Raw Data
• Test Results	• Improvements		• Real Time Monitoring	**Data Verification Team**

Figure 16.1: N2-Chart of Test Management.

Preparation: The AFC module will be installed by the Test/Configuration Preparation Team. They will install and calibrate the MS accordingly. It is their responsibility to prepare the aircraft for flight and they will also check the configuration post-flight. Once the testing mission is completed they will uninstall the AFC module.

Execution: The execution team exists of engineers, specialists and pilots which are required to execute the test. They will test according to the test script unless real time test improvements are suggested. The measurements will be performed and the acquired test data will be stored on-board. Preliminary processing will be done on-board and checked by the Verification & AFC Development team. Post flight data will be given to the client on a hard disk for further processing, guaranteeing discretion towards the client concerning the test data.

Verification (client): The verification team is responsible for the processing of the data. They will receive the data and check if quality is sufficient. Test results will be presented by the data Verification team to the AFC Development team. In case data requirements are not met, the whole test result will be considered as failure. A new test needs to be done to obtain correct data.

16.4.3 Test deliverables

After a test all of the following should be delivered to the client:

- All the flight data recorded over the entire flight, as specified under Subsection 16.1.2.
- The timestamps of the moment when AFC system(s) were turned on.
- Data of the test aircraft in comparable conditions without the AFC systems operating to show the difference made by the AFC system(s). This will serve as a reference, or normal measurement.

16.4.4 Test pass/fail criteria

The flight test can be considered to be a success once all the tests required by the client have been completed and the required data has been collected successfully, as described in Subsection 16.1.2. Successfully means that all the data has been measured with sufficient accuracy and reliability.

16.4.5 Suspension criteria and resumption requirements

If at any point during a test the safety is jeopardised, the test has to be aborted immediately. In case of an event that cannot be fixed immediately in flight, the aircraft needs to land as fast as possible. Further testing is postponed until the cause of the event has been identified and resolved. In case of stability issues the centre of gravity could be moved forward or backwards to continue testing safely.

16.5 Liabilities

First the operational risks are discussed in order to identify potential failure events. Some tests need to be performed under certain conditions. The performance of both AFC systems and measurement equipment constrain the situations in which testing can occur. Both environmental and staffing constraints are considered. Finally, the responsibilities of every party is noted in Subsection 16.5.5.

16.5.1 Operating risks

Several parts of the project cannot be influenced by the operating teams. These could pose a threat to the test mission or the entire project. The risks should be identified and it should be checked that they will not interfere with the process. If they do they will need to be addressed according to the appropriate mitigation plan. The risks related to the AFC module obtained from client can be found below:

- Any identified risk during unit testing or any past history of failure of the specific AFC actuator.
- Delayed development or production of AFC systems by the client which delays the flight test.
- Requirements from the AFC development team that cannot be met with the existing platform.
- AFC module provided does not meet the size, structural or power requirements and cannot be mounted on the aircraft.
- Unidentified risks of the use of such actuator(s) in-flight

Risks which are inherent to the testing mission itself, are identified as follows:

- Safety risks of flying
- Safety risks of using PIV laser
- Any measurement or processing deficiencies or errors that lowers quality of data (calibration error, unfavourable weather/environment, etc) beyond an acceptable level.
- Human errors within the teams (communication or handling). This includes misunderstanding of requirements, test manual, configuration manual, and other.
- Unexpected interference of systems and flow disturbances

16.5.2 Safety measures

With the risks identified in Subsection 16.5.1 certain safety measure have been taken. These measures will assure data quality and flight safety. They are given as follows:

- The pilots have an emergency override switch for the AFC systems. This override switch ensures that the pilot can gain control easily without communication with and action from the engineering booth.
- The flight systems and measurements system cannot be operational on the APU at the same time. By eliminating the MS and DH systems in case of an emergeny from the APU the energy requirements for the flight systems can be guaranteed.
- Fuel tank protection has been installed for emergency belly landings.
- Space for an engineering booth has been reserved such that initial verification can be done during flight.

16.5.3 Environmental constraints

PIV requires flight at dusk as sunlight can interfere with the cameras too much.

On the other hand, areas with clear weather are required to give 'clean' measurements, as well as to make the validation process easier. The required weather varies with the test objective, as data needs to be gathered in various weather conditions to investigate weather influence. Landing and take-off testing can only be done during daytime due to visual flight rules.

16.5.4 Staffing needs

Engineers representing the client providing the AFC system are given the possibility to join the test flight. They can perform a pre take-off inspection to ensure that all the MS are connected correctly and work properly. The pilots need to be knowledgeable about the test aircraft, and need to know in advance what kinds of tests are requested by the client.

16.5.5 Responsibilities

Ensuring the safety of every person on board is the responsibility of the pilots. The pilots will make the final decision regarding any flying manoeuvre. It is the responsibility of the client to have their AFC system ready and fulfilling the requirements in time before the flight test. In case of delays caused by the client, costs arising due to this delay are paid for by the client.

17
Risk assessment

This chapter concerns the identification, mapping and assessment of technical risk during the detailed design phase. First, possible failure events which might occur during this phase of the design process are indicated. Secondly, the probability and consequence of these events are determined. If these are found to be unacceptably high, mitigation strategies are proposed. This is done in order to keep the design both within budget and on time with adequate technical performance.

17.1 Failure event identification

In order to assess technical risk, one first has to know which events might lead to failure. To this end, a list of possible calamities was constructed. The classification of these events follows the same structuring as the preliminary design phase followed, due to the fact that during the detailed design phase this same division will be used to investigate the subjects in more detail. In Table 17.1 the list of events which were identified with their respective code is given.

17.2 Risk map

In this section, the risk each of the failure events defined in the previous section is assessed and represented graphically in a risk map. Risk in the following is defined as the product of the probability of occurrence and the consequence on technical performance.

The probability that these events actually arise is graded between 0 and 1 in five categories. These range from proven flight design (0-0.2), to extrapolation from existing flight design (0.2-0.4), to based on existing non-flight engineering (0.4-0.6), to a working laboratory model (0.6-0.8) to feasible in theory (0.8-1.0). The consequences on technical performance were also divided in five classification. These effects were ordered from negligible (1), to marginal (2), to considerable (3), to critical (4) to catastrophic (5).

The product of these two result in a certain risk value. Three contingency strategies can be applied. The failure events with a risk value of less than or equal to 0.8 will be accepted (A). No further action is taken regarding these specific events since even when they would occur, their influence would be limited. Risks with a value higher than 0.8 and lower than or equal to 1.5 will be watched (W). This means they will be actively monitored, but counteracting measures will only be taken when the risk tends to increase. For the events of the highest level of risk, which are graded above 1.5, specific mitigation strategies will be developed. This is done in order to prevent the failure event from happening in the first place and reduce risk to a tolerable or acceptable level. The mitigation strategies will be presented in the next section. The results of the risk assessment are summarised in Table 17.2, whilst the visual representation can be found in Figure 17.1. This risk map plots the probability versus the consequence. The items in the upper right corner induce the most risk during the detailed design phase.

17.3 Mitigation strategies

Mitigation strategies will be made and applied to the design failure events associated with the highest level of risk. The design failure risks of concern are expressed in WD-1, WD-3, WD-4, WD-5, WD-6, PD-4

Table 17.1: List of failure events for each category with the respective code

		Code
	Aerodynamic properties	
1.	Additional forces and moments induced by the AFC systems adversely affect the controllability of the aircraft.	AP-1
2.	Additional forces and moments induced by the AFC systems adversely affect the stability of the aircraft.	AP-2
	Weight estimation	
1.	Reinforcements of skin panel cut-outs become too heavy in order to not exceed the certified MTOW.	WE-1
2.	Reinforcements of the fuselage structure become too heavy in order to not exceed the certified MTOW.	WE-2
3.	Reinforcements of the modified wing box become too heavy in order to not exceed the certified MTOW.	WE-3
	Wing design	
1.	Modular skin panels cannot be attached to the rest of the wing structure without carrying loads.	WD-1
2.	Ice protection cannot be accommodated for AFC and test measurement systems installed on the main wing.	WD-2
3.	Future AFC systems cannot be integrated at the desired location.	WD-3
4.	AFC and test measurement systems installed on the main wing cannot be protected against dust and interfering particles.	WD-4
5.	Higher wing loads induced by experimental wing tips exceed the limit load of the main wing at that position.	WD-5
6.	Critical stress locations in the wing box are not accurately analysed.	WD-6
7.	Wiring and piping cannot be installed at the desired location over the span of the main wing.	WD-7
	Pylon design	
1.	Loads induced by the pylon exceed the limit load of the fuselage.	PD-1
2.	Tail effectiveness becomes unacceptable due to the wake of the test section under stall conditions.	PD-2
3.	Ice protection cannot be accommodated for AFC and test measurement systems installed on the test wing.	PD-3
4.	AFC and test measurement systems installed on the test wing cannot be protected against dust and interfering particles.	PD-4
	Interior design	
1.	Redesign of the interior and installation of necessary equipment introduce a load concentration exceeding the limit load for the fuselage.	ID-1
2.	Active fuel system cannot fulfil the required centre of gravity shifts.	ID-2
3.	Fuel vapour management of the redesigned fuel tanks fail.	ID-3
4.	Required power exceeds the peak power of the electrical system.	ID-4
5.	Installed isolation for electrical and magnetic fields does not provide adequate interference protection.	ID-5
6.	Existing access points to the interior prohibits the installation of critical components inside the aircraft.	ID-6
7.	Modifications to the upper interior compartment of the fuselage require a redesigned floor structure.	ID-7

Table 17.2: Risk values for the identified failure events and the corresponding strategy

Code	Risk value	Strategy	Code	Risk value	Strategy
AP-1	0.9	(W)	PD-1	1.5	(W)
AP-2	0.9	(W)	PD-2	1.2	(W)
WE-1	0.6	(A)	PD-3	1.0	(W)
WE-2	0.8	(A)	PD-4	1.8	(M)
WE-3	1.2	(W)	ID-1	0.4	(A)
WD-1	2.0	(M)	ID-2	0.6	(A)
WD-2	1.0	(W)	ID-3	0.5	(A)
WD-3	1.8	(M)	ID-4	0.8	(A)
WD-4	1.8	(M)	ID-5	0.9	(W)
WD-5	2.0	(M)	ID-6	1.6	(M)
WD-6	3.5	(M)	ID-7	0.6	(A)
WD-7	0.9	(W)			

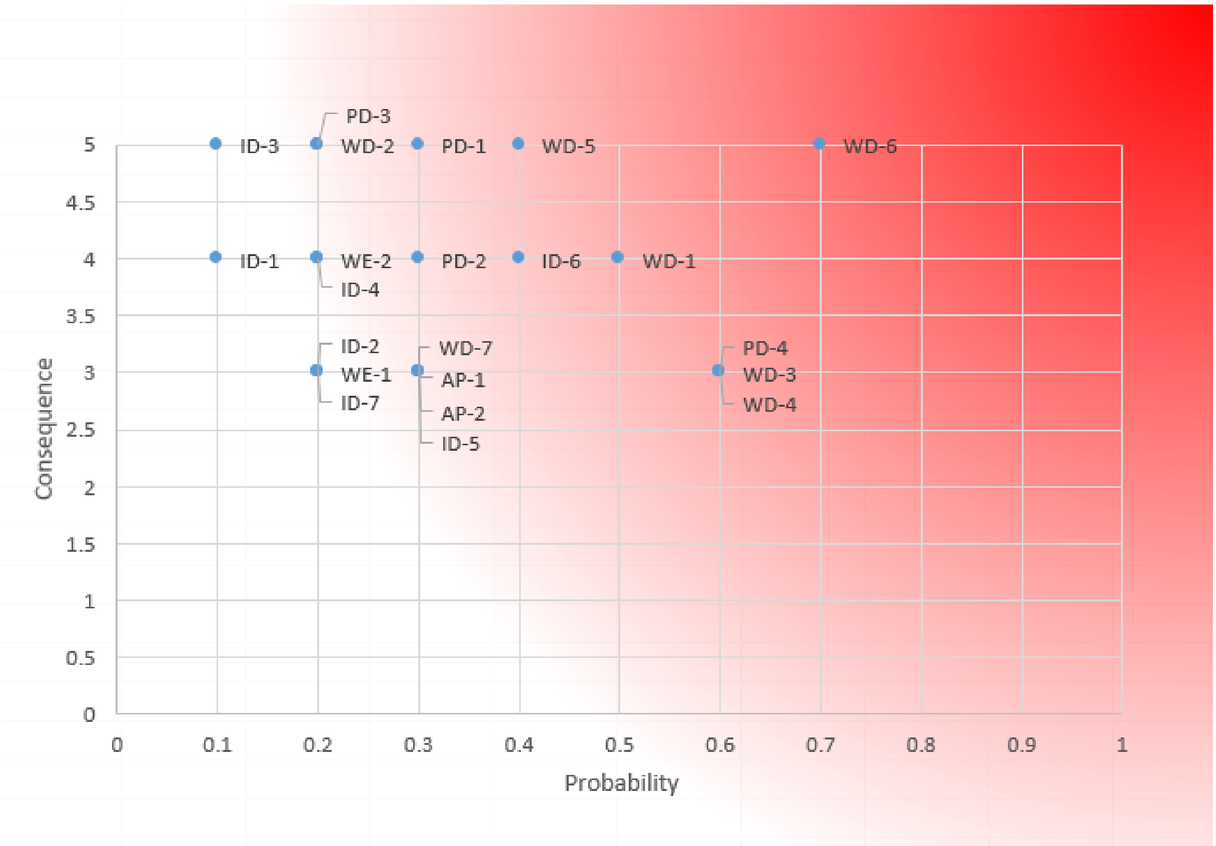


Figure 17.1: Risk map of possible failure events

and ID-6. The mitigation strategy to reduce the technical risk, addressing each of these failure events, will be described next.

WD-1
Modular skin panels cannot be attached to the rest of the wing structure without carrying loads. The modular skin panels will be attached to the load carrying top skin of the redesigned wing box by means of screws and nuts. However, loads will be carried to the modular skin panel through the screws. An option is to leave some space at the connection points where the rest of the structure can deform, in order to leave the modular skin panel relatively untouched. More research needs to be done about these joints and connection methods to ensure that the modular skin panel will carry as little loads as possible to reduce deformation which could influence the measurements.

WD-3
Future AFC systems cannot be integrated at the desired location. When this happens, for example due to size or power input constraints, an AFC actuator will not be as effective as it was designed for when placed at a different location. Since the areas where AFC systems are placed are fixed at this point in the design, research should be done on the effectiveness of each AFC actuator on multiple positions of the wing. When this is known, the performance of the AFC system can still be assessed. Another option is to test these future AFC systems on the pylon mounted test wing, which can be build to suit a certain AFC system (taking into account constraints on the Reynolds number).

WD-4 & PD-4
The AFC and measurement system cannot be protected against dust and interfering particles. The experimental testing of AFC systems in real life conditions introduces the complication that adequate protection needs to be offered to the systems. In addition to cleaning the wing surface by flying through rain and clouds [34], some active measures can be taken. These could include coatings, fluid layers or liquid/foam systems [34]. Further research in this matter, of which many systems are still experimental, is needed.

WD-5
Higher wing loads induced by experimental wing tips exceed the limit load of the main wing at that position. The redesign of the wing should include reinforcement for the higher loads which can be induced by different types of wing tips. To avoid loads which are too high on the outboard part of the wing, a certain range of loads should be determined which the wing box can handle. Preliminary research should be done on each experimental wing tip that is going to be used. By doing so, the loads induced by these tips can be predicted and a decision can be made on whether they will be installed on the wing or not.

WD-6
Critical stress locations in the wing box are not accurately analysed. More accurate design tools, capable of calculating stresses in complex geometries, should be used for the next design step. This should be done in order to avoid using higher safety factors than needed, which make the structure heavier. Even more important, this more detailed assessment of the stresses will avoid catastrophic structural failure.

ID-6
Existing access points to the interior prohibits the installation of critical components inside the aircraft. When choosing the hardware to be installed in the fuselage, dimensions of the hardware and the access points to the interior should be taken into account. When certain parts will not fit through the main door, the fuselage cargo door can be used. However, when the fuselage cargo door will be used, a hatch needs to be made in the floor panel if the part needs to be installed on the upper deck of the aircraft. This adds weight which should be kept to a minimum.

18

Discussion and future work

In this chapter the compliance of the design to the requirements is discussed. A compliance matrix, which provides an overview of which requirements were met or which were not met, is constructed and presented in Section 18.1. Section 18.2 explains the reasons why certain criteria were not met. Finally, recommendations are given for future work in Section 18.3.

18.1 Compliance matrix

In Table 18.1 the compliance matrix can be found. In this matrix, each individual requirement is listed by its identifier and assessed whether it is met or not. The explanation of the identifier can be found in Chapter 7. When the requirement is fulfilled, a reference is given to the relevant part of this report dealing with that specific aspect of the design. The non-compliance of certain requirements is discussed in the next section.

Table 18.1: Compliance Matrix, ☑indicates a complied requirement, ☐indicates an uncomplied or in-progress requirement

Identifier	Report reference	Compliance
Top level		
TLR-AD-01	Chapter 8	☑
TLR-AD-02	Section 13.2	☑
TLR-AD-03	Chapter 8	☑
TLR-AD-04	Chapter 12	☑
TLR-AD-05		☑
TLR-AD-06		☐
TLR-AR-01	Chapter 12 & Section 11.2	☑
TLR-TS-01	Section 13.3	☑
Airframe		
FR-TS.1-01	Chapter 12	☑
FR-TS.1-02	Section 13.7	☑
FR-TS.1-03	Section 13.7	☑
FR-AR.1-03	Section 11.2	☑
SR-AD.5-01	Section 11.2	☑
SR-AD.5-02	Section 11.2	☑
SR-AR.1-01	Chapter 12 & Section 11.2	☑
SR-AR.1-02		☐
SR-TS.1-01		☐
SR-TS-07	Chapter 10	☑
SR-AD.2-05	Section 10.3	☑
SR-AD-10	Section 11.1	☑
SR-AD-11	Section 11.1	☑

Identifier	Report reference	Compliance
SR-AD-13	Chapter 12	☑
SR-AD-20		☐
Propulsion		
SR-AD.2-01	Section 10.1	☑
Electrical power		
FR-AD-01	Section 13.5	☑
FR-TS.1-04	Section 13.5	☑
FR-TS.1-05	Section 13.5	☑
FR-AR.1-02	Section 13.5	☑
SR-AD-01		☐
SR-AD-02	Section 13.5	☑
SR-AR.1-04	Section 13.5	☑
SR-TS-08		☐
SR-TS-09		☐
SR-AD.2-06		☐
SR-AD-14	Section 13.6	☑
SR-AD-16	Section 13.6	☑
SR-AD-17		☐
SR-AD-18		☐
SR-AD-19		☐
SR-AD-21	Section 13.6	☑
Stability & control		
SR-AD-06	Section 12.1	☑
SR-AD-07		☐
SR-AD-08	Section 12.1	☑
SR-AD-09		☐
AFC systems		
SR-AR-01		☐
SR-AR.1-05	Section 13.4	☑
SR-AR.1-03	Section 13.2	☑
SR-AR-03	Section 11.2	☑
Data handling		
FR-TS-01	Section 13.7	☑
FR-TS-02	Section 13.7	☑
FR-TS-04	Section 13.7	☑
SR-TS.1-02	Section 13.7	☑
SR-TS-01		☐
SR-TS-02		☐
SR-AD.2-03		☐
SR-TS-10	Section 13.7	☑
SR-TS-11		☐
SR-TS-12	Section 13.7	☑
Measurement equipment		
FR-TS-05		☐
FR-TS-06		☐
FR-TS-07		☐
FR-TS-08		☐
SR-TS-04		☐
SR-TS-05		☐
SR-TS-06		☐
SR-AD.2-04		☐

18.2 Discussion of compliancy

This section provides a brief discussion of the compliance matrix in the previous section. For every classification that was made, reasons why the specific requirement(s) was/were not met are given. When the details of the requirement can only be estimated in a later design phase this is stated as well.

18.2.1 Top level

One of the top-level requirements is not complied with. For the other top-level requirements their compliance cannot be verified yet. TLR-AD-06 will not be complied with, since an already very aggressive preliminary cost estimation places the design at a cost of €17.8 million as found in Section 10.3. During the design process, the costs are likely to increase even further. After discussion with the stakeholders, it was established that this requirement is rather lenient, and not meeting it does not come with grave consequences as long as the budget remains within realistic boundaries. TLR-AD-05 can only be checked once the final product is ready for testing, due to the fact that only then the designed provisions to assure modularity can be tested.

18.2.2 Airframe

The test platform was designed during the preliminary design to accommodate the necessary AFC actuators, test measuring systems and data handling equipment. Sections of the main wing, flaps, slats and wing tips are available for testing. In addition, AFC systems can be integrated over the complete span of the test section wing as well. This ensures that the desired AFC system can be implemented at the desired location, fulfilling SR-AR.1-01. SR-AR.1-02 and SR-TS.1-01 both deal with installation times, which can only be verified when the test platform is finished. Requirement SR-AD-20 was not complied with, since the analysis was deemed too complex for the time being. Safety measures to protect the fuel tanks located in the lower part of the fuselage however have been taken, as explained in Section 13.2.

18.2.3 Propulsion

The only requirement that was set for the propulsion system has been met. It states that aircraft should be able to hold at least 18700 kg of fuel. The amount of fuel was determined after a Class I estimation. Further analysis proved that around 1600 kg of additional fuel can be accommodated for.

18.2.4 Electrical power

Where possible, the electrical system was sized to comply with the prescribing requirements. Some factors are, at this stage of the design, not determined yet and as such, those requirements were not met. The system is able to deliver more power than strictly necessary and should be able to provide enough peak power (SR-AD-01), but this can only be verified in a later phase. The exact weight and reliability can only be confirmed during testing (SR-TS-08 & SR-TS-09). It proved to be impossible to come up with a preliminary system cost for the electrical power system (SR-AD.2-06). Up to this point, only provisions have been made to place the cables and nothing has been investigated about their properties (SR-AD-17, SR-AD-18 & SR-AD-19). This was deemed to detailed at this preliminary stage of the design and will be determined during the detailed design.

18.2.5 Stability & control

The static stability was carefully checked during the positioning of the test section. Therefore, also with the pylon-mounted test wing in place, SR-AD-06 is met. The dynamic stability however (SR-AD-07), and the controllability according to CS25 regulations (SR-AD-09) are a lot harder to ascertain. Flight tests of the complete test platform are needed to make sure these requirement is met.

18.2.6 AFC systems

The different inputs needed for the AFC systems have been taken into account during the preliminary design (SR-AR.1-03 & SR-AR.1-05). To ensure stability and not introducing unwanted moments, all AFC systems are placed symmetrically, complying with SR-AR-03. The exact maximum weight is not specified, so SR-AR-01 is not met. However, no significant problems are expected, since the AFC systems are lighter than the fuel that has been taken out of the wing and the wing box has been reinforced.

18.2.7 Data handling

Due to the lack of a comprehensive cost and weight estimation, SR-TS-01 and SR-AD.2-03 are not met. The estimations would need to be more accurate than the Class II estimation performed in this preliminary design. The weight of the data handling system is conceived to be negligible with respect to other systems. The reliability criterion (SR-TS-02) can only be assessed during operations. The storing system is made redundant, meeting SR-TS-10. SR-TS-11 can be checked when a specific type of processing unit is selected. Since the data handling system can handle data coming from more than two different measurement systems, redundancy is guaranteed and SR-TS-12 is met.

18.2.8 Measurement equipment

As mentioned earlier, weight and cost are difficult to accurately predict and SR-TS-05 and SR-AD.2-03 remain unchecked. For the reliability requirement, the same explanation holds as above and the accuracies of the systems strongly depend on the exact model and manufacturer of the measurement systems which will be chosen during the detailed design. Furthermore, none of the required accuracies have been determined, leaving the remaining requirements unchecked as well. More research needs to be done regarding the exact accuracy levels that the state of the art measurement equipment attain.

18.3 Recommendations for future work

As with every design project, certain aspects of how the preliminary design of the test platform was performed could be improved. These will be elaborated upon in the following.

A more accurate method of estimating both cost and weight are required to provide a more detailed overview of the distribution of both parameters over the complete test platform. At this point in the design, rather old methods were used and more up to date techniques should be considered.

Furthermore, a more comprehensive literature study regarding the exact specifications (such as the accuracy) of the test measurement systems needs to be done. Due to the time constraint, this was not explored exhaustively.

The preliminary design analysis was performed using mostly statistical relations which were used on separate parts of the aircraft. The characteristics of the complete aircraft is more than the sum of its different parts. For the lift and drag estimation for example, interference effects were not effectively taken into account. When more iterations of the design cycle are done, this can be integrated. Another limitation of the methods employed during the preliminary design, is that for the different parts of the aircraft (wing, pylon, fuselage) different techniques of estimation of the structural modifications was used. In the next phase, a more detailed and coherent method should be chosen.

A Finite Element Method (FEM) could be used to find the approximate solutions to boundary value problems which were deemed to complex to work out during the previous phase. Also for aerodynamic considerations of both the original aircraft and the modified test platform with pylon-mounted wing would benefit significantly by using a CFD tool. It would provide more detailed information about how the different parts interfere with each other and how the wake of the test wing influences the functionality of the tail. A more elaborate and detailed analysis regarding stability and control could also be done using this tool.

To accurately perform these calculation techniques, more information should be available about the exact geometry of the structural components of the base aircraft. Since most of these parameters are classified, more research should be done to correct for the very conservative estimates which were used for preliminary sizing. More realistic values limit the risk of overdesigning the structure and introducing more weight increase than absolutely necessary.

The design thus far has primarily looked at state of the art AFC systems and test measurement techniques which are entering their experimental phase at this moment. Although these will certainly cover the testing need for the coming years, future systems and techniques which are currently at a lower level of technical readiness should be considered as well. To avoid costly redesign in later stages of the product its lifetime, accommodations for these systems should already be investigated. A comprehensive literature study regarding these systems should be undertaken.

19 Conclusion

The purpose of this project was to develop an experimental cost-efficient aircraft that is able to test current and future flow control technologies in flight, by ten students in eleven weeks. To structure the process of development, the project was divided in four different phases: project organisation, mission analysis, conceptual design, and preliminary design. All four phases have been completed successfully. This report emphasises the results of the fourth and final phase. The resulting design will be able to meet most top level requirements set up during the project, with exception of the budget requirement. Production and further development of the test platform will exceed the budget with 7.8 million euros. The design result consists of a redesigned B737-500 with the following modifications:

- The main wings will be modified such that the outboard sections contain modular skin panels, wingtips, flaps, and slats. Measurement equipment and AFC actuators can be installed in a 'plug 'n play'-like way in these modular sections. The fuel from the outboard tanks will be reallocated and structural reinforcements will be made.
- A pylon wing was designed that is able to hold measurement equipment and AFC actuators (in a similar way as the wings). The pylon will be installed on top of the fuselage and can be used to represent swept, straight and unconventional wing configurations.
- The interior will be stripped. It has been redesigned such that it will be able to accommodate the supply systems for the AFC actuators and measurement equipment.

The preliminary design phase is finished. The overall system configuration is defined. Schematics, diagrams, and lay-outs of the design have been included. The next step in the design process will consist of the detailed design phase. In this phase product parts will be analysed and designed thoroughly.

Bibliography

[1] NetMBA, ``Market analysis.'' http://www.netmba.com/marketing/market/analysis/, Retrieved 25-04-2013.

[2] E. Greitzer, ``Subsonic civil transport aircraft for a 2035 time frame (an industry-nasa-university collaborative enterprise),'' November 2012. Retrieved 06-05-2013.

[3] W. Gilroy, ``Boeing and notre dame enter into research agreement,'' October 2009. Retrieved 06-05-2013.

[4] R. Borek and A. Pool, *Basic Principles of Flight Test Instrumentation Engineering*. Neuilly Sur Seine, France: AGARD, 1994.

[5] Controller.com, ``1991 boeing 737-500,'' 2013. Retrieved on 16-05-2013.

[6] J. S., *System Engineering for Commercial Aircraft*. Croft Road, Hants, GU11 3HR, England: Ashgate Publishing Company, 1997.

[7] G. L. Rocca, ``A/c preliminary sizing.'' Internal publication TU Delft, April 2011.

[8] N. Meier, ``Civil turbojet/turbofan specifications,'' April 2005.

[9] E. Torenbeek, *Synthesis of Subsonic Airplane Design - Part VIII*. Delft University Press, 1976.

[10] G. L. Rocca, ``Weight estimations and iterations in a/c design.'' Internal TU Delft publication, March 2013. Lecture slide 42.

[11] T. H. G. Megson, *Aircraft Structures for Engineering Students*. 30 Corporate Drive, Suite 400, Burlington, MA 01803, USA: Butterworth-Heinemann, 2007.

[12] Boeing, *737-300/400/500 Maintenance Planning Data*. 32 Avenue of Americas, New York, NY 10013-2473, USA: Unpublished, Sep 2012.

[13] G. W. Inc., ``Power dividers, combiners & unequal splitters.'' Datasheet.

[14] R. Faye, R. Laprete, and M. Winter, ``Blended winglets for improved airplane performance,'' 2013. Retrieved June 7th, 2013.

[15] W. . Freitag and T. Schulze, ``Blended winglets improve performance,'' 2013. Retrieved June 7th, 2013.

[16] C. Brady, ``The boeing 737 technical site,'' 1999. Retrieved on 03-05-2013.

[17] G. L. Rocca, ``Requirement analysis and design principles for a/c stability & control (part 1).'' University Lecture, 2013.

[18] G. L. Rocca, ``Wing design part 2.'' University Lecture, 2011.

[19] D. Raymer, *A conceptual approach*. American institute of Aeronautics and Astronautics, 1989.

[20] ``E-3 sentry (awacs),'' 2012. Retrieved on 21-05-2013.

[21] Boeing, ``Boeing randy's journal,'' March 2007. http://www.boeingblogs.com/randy/archives/2007/03/.

[22] HacknMod.com, ``How to combine video displays into a 120 wall,'' June 2013.

[23] MaerskMech, ``Garrett vs allied signal apu fuel burn,'' March 2005.

[24] A. C. I. F. Forwarding, ``Aircraft specifications,'' June 2013. http://www.alliance-cargo.de/media/b14116ca89547745ffff81b8ac144233.pdf.

[25] B. Milton, ``World jet fuel specifications,'' 2005.

[26] D. Bein, ``Ullage protection ownership cost for kc-130j: Explosion suppressant foam (esf) vs. on-board inert gas generation system (obiggs).''

[27] PCO, ``pco.1600 cooled digital 14 bit ccd camera system.'' Datasheet for pco.1600 camera system.

[28] Quantel, ``Evergreen, a state of the air laser for piv.'' Datasheet.

[29] I. Power Equipment Direct, ``Ingersoll rand 50-hp rotary screw total air system (230v 3-phase)(150 psi),'' 2013. Retrieved June 7th, 2013.

[30] H. S. USA, ``Honeywell 6033, air cooled 15kw home standby generator,'' 2012.

[31] U. A. A. Group, ``Uiuc airfoil data site.'' Retrieved on 03-05-2013.

[32] M. Sadraey, *Aircraft Performance: Analysis*. VDM Verlag Dr. Müller, 2011.

[33] D. Shevell, *Fundamentals of Flight*. Prentice Hall, 1988.

[34] T. Young and B. Humphreys, ``Liquid anti-contamination systems for hybrid laminar flow control aircraft--a review of the critical issues and important experimental results,'' *Journal of Aerospace Engineering*, vol. Part G, pp. 267--277, April 2004.

A
Individual contributions

The distribution of all tasks performed during the project can be found in Table A.1. It should be noted that not all tasks have a section or chapter in a report. The task description follows the same fashion as the Gantt chart. The names of the group members have been abbreviated as follows:

- **SB**: Simon Bal
- **FB**: Floris van de Beek
- **ID**: Irfan Dogan
- **SH**: Sander van der Horst
- **SJ**: Stevie-Ray Janssen
- **AK**: André Kil
- **TO**: Tom Onnink
- **KR**: Krispijn te Riet
- **BR**: Bob Roos
- **JS**: Jocelijn Steiner

Table A.1: Work distribution among group members

Task	**SB**	**FB**	**ID**	**SH**	**SJ**	**AK**	**TO**	**KR**	**BR**	**JS**
Project setup										
MNS and POS										x
Organisational structure						x				
Procedures and regulations			x			x				x
WBS	x					x				
Project phasing and WFD	x					x				
Gantt chart			x			x				x
Schedule risk analysis										x
Sustainable-development strategy										x
Mission analysis										
Initial literature study										
Aircraft design								x		x
Flow control systems				x			x		x	
Test flight procedures	x	x	x		x	x				

Task	**SB**	**FB**	**ID**	**SH**	**SJ**	**AK**	**TO**	**KR**	**BR**	**JS**
FBS and FFBD	x					x				x
Requirements analysis	x	x	x	x	x	x	x	x	x	x
Requirements validation		x				x				
DOT										
Aircraft design								x		
Swept/straight wings										x
AFC actuators				x			x		x	
Measurement system	x	x	x	x	x					
Data handling					x	x				
Integration of equipment system						x				
Cg manipulation		x							x	
Market analysis							x			x
Risk assessment			x						x	
Create contingency plan			x						x	
Resources breakdown					x		x	x	x	x
Sustainable-development strategy	x				x					
Conceptual design										
Define interfaces			x				x			
Create N2 chart			x				x			
Concept generation										
MD 87		x	x							
MD 90	x			x						
Boeing 737-500					x		x			
Boeing 737-400						x				x
Boeing 737-300								x	x	
Trade-off criteria	x	x	x	x	x	x	x	x	x	x
Perform trade-off	x	x	x	x	x	x	x	x	x	x
Report concepts	x	x	x	x	x	x	x	x	x	x
Reporting final concept		x					x	x		
Int/Ext configuration		x					x			
Stakeholder request research	x			x				x	x	x
Sensitivity analysis	x		x							x
Risk assessment	x		x				x			x
N2 chart revision	x		x							x
Method followed and organisation			x			x				x
Sustainable development strategy revision		x	x			x				
RAMS						x		x		x
Flight test manual					x				x	
Project organisation revision						x				x
Preliminary design										
Resources breakdown										
Power budget						x				x
Cost budget		x						x		
Weight budget				x				x	x	
Aerodynamic Characteristics	x						x			
Prepare tools design		x								
Wing design										
Structural design		x			x				x	
Module integration	x	x			x				x	
Example module		x							x	

Task	SB	FB	ID	SH	SJ	AK	TO	KR	BR	JS
Wiring		x								x
Reporting		x			x				x	x
Pylon design										
Structural design			x	x					x	
Influence on stability and controllability			x	x						
Integration	x		x	x						
Example module				x						
Reporting			x	x						
Interior design						x				x
Electrical block diagram						x				x
Data handling block diagram						x				
Aircraft internal lay-out design										
Cabling						x				x
Engineering booth/Data verification possibilities						x				x
Power						x		x		x
Pressure								x		
PIV				x						x
Fuel system						x				x
Pylon integration	x		x	x						
Reporting						x		x		x
Compliance matrix	x						x	x		
Second aerodynamic characteristics	x						x			
Sustainable development strategy					x	x				
Final functional analysis										
Function breakdown structure						x				
Function flow block diagram						x				
Final concept configuration		x								
Final resource breakdown								x		
Operations and logistics concept description										
Test plan update				x		x				
Panel integration manual									x	
Verification and Validation	x	x	x	x	x	x	x	x	x	x
Risk assessment	x									
Design summary		x					x			x
Project development strategy						x				x
Final Gantt chart						x				x
Editing Final Report	x	x	x	x	x	x	x	x	x	x
Preparing Final Review	x	x	x	x	x	x	x	x	x	x
Jury summary			x							x
Poster preparation		x			x					
Executive summary	x						x			

Faculty of Aerospace Engineering
Kluyverweg 1
2628 HS Delft
The Netherlands